AD NUMBER
AD378363
CLASSIFICATION CHANGES
TO: unclassified
FROM: confidential
LIMITATION CHANGES
TO: Approved for public release, distribution unlimited
FROM: Distribution: Further dissemination only as directed by Federal Aviation Agency, Washington, DC 20553, Sep 1966. or higher DoD authority.
AUTHORITY
FAA ltr, 10 Oct 1972; FAA ltr, 10 Oct 1972

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DEPENDABLE ENGINES

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PWA FP 66-100
SEPTEMBER 1966
C28-6C 2360. 11 C

~~(UNCLASSIFIED TITLE)~~

(6) ENGINE PROPOSAL
FOR PHASE III OF THE
SUPERSONIC TRANSPORT DEVELOPMENT PROGRAM,

VOLUME III,
TECHNICAL / ENGINE.

REPORT B,

ENGINE DESIGN (11).

(11) Sep 66

(12) 545p.

(COMPETITIVE DATA)

(15) FA-SS-66-8

PREPARED FOR
FEDERAL AVIATION AGENCY
OFFICE OF SUPERSONIC TRANSPORT DEVELOPMENT
WASHINGTON, D. C.

(14) PWA-FP-66-100-Vol-3B

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CONTENTS

CONTENTS (Continued)

SECTION I
INTRODUCTION

A. DESIGN OBJECTIVES

The Pratt & Whitney Aircraft JTF17 engine was designed to provide a high performance, reliable and long-life propulsion system for the Supersonic Transport. The engine is a two-spool, moderate bypass ratio, duct-heating turbofan. The engine performance is defined in detail in Engine Model Specifications 2698A and 2710 for the Lockheed and Boeing installations, respectively. The component designs are explained in Section II - Major Components. The primary design objectives for the engine, which were selected to meet the supersonic transport mission requirements, are as follows:

1. High Performance
   a. SLTO Thrust 61,000 lb
   b. SLTO SFC 1.77
   c. Cruise Thrust 16,000 lb
   d. Cruise SFC 1.58 (at Mach 2.7 and 65,000 ft altitude)
2. Efficient Use of Structure and Materials
   a. Basic Engine Weight 9860 lb (-21L); 9910 lb (-21B)
3. Reliability, Structural Integrity and Safety
4. Engine Life
   a. Major Cases - 50,000 hours
   b. Disk - 20,000 hours
   c. Easily Replaceable Parts - 10,000 hours
   d. Hot Section Inspection - 5000 hours.
5. Superior Growth Potential
6. Acceptable Noise Levels (Less Than 116 PNdb at Takeoff)
7. Versatility to Match Aircraft Requirements
8. Advanced Concepts to Facilitate Airline Maintenance
9. Minimum Development Risks

B. SUPERSONIC TRANSPORT MISSION REQUIREMENTS

To establish requirements, mission profiles (shown in figure 1) were generated based on JTF17 engine performance for typical aircraft L/D and gross weight and appropriate ground overpressure criteria, using mission requirements and average mission lengths (1455 statute miles for domestic flights and 1980 statute miles for international flights) defined by the economic ground rules contained in FAA Report SST 66-3. The international mission was selected for cruise life criteria, because a large percentage

of the total flight time is at cruise. The domestic mission was selected for mission cycle criteria, because the shorter mission requires a greater number of cycles for a given airframe operating time. The criteria for hot-day and transient operation as a percentage of cruise hours were established by the airframe manufacturers and are defined by the engine model specifications. The selection of time percentages for reverse thrust, and the number of engine operating cycles for idle to SLTO power were based on expected service operating techniques and previous engine experience. The normal operating envelopes are shown in figures 2 and 3 and the reverse thrust envelopes in figures 4 and 5.

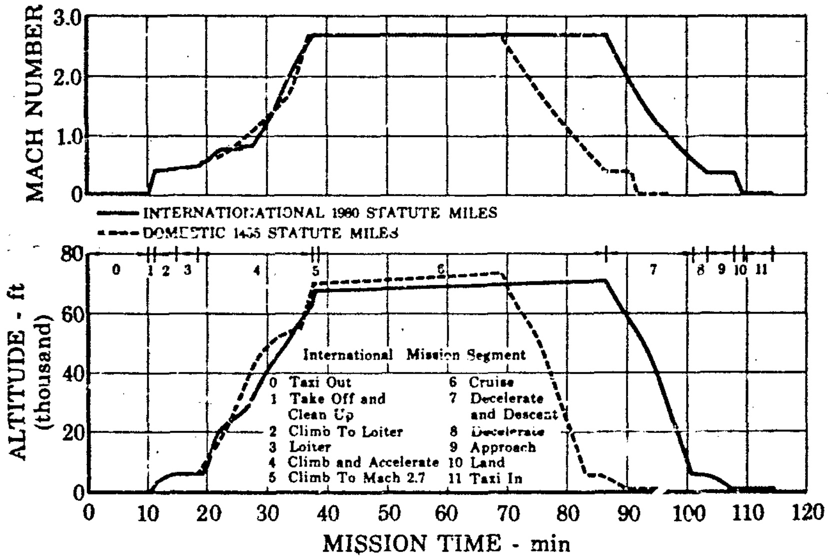

Figure 1. Average Mission Profiles FD 16381 BI

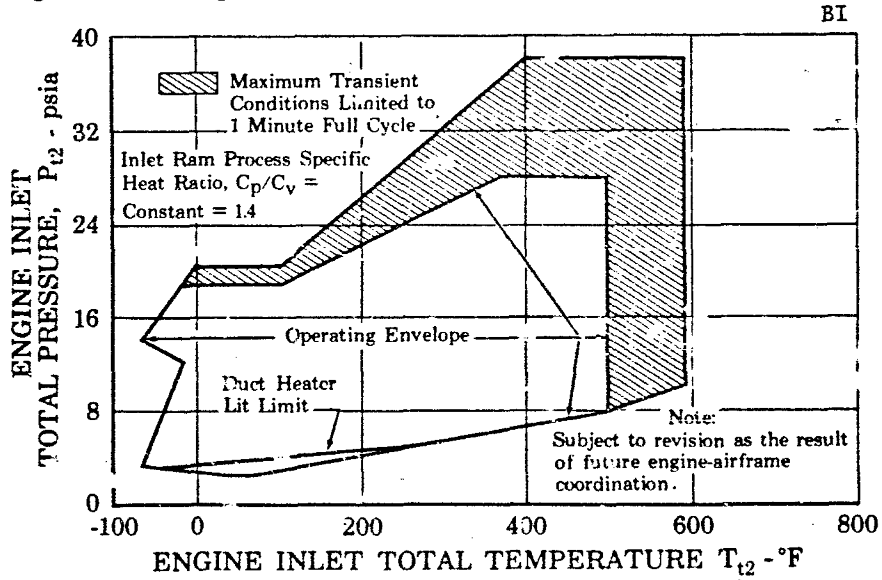

Figure 2. Engine Operating Envelope - Boeing FD 16330 BI

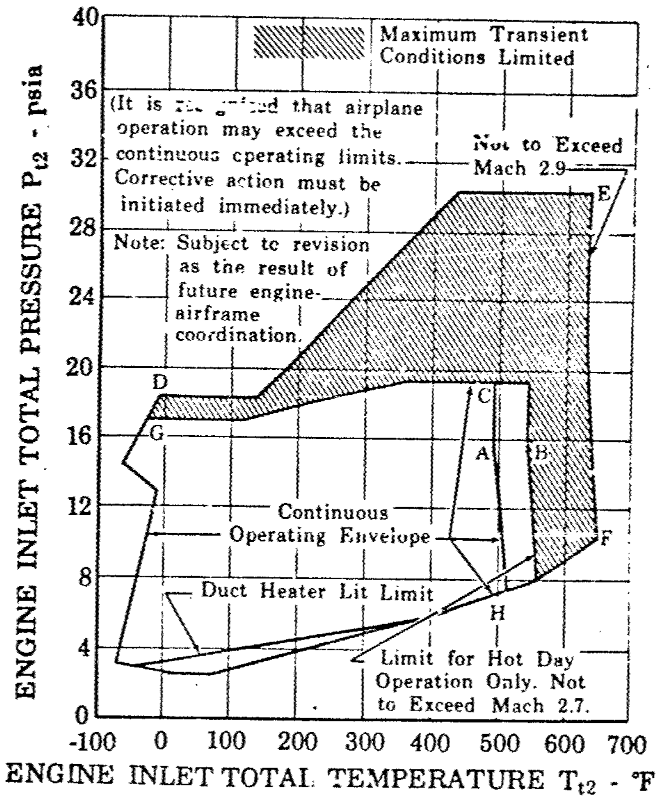

Figure 3. Engine Operating Envelope - Lockheed FD 17609
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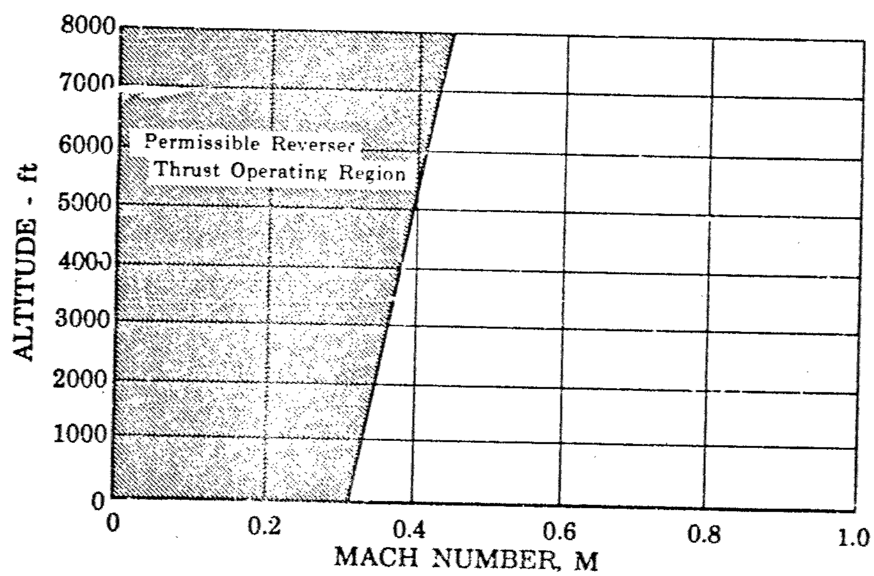

Figure 4. JTF17 Reverser Thrust Operation - Boeing FD 16332
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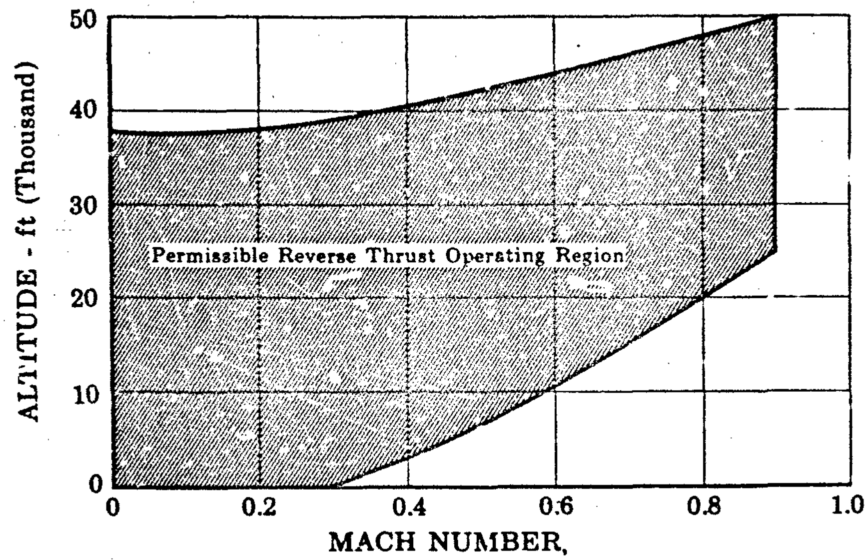

Figure 5. JTF17 Reverser Thrust Operation - Lockheed FD 16331 BI

Time percentages established were applied to the engine life objectives to obtain the requirements given below. Points indicated for various limiting conditions refer to figures 2 and 3.

For major cases, 50,000-hour life:

1. 30,000 mission cycles
2. 50,000 engine acceleration cycles - idle to SLTO
3. 21,000 hours at cruise (Point A)
4. 1000 hours at SLTO (M = 0-0.3)
5. 1500 hours at maximum compressor inlet temperature and pressure (Point C)
6. 750 hours at envelope limit (Line GCAH)
7. 50 hours at maximum transient point (Line DEF)
8. 4250 hours at hot day cruise (Point B)

For disks, 20,000-hour disk life:

1. 12,000 mission cycles
2. 20,000 engine acceleration cycles - idle to SLTO
3. 8500 hours at cruise (Point A)
4. 400 hours at SLTO (M = 0-0.3)
5. 600 hours at maximum compressor inlet temperature and pressure (Point C)

6. 300 hours at envelope limit (Line GCAH)
7. 20 hours at maximum transient point (Line DEF)
8. 200 hours at maximum reverse thrust
9. 1700 hours at hot-day cruise (Point B)

For 10,000-hour life of easily replaceable engine parts:

1. 6000 mission cycles
2. 10,000 engine acceleration cycles - idle to SLTO
3. 4250 hours at cruise (Point A)
4. 200 hours at SLTO (M = 0-0.3)
5. 300 hours at maximum compressor inlet temperature and pressure (Point C)
6. 150 hours at envelope limit (Line GCAH)
7. 10 hours at maximum transient point (Line DEF)
8. 100 hours at maximum reverse thrust
9. 850 hours at hot-day cruise (Point B).

The minimum design requirements consistent with the 5000-hour hot section inspection are:

1. 3000 hours at full turbine temperature
2. 10,000 engine acceleration cycles - idle to SLTO.

Rotor speeds used for structural design of disks and blades for both engine models are shown in figures 6 and 7. Curve A in each figure represents the scheduled design rotor speed as a function of compressor inlet temperature. Curve B is the maximum overspeed expected due to control tolerance. For the low compressor rotor, curve B also represents overspeed expected from pilot correction for performance deterioration resulting from erosion and dirt accumulation. Curve C for the low compressor represents the maximum overspeed expected as a result of control malfunction.

C. PHASE III DESIGN TASKS

The major design tasks to be achieved in Phase III are:

1. To improve the design whenever development testing indicates that design objectives or mission requirements are not met.

2. To provide design support for rig, development, ground and flight testing, to use the feedback from these tests for design improvements, and to establish a production engine design.

3. To create alternate designs where supporting data are not adequate.

4. To incorporate new materials and processes which result from research programs. (Refer to Report F.)

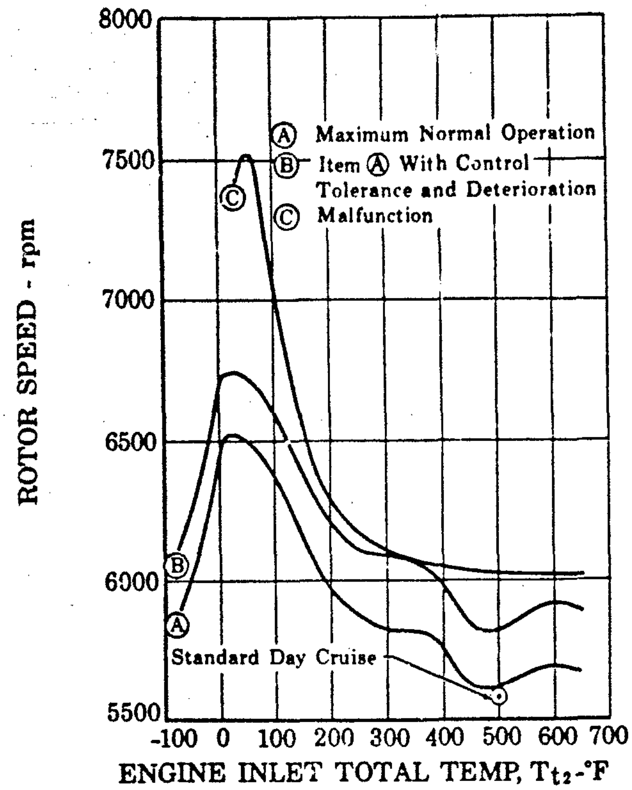

Figure 6. JTF17 Low Rotor Speed FD 16337 BI

D. BACKGROUND

The design of the JTF17 engine has been firmly based on experience which Pratt & Whitney gained from more than 39,000,000 hours of subsonic commercial turbine engine operation and from extensive Mach 3+ operation of the J58 engine in the YF-12A and SR-71 aircraft. With this experience, time between overhauls on the JT3D engine has steadily increased until now it is 8000 hours in airline operation. Valuable know-how has been gained in operating production J58 engines at Mach 3 and above.

The Pratt & Whitney Aircraft design and development engineering team assigned to the Supersonic Transport Engine Program is particularly experienced in:

1. Mach 3 cruise turbojet engine (J58)
2. Subsonic commercial turbojet and turbofan engines (JT-3C, JT-3D, JT-4A, JT-8D)
3. High temperature turbine design (J58 2000+°F).

The application of this experience should pay off in the development of a reliable, long-life engine, which fully meets the requirements of the supersonic transport.

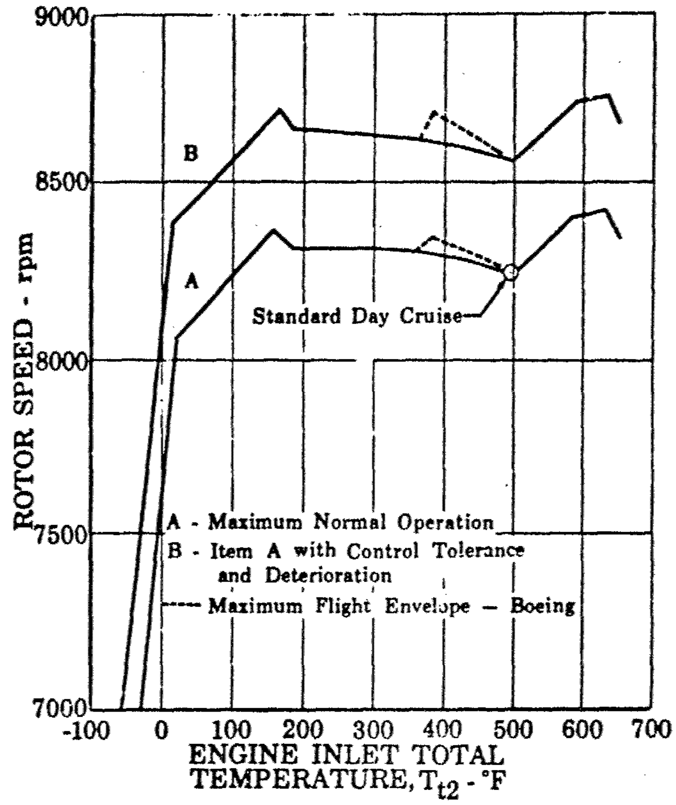

Figure 7. JTF17 High Rotor Speed FD 16338 BI

E. DESIGN APPROACH

The design objectives for the Supersonic Transport engine represent a new plateau in performance. No engine on the shelf now will fulfill those objectives. Pratt & Whitney Aircraft is taking full advantage of its extensive design and development experience in turbine engines, but is using advanced design concepts and improved materials and fabrication processes to achieve the required performance and engine life. Because the supersonic transport pays off ultimately on low operating and maintenance costs and safe, reliable operation, the design approach has been to stress low fuel consumption, reliable operation, and long life. The use of an annular ram-induction burner for the primary combustor and duct heater and advanced compressor and turbine designs permit a four-bearing rotor system that increases reliability by reducing the number of bearing compartments, seals and rotor supports. The relatively short major components of the engine are unitized designs that facilitate balancing and permit ready replacement, thus contributing to ease of maintenance. Low fuel consumption is achieved by the selection of an efficient, high-temperature cycle and good efficiencies in the compressor and turbine designs.

The use of improved materials and advanced cooling concepts has ensured that this engine will meet the long-life objectives outlined in preceding paragraphs.

This Design Report contains substantiating data on the design of the engine and major components. A description of the component, particular design objectives and requirements, and the criteria and design approach to meet these goals are presented. A summary of product assurance considerations for each component is included. Although performance is presented in Report A of this Volume and is not repeated here, it is closely related to the design and should be considered with it.

F. ENGINE DESCRIPTION

The JTF17 engine is a two-spool, duct-heating turbofan engine designed to meet the requirements of the supersonic transport. It has an airflow of 687 lb/sec and furnishes 61,000 lb of thrust and 1.77 SFC at sea level takeoff conditions. The engine we propose for the Phase III program is substantially identical to the demonstrator engine designed and tested in the Phase II-C program. The engine is designed to cruise at Mach No. 2.7 at 65,000 feet altitude and is designed to give 16,000 lb of thrust and 1.58 SFC at those conditions. The weight of the engine is 9860 lb (-21L); 9910 lb (-21B) and the external dimensions in inches of the engine are as follows:

	Length	Height	Exit Diameter
Boeing	212	95	80
Lockheed	212	90	80

The major components are described briefly in the following paragraphs of this section and more detailed descriptions are provided in Section II. A cross-sectional view of the engine is shown in figure 8.

There are no design differences between the prototype and production engines for either the Boeing or Lockheed aircraft except those defined in P&WA Engine Model Specifications 2698A and 2710, except to meet the two airframe contractors' installation requirements.

1. Fan and Compressor

The two-stage overhung fan is driven by a two-stage turbine and produces a Sea Level Takeoff (SLTO) pressure ratio of 2.9. The use of an overhung fan eliminates the conventional inlet guide vane structure and the inherent anti-icing problems associated with inlet guide vanes. It also provides a more durable compressor section by increasing resistance to damage by foreign object ingestion. The six-stage compressor is driven by a single-stage, high-work turbine and produces a pressure ratio of 4.8 for an overall engine pressure ratio of 12.95. The variable inlet guide vane for the high pressure compressor permits optimum compressor performance over the required operating range and also serves as a windmill brake when required.

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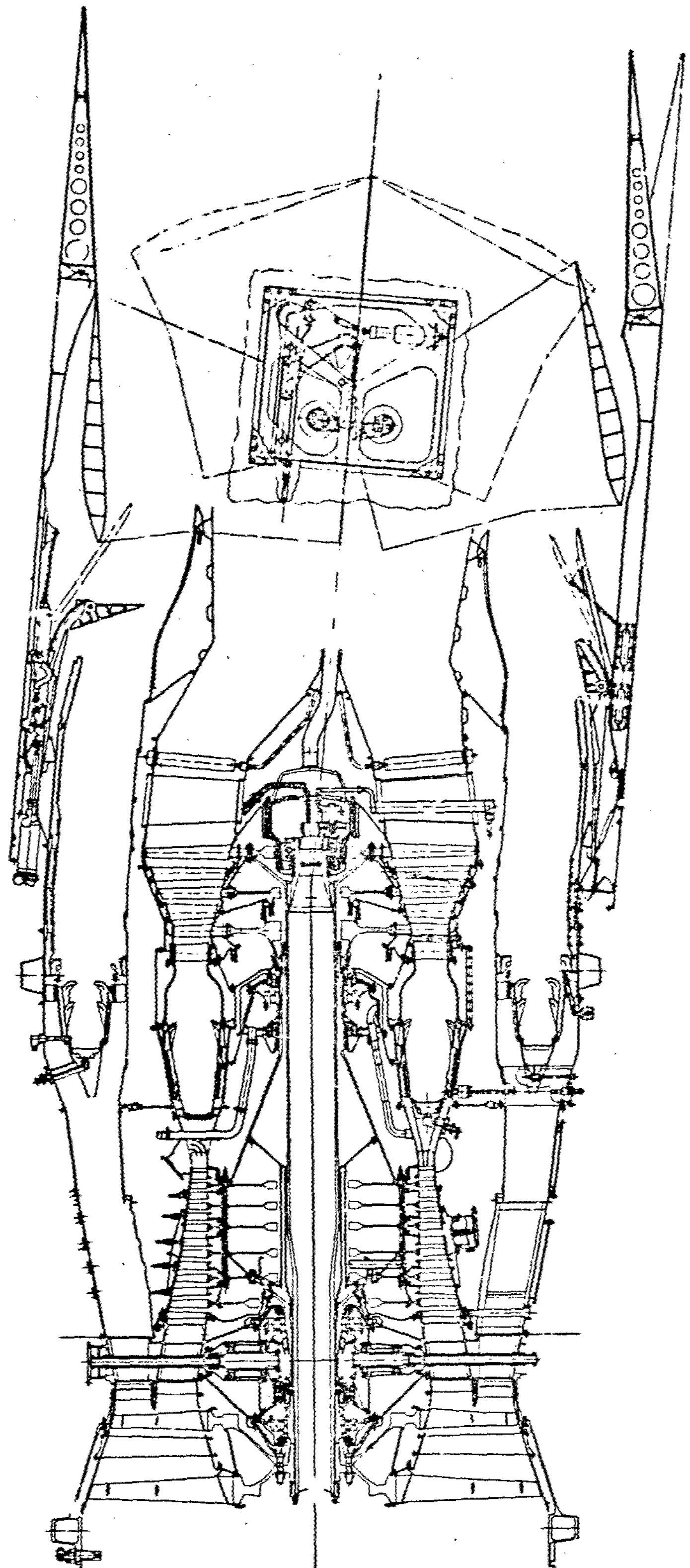

Figure 8. JTF17 Cross Section

2. Burners

The annular ram-induction primary combustor and duct heater have already demonstrated their ability to attain high heat release rates with high efficiency, uniform temperature profile, low pressure drop, and smooth ignition characteristics. A J58-type, balanced-flap nozzle is used to control fan back pressure and duct airflow for inlet matching.

The concept of duct augmentation makes it possible to use relatively cool fan air for augmentor combustion. Use of low temperature air facilitates cooling of the burner parts, thus increasing burner life, and the cool air forms a blanket to isolate hot burner surfaces from the outer case structure. Airframe nacelle and structure adjacent to the engine also receive the benefits of lower engine outer case temperature.

3. Turbine

A single-stage high-pressure turbine drives the six-stage compressor. Advanced aerodynamic design and proven airfoil and disk cooling techniques have been incorporated to achieve the efficiency and long life required. A two-stage, controlled-vortex low pressure turbine drives the two-stage fan. The controlled-vortex design permits higher efficiencies than the conventional free vortex design. Highly effective convective cooling techniques have been incorporated in all three stages of the turbine to optimize performance and enhance the life of the turbine disks and blades. High creep strength material (IN-100) and advanced coating (PWA 64) will give the turbine blades the long life required in the supersonic transport engine.

4. Bearing System

A four-bearing rotor system has been designed to meet the design goal of 10,000 hours life. There are two single-row, balanced cage construction bearings supporting each rotor shaft. This arrangement is made possible by the short burner design. The bearing design and materials have been developed in extensive rig and bearing tests.

5. Reverser-Suppressor

The reverser-suppressor, which includes the engine exhaust nozzle, performs three basic functions: (1) the controlled expansion of the exhaust gases at all flight conditions for optimum overall airplane performance; (2) redirection of exhaust gases for reverse thrust; and (3) suppression of engine exhaust noise. The reverser-suppressor is comprised of five major sections: (1) the main structure, which includes the support framework, the fixed outer skin, and engine attachment ring; (2) tertiary air doors between the forward structural members; (3) an integral shroud that, with the clamshells, forms the ejector throat; (4) clamshells that form the movable portion of the ejector throat and provide exhaust gas blockage for reverse thrust; and (5) the movable exit flaps, forming the variable diameter discharge portion of the ejector nozzle.

6. Controls and Accessory Drive Systems

Accessory and external component drives are provided by three towershafts driven by the high rotor through a bevel gear system. One towershaft drives an accessory gearbox, located on top of the engine that provides power takeoffs for aircraft accessories and for the engine starter. The left side towershaft drives the main engine gearbox, on which are mounted the gas generator fuel pump, the unitized fuel control, and the engine hydraulic pump.

The gas generator fuel pump is a two-stage unit incorporating a centrifugal-type boost stage, a full-flow micronic filter, and a gear-type high pressure stage. The unitized fuel control is a hydromechanical unit that schedules the fuel flow to the primary combustor and duct heater, controls the operation of the duct heater exhaust nozzle, and schedules operation of the high rotor inlet guide vanes, the compressor starter bleeds, and the thrust reverser system. The engine hydraulic pump is a piston-type pump that provides high pressure fuel to actuate the duct heater exhaust nozzle and thrust reverser system.

The right side towershaft drives the main engine oil pump, and, for the Lockheed installation, an aircraft air compressor.

Duct heater fuel is pumped by an engine-air driven turbopump located on the right side of the engine. Engine fuel-oil coolers and an oil tank are also located on the right side of the engine. Dual low-voltage ignition systems are provided for both the primary combustor and the duct heater.

7. External Configuration

Certain differences in external configuration result from the differing requirements for the Lockheed and Boeing installations. The version of the engine for the Lockheed aircraft incorporates a 5-degree downward bend of the engine exhaust nozzle system, front mount points located 38 degrees above the horizontal centerline, and rear mount points located 45 degrees above the engine horizontal centerline. The upper power takeoff drive gearbox axis is transverse to the engine axis, and an airframe air compressor drive is provided on the right side of the engine. The main engine gearbox and associated engine components are mounted on the lower left side of the engine because of the requirements of the Lockheed engine support frame. Front, left side, and right side views are shown in figures 9, 10, and 11, and 12, 13, and 14, for Boeing and Lockheed, respectively.

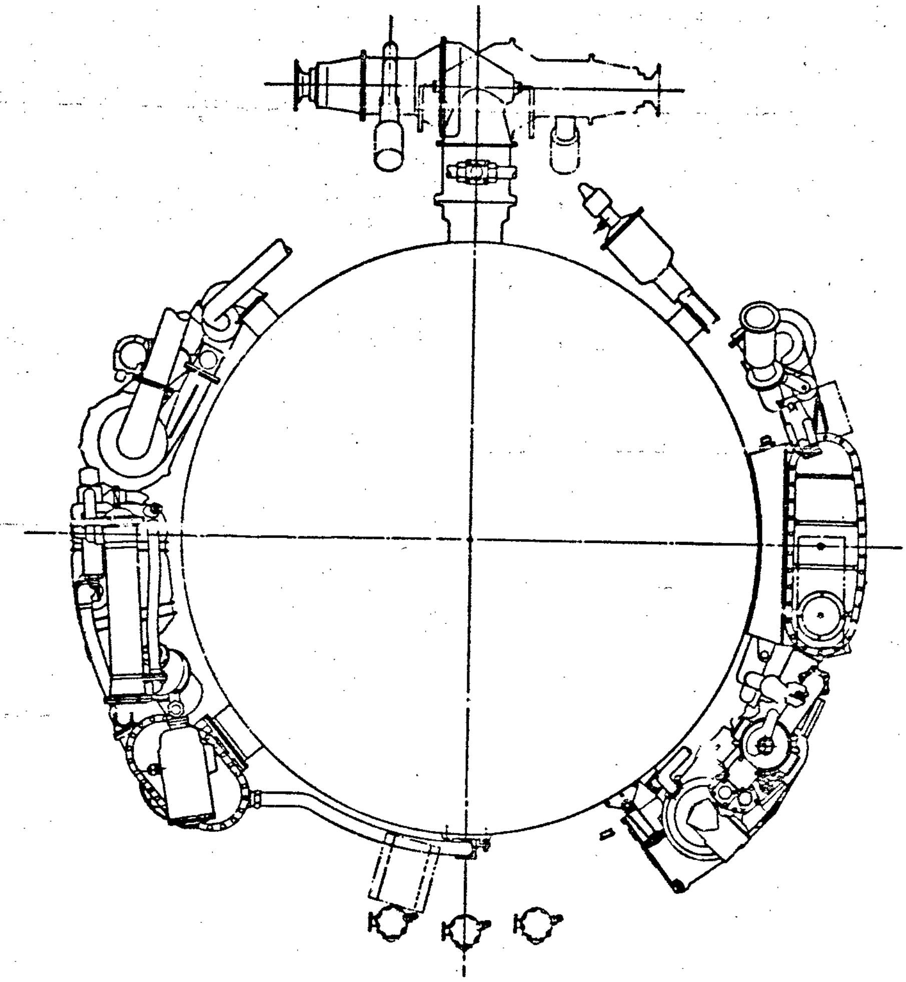

Figure 9. JTF17 Front View - Lockheed

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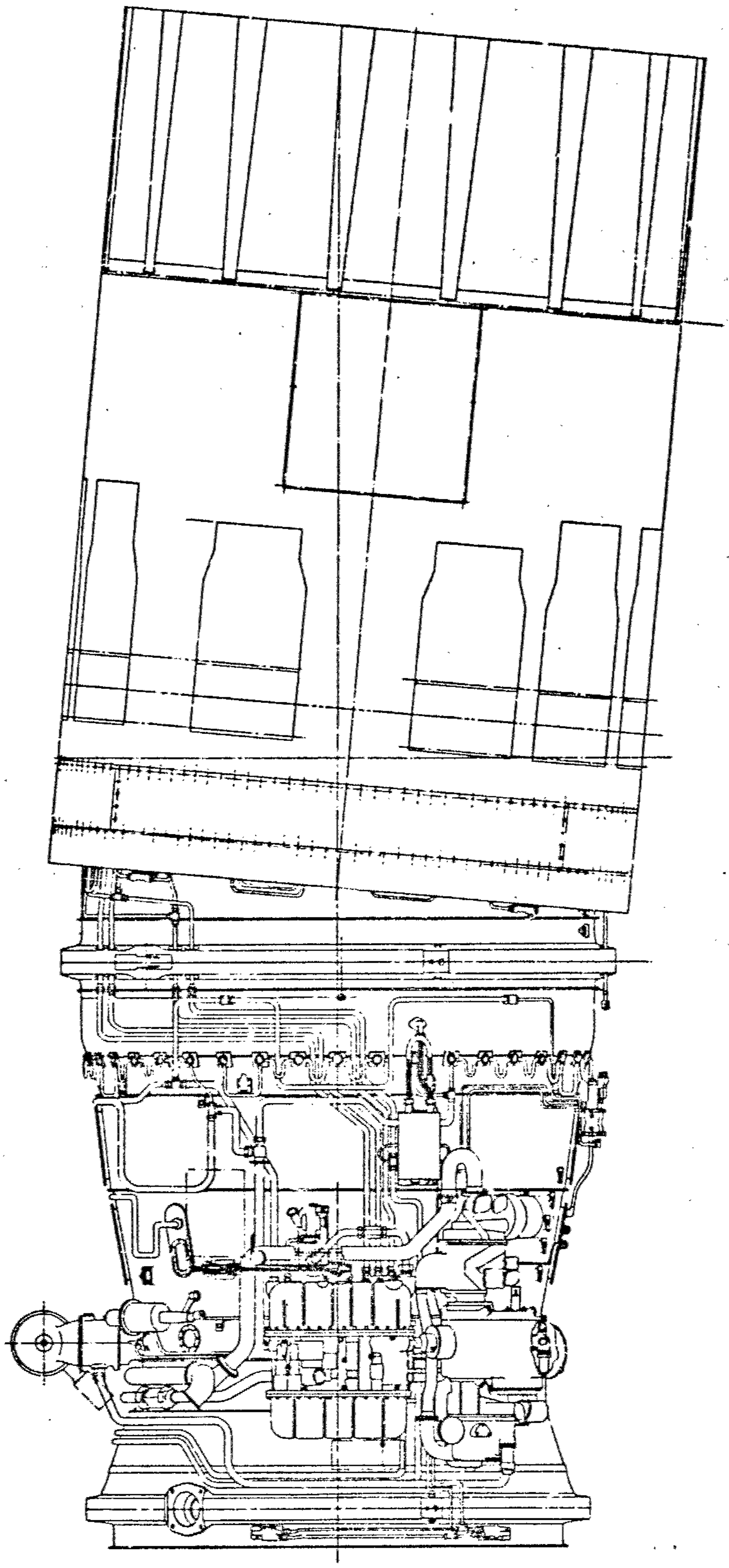

Figure 10. JTF17 Left Side View - Lockheed

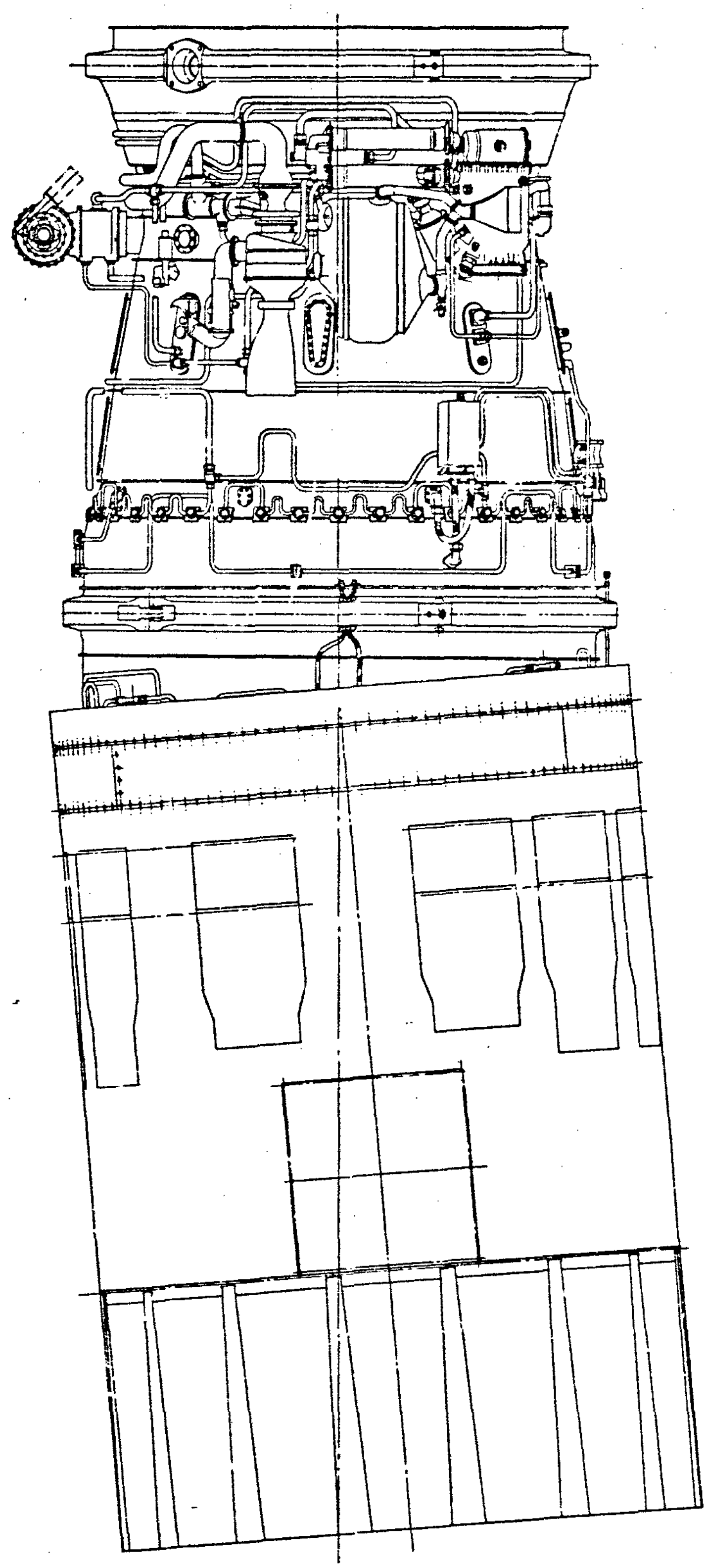

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Figure 11. JTF17 Right Side View - Lockheed

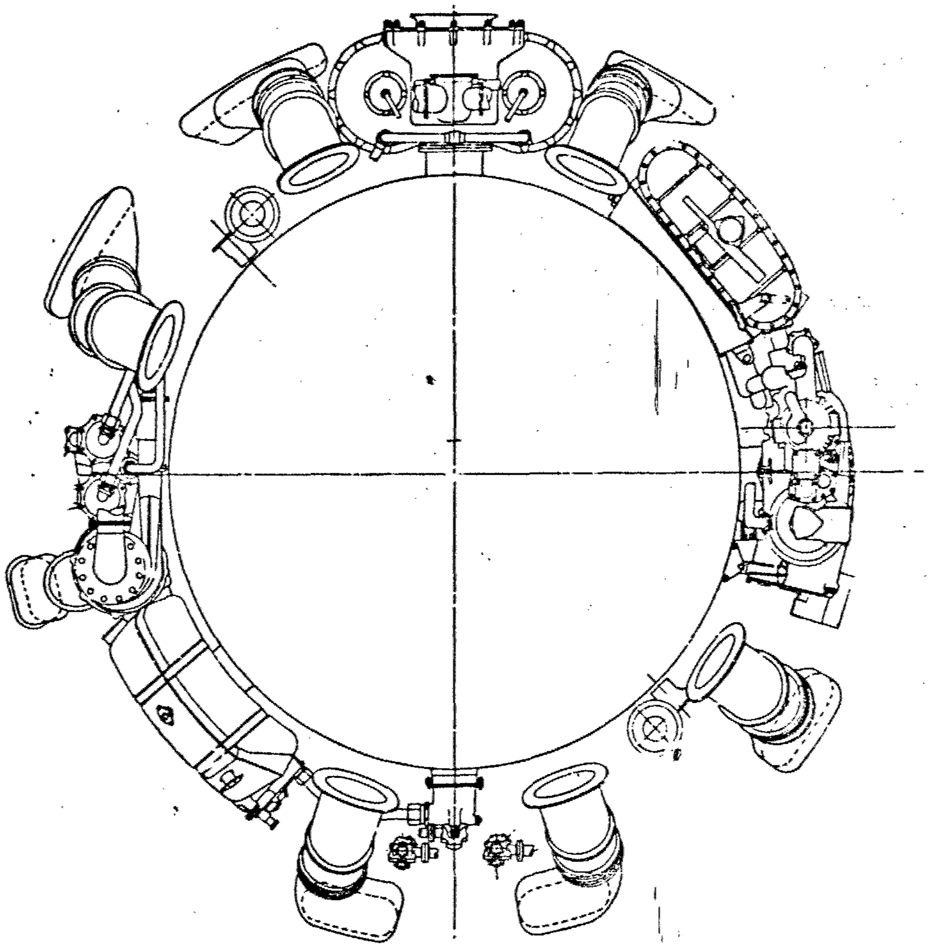

Figure 12. JTF17 Front View - Boeing

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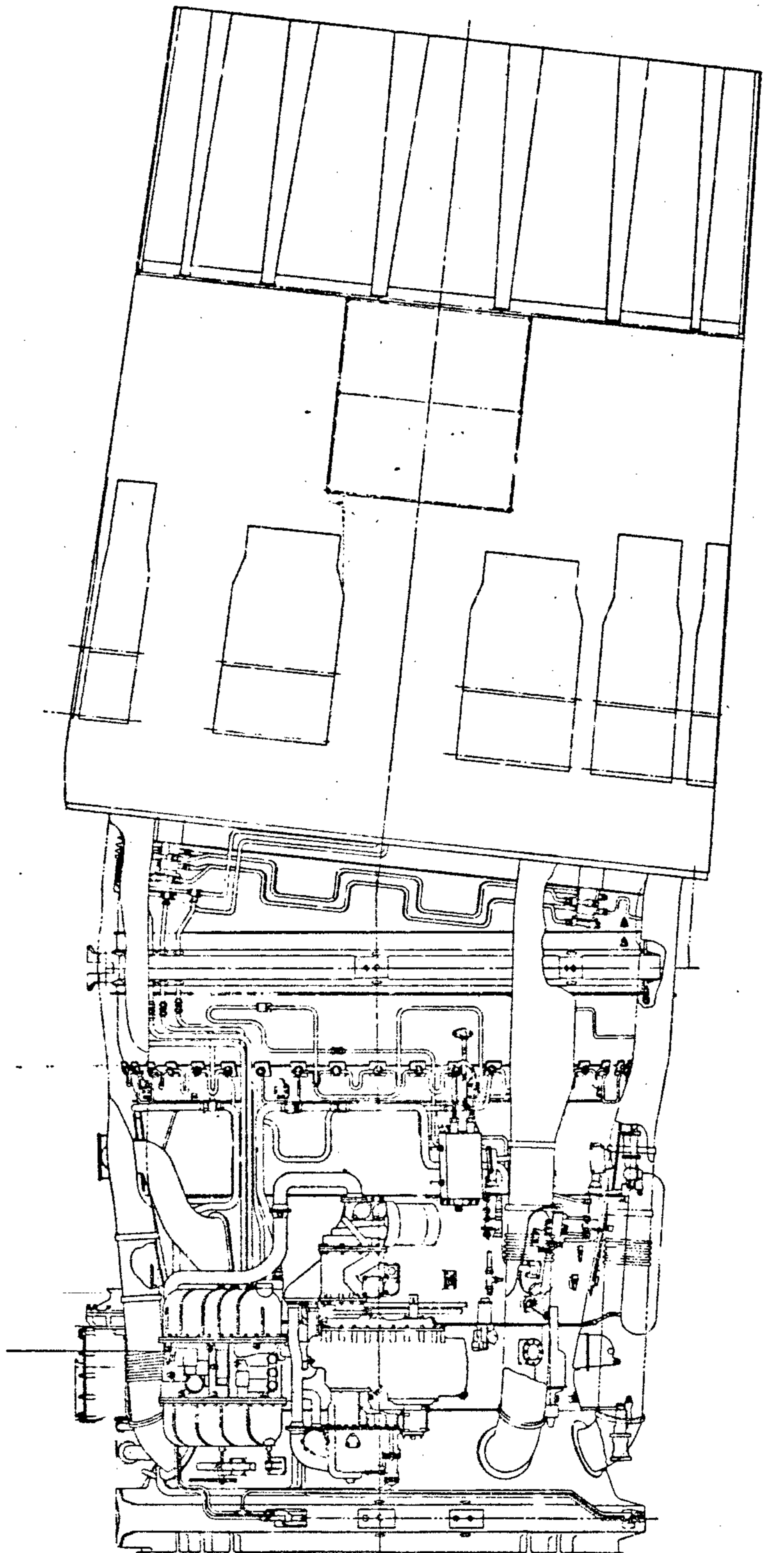

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Figure 13. JTF17 Left Side View - Boeing

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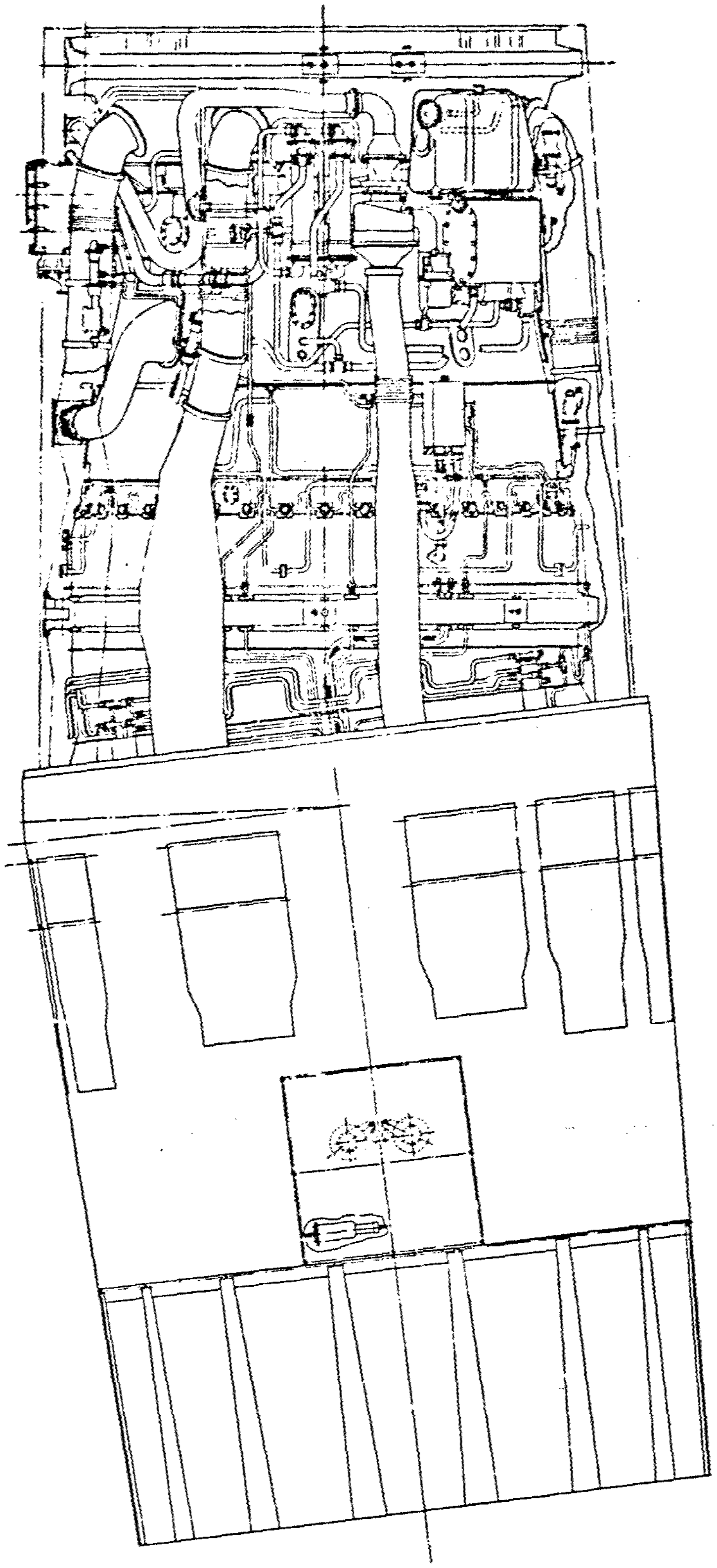

Figure 14. JTF17 Right Side View - Boeing

SECTION II
MAJOR COMPONENTS

A. FAN AND COMPRESSOR

This section will be discussed in two major parts: (1) Fan Subsection and (2) Compressor Subsection. Each subsection will be complete in itself except as follows: the general subjects of Blade Containment, Disk Yield Margin, Foreign Object Ingestion, and Blade Flutter are contained in the Fan Subsection. The general subjects of Low Cycle Fatigue and Disk Burst Margin are contained in the Compressor Subsection.

I. Fan Section

a. Design Description

(1) Function

The two-stage fan has been designed for a total airflow of 687 lb/sec to a pressure ratio of 2.90 on the duct side and 2.68 on the engine side. The fan exit geometry is arranged so the duct receives 388 lb/sec of this air with the remaining 299 lb/sec supercharging the gas generator, resulting in a bypass ratio of 1.30. The efficiency goals for this fan are 79% for the duct flow and 89% for the engine flow.

(2) Configuration (See figure 1)

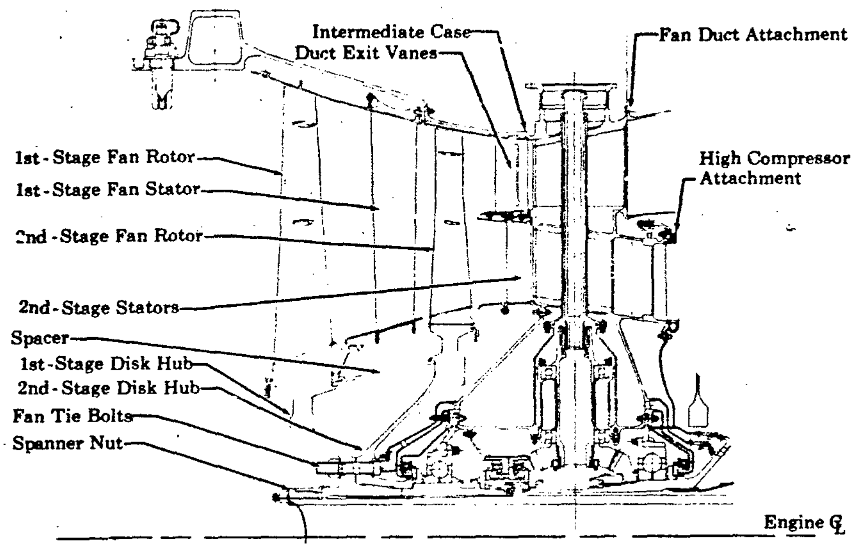

Figure 1. JTF17 Fan Assembly FD 16418 IIA

The fan rotor is overhung from the front bearing of the low-pressure spool. The rotor is a trussed-hub configuration to provide maximum stiffness with minimum weight.

The choice of this configuration was the result of P&WA's analysis of J58 engine disk low-cycle fatigue resulting from both ascent and descent from high Mach number and from acceleration and deceleration from normal operating speed. Specifically, it is necessary to:

1. Avoid bolt holes in the main disk body
2. Design a compactly shaped configuration
3. Use a material with the unique properties of titanium.

More detailed discussion of this design is presented in the Fan Material portion of the Fan Subsection and in the Compressor Disk Low Cycle Fatigue and Materials portion of the Compressor Subsection.

Overhung support of the fan rotor eliminates the need for a stationary structural vane section in front of the rotor. This in turn eliminates the requirement for an anti-icing system, eliminates a source of noise generation (close proximity of rotors and stators) and pressure loss, and results in a better foreign object ingestion capability. The intermediate case provides the thrust bearing support for both the low and high pressure spools. This case also structurally bridges the airflow path in both streams by means of eight struts. These struts house three power takeoff shafts, plus lines for oil pressure, seal air pressure, and breather lines. The 1st-stage stator vanes in back of the 1st-stage rotor are full length, whereas the 2nd-stage stator assembly is composed of two sets of shorter vanes; one set in the duct stream and one in the engine flowpath separated by the airflow splitter. Both fan blades have two part-span shrouds.

The No. 1 bearing supports forward thrust from the low (N_1) rotor shaft. The No. 2 bearing supports rearward thrust from the high (N_2) rotor shaft.

The intermediate case and bearing compartment is arranged to permit easy access to both main thrust bearings and seals from the front of the engine by removing only the fan rotor and 1st-stage stator.

A bevel gear drive from N_2 provides output power through three radial shafts to external accessory gearbox drives.

b. Design Objectives and Requirements

1. High performance and efficient utilization of structure and materials
2. High degree of reliability, structural integrity, and safety
3. Minimum noise generation
4. Engine part life consistent with 50,000 airframe hours
5. High degree of maintainability.

c. Design Approach

(1) Aerodynamic Design

The two-stage fan has been designed for a total airflow of 687 lb/sec, with a pressure ratio of 2.902 on the duct side and a 2.680 on the engine side. Blading and areas were chosen for the duct to receive 388.3 lb/sec of this air with the remaining 298.7 lb/sec supercharging the gas generator, resulting in a bypass ratio of 1.30. The sea level static efficiency goals for this fan are 79% for the duct flow and 89% for the

engine flow. This point plus the long-time cruise performance design requirements are summarized in table 1.

Table 1. JTF17 Fan Performance

	Sea Level Static (SLTO)	Cruise
Total Inlet Corrected Airflow - lb/sec	687	417
Fan Duct Corrected Airflow - lb/sec	388.3	271.9
Fan Engine Corrected Airflow - lb/sec	298.7	145.0
Fan Duct Pressure Ratio	2.902	1.620
Fan Engine Pressure Ratio	2.680	1.579
Fan Duct Efficiency - %	0.788	0.808
Fan Engine Efficiency - %	0.888	0.893
Corrected Speed	6493	4142
Bypass Ratio	1.30	1.88

The two-stage fan configuration for the JTF17 engine was selected on the basis of comparative performance and weight studies with other candidate configurations. A comparison between the selected configuration and a three-stage fan showed a higher efficiency for the two-stage fan. An efficiency improvement for the three-stage fan could only be obtained by adding a third stage to the low pressure turbine and decreasing the low spool rpm. This three-stage configuration, although capable of obtaining the required performance at sea level static, resulted in a considerable weight increase over the selected configuration. Comparison studies showed that all single-stage fan configurations required impractical wheel speeds, both from Mach number and disk weight considerations.

The aerodynamic design of the JTF17 fan was significantly influenced by three critical operating points on the flight envelope. The three points were cruise, sea level static and transonic acceleration. At the cruise operating condition fuel consumption and therefore, component efficiency level is extremely important. At the sea level static and transonic acceleration conditions thrust level is of primary concern, therefore flow capacity is important. For these reasons, the primary emphasis in the design of the fan was focused on efficiency levels at cruise and on flow capacity at sea level static and transonic acceleration conditions.

Control of two primary parameters, the local aerodynamic loadings ($\Delta P/Q$ and D factor) and local relative Mach numbers, is required in the design of any fan or compressor. The aerodynamic loadings must be maintained within established limits to prevent flow separation, whereas high local Mach numbers can result in high losses and/or flow separation. A unique method, developed by Pratt & Whitney Aircraft, was used to accurately predict the primary parameters in the design of the JTF17 fan. The method is based on fundamental aerodynamic and thermodynamic theory and design limits obtained from more than 500,000 hours of rotating turbomachinery and cascade testing. The calculation procedure accounts for full radial equilibrium equations for flow pattern evaluation, predicts losses and turning capabilities of the blading, and is used to evaluate the "off-design" performance of the

fan. The method has been programmed into high speed computers which allow rapid and accurate tradeoff evaluations to be made between various design parameters.

The critical aerodynamic parameters which describe the JTF17 fan are listed in table 2. A stream tube distribution sea level takeoff and cruise operating conditions are shown in figure 2. The local aerodynamic loadings are shown for the most severe conditions and compared with demonstrated capabilities of similar tested fan stages. These data indicate that the aerodynamic loading levels can be achieved in the JTF17 fan. (See figures 3 and 4) The local relative Mach numbers are compared with values from other fan stages (figure 5), including supporting rig data from Phase II-C. Further substantiating data from the Phase II-C rig testing program may be found in the Fan Performance Section of this proposal (Section III, Volume III, Report A).

Table 2. JTF17 Fan Performance Parameters

Hub/Tip Ratio, 1st	0.383	
Hub/Tip Ratio, 2nd	0.538	
Tip Diameter - ft	4.98	
Number of Stages	2	
	SLTO	CRUISE
Specific Flow - $lb/sec/ft^2$	41.2	25.0
Average Pressure Ratio	2.840	1.600
Stage Pressure Ratio, 1st	1.712	1.283
Stage Pressure Ratio, 2nd	1.653	1.246
Corrected Tip Speed, 1st - ft/sec	1694	1077
Corrected Tip Speed, 2nd - ft/sec	1480	895
Stage Temperature Ratio, 1st	1.189	1.0808
Stage Temperature Ratio, 2nd	1.190	1.0800
Inlet Axial Mach Number	0.592	0.309
Duct Exit Axial Mach Number (Actual Area)	0.449	0.529

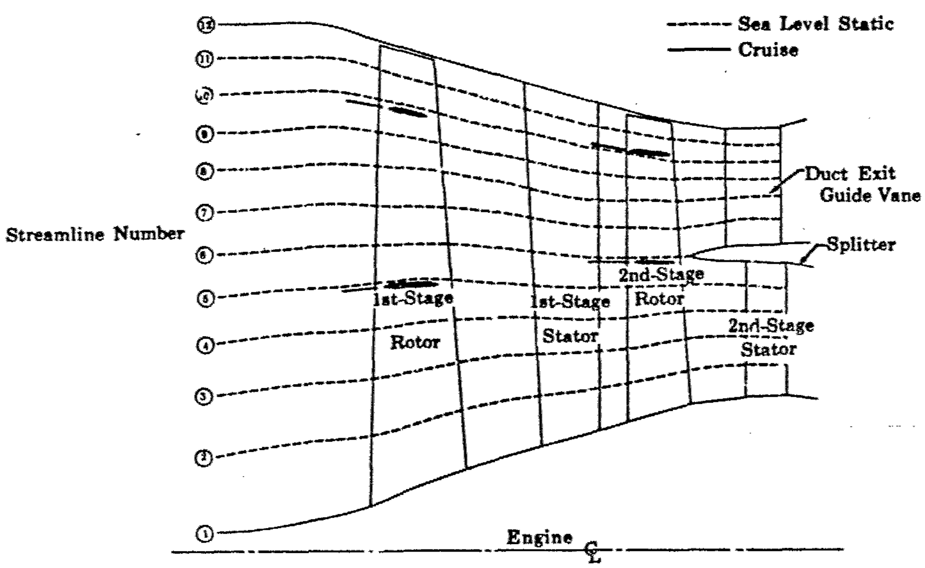

Figure 2. JTF17 Fan Flow Path Design Point Streamlines

FD 16550
IIA

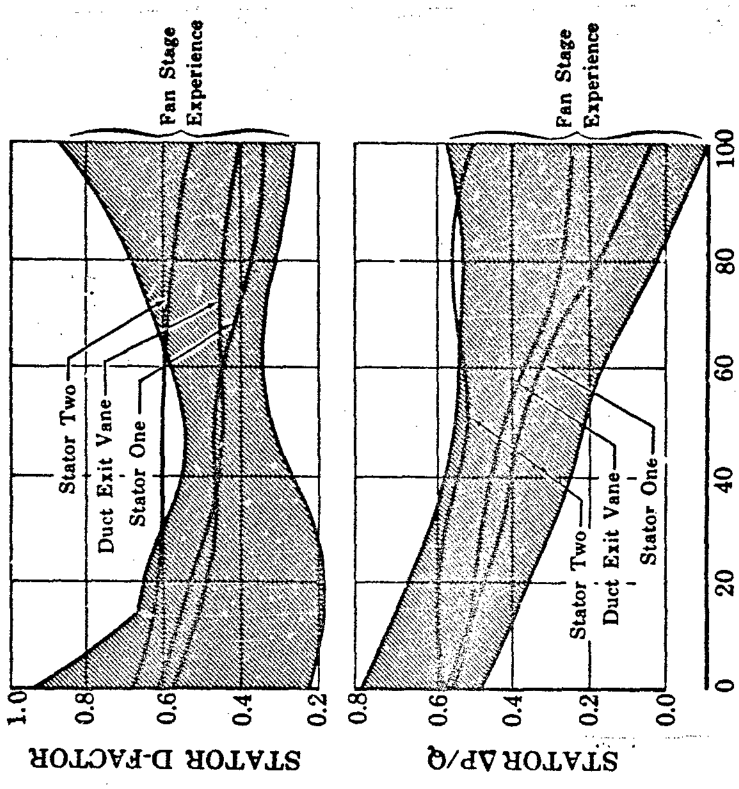

Figure 4. JTF17 Stator Aerodynamic Loading FD 17500 IIA

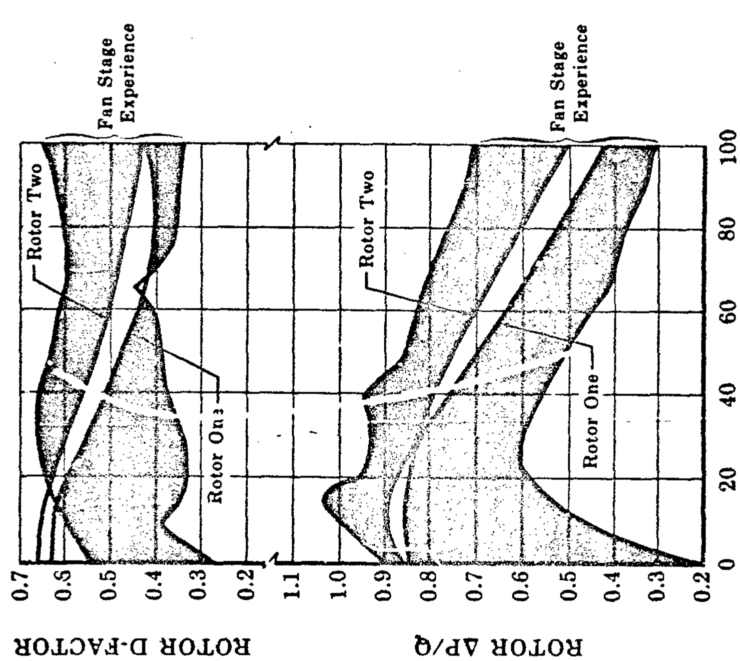

Figure 3. JTF17 Rotor Aerodynamic Loading FD 16278 IIA

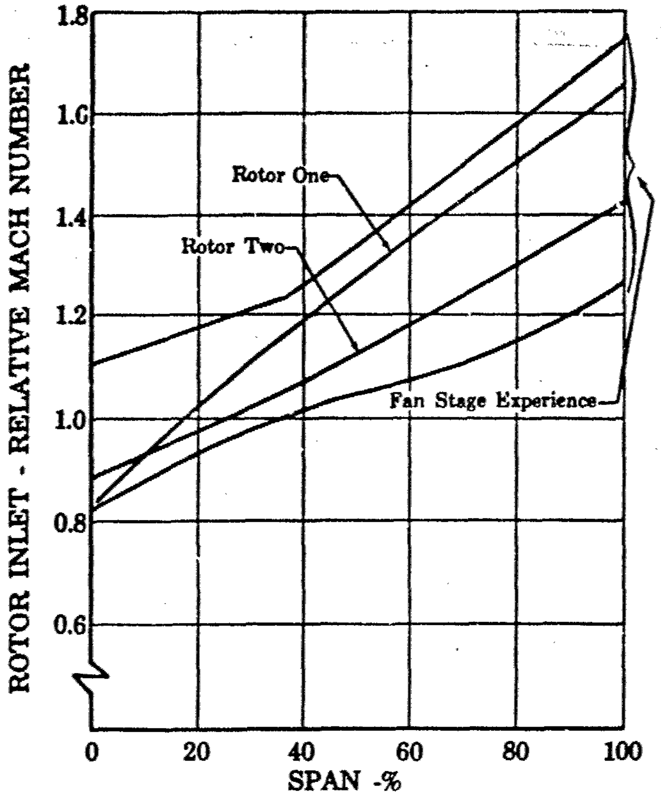

Figure 5. JTF17 Rotor Mach Number FD 17199 IIA

(2) Fan Rotor Design

The fan rotor is designed to meet the performance, reliability, and component life requirements for the environmental conditions encountered throughout the mission cycle. These conditions include maneuver loads due to G forces, gyro moments generated by angular pitch and yaw velocities, inlet temperatures and pressures, and instantaneous transient loads caused by bird ingestion and compressor surge. (See figures 6 and 7 for The Boeing Company, and figure 8 for Lockheed California Company data.) The disks are designed to have 20% burst margin at maximum overspeed condition, as well as the capability of withstanding LCF stresses for 20,000 hr. The rotor is designed as a vibratory system to ensure no critical speeds or low-order disk and blade nodal frequencies in the running range. The rotor is also designed to meet certain types of failure criteria, such as a 10% blade loss, and confine the failure to relatively small masses such as blades and vanes and maintain rotor integrity.

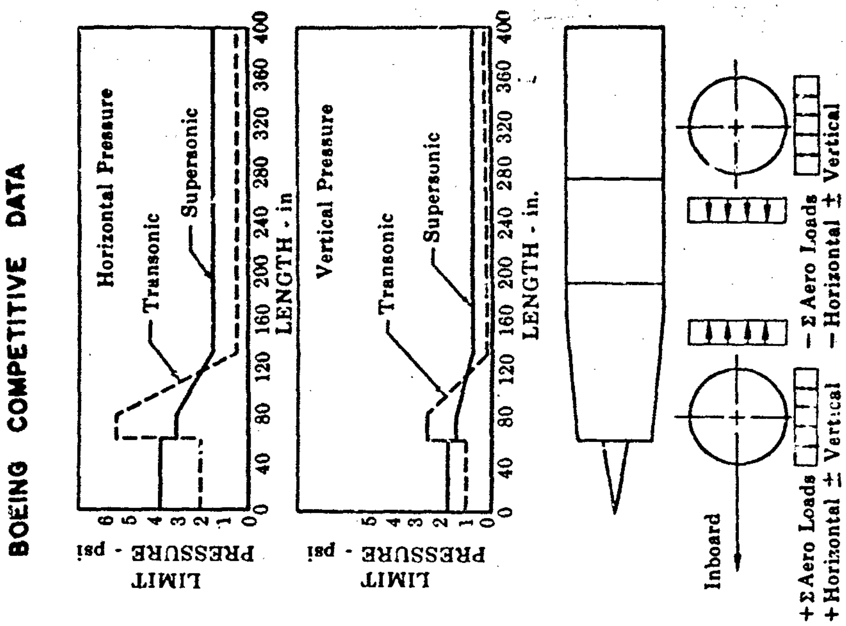

Figure 7. Aerodynamic Loads on Engine Pod (Boeing)

FD 16286
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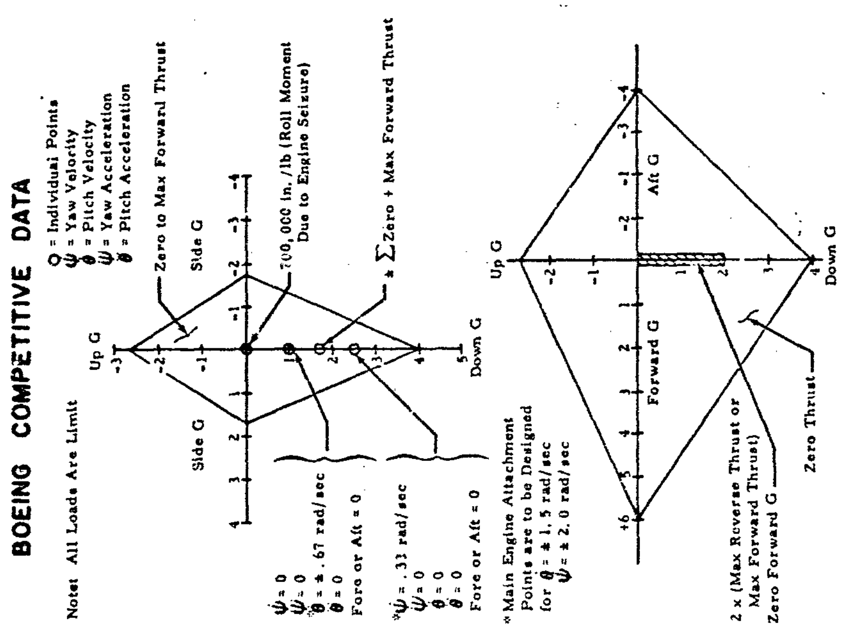

Figure 6. JTF17 Maneuver Load Diagram - Boeing Installation

FD 16287
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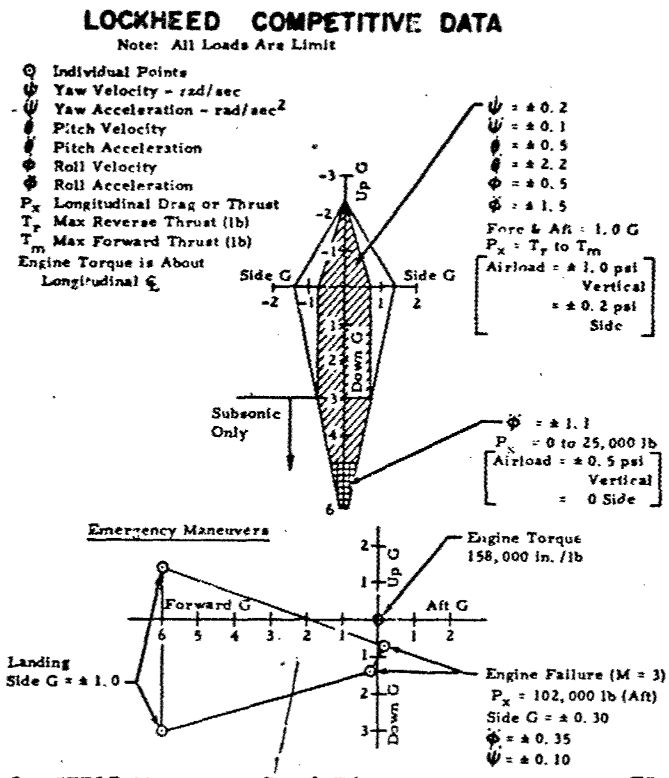

Figure 8. JTF17 Maneuver Load Diagram - Lockheed Installation FD 16285 IIA

Rotor integrity is assured by application of the following rotor tiebolt or axial retention criteria:

1. The moment imposed by loss of 10% of the blades of any one stage shall not stress the tiebolts more than the 0.2% yield strength of the material.
2. The strain energy produced from loss of 10% of the blades of any one stage must be within the strain energy absorption capability of the tiebolts.
3. In addition, the tiebolts must prevent flange separation during normal operation at the extreme conditions of maneuver and flight envelope.

Energy absorptive techniques are used to prevent gross rotor failures. Generally, this technique is to utilize energy by deformation or rubbing friction on less critical or smaller mass parts to prevent the sudden release of large masses of energy. This criteria is applied to rotor-stator axial spacing.

The minimum parts life requirements shown below are established on the basis of the general engine parts life requirements described in Section I (Introduction to Report B).

Fan disks	20,000 hours*
Blade lock rings	20,000 hours
Fan rotor blades	50,000 hours
Rotor tiebolts	10,000 hours

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The actual component life based on the detailed design is summarized in table 3, and is shown to exceed these requirements.

Table 3. Fan Component Life

Component	Stress, psi	LCF cycle	Stress Life, hr
1st-Stage Disk	52,900	25,000	>50,000
2nd-Stage Disk	51,800	22,000	>50,000
1st-Stage Blade	36,000	--	>50,000
2nd-Stage Blade	24,000	--	>50,000
Tie Bolts	115,000	--	>50,000

The fan also meets maintainability requirements, such as modular major component replacement and accessibility provisions, as formulated by Pratt & Whitney Aircraft and airline maintenance procedures. The disks, disk support cones, and rim spacer form a closed truss in which loads and deflections are interrelated. The truss was, therefore, analyzed and designed as a system. (See figure 1.) The truss configuration is advantageous in obtaining necessary critical speed margin for the rotor with a minimum weight. This margin also adds a stiffening effect to the disks to meet the disk-plus-blade vibration criteria. To assure that this effect is achieved, the truss system is designed to maintain adequate axial pinch on the rim spacer to avoid axial separation in all situations including compressor surge. The axial pinch is generated by the centrifugal load on the disks. As the disks grow, they tend to swing together on the supporting column. The axial spacer prevents this movement and a compressive load results.

The entire fan is mounted on the 1st-stage disk hub, which is splined to the low rotor shaft. This hub is double piloted; directly to the shaft in the front, and indirectly through the 2nd-stage hub to the bearing sleeve in the rear. The fan rotor is retained on the shaft by a spanner nut which applies load to the 1st-stage hub through a steel spacer. This nut also retains a long stackup of bearing and seal parts. The long stackup provides a low spring rate that makes it possible to maintain tightness at elevated temperatures when the low expansion of the titanium hub tends to reduce the load. The 2nd-stage hub is double piloted to the 1st-stage hub. The rear pilot is extended onto the bearing sleeve to act as a second pilot for the entire rotor. Tiebolts hold the two hubs together and transmit rotor torque from the second stage. In the event of a failure, such as blade loss, the second stage is not completely dependent upon the tiebolts to support the disk. Redundancy exists, considering the blade loss criteria, in that not only are the bolts designed to take the resultant moment, but the pilot diameters have adequate wheel base to give moment support.

For ease of maintainability, provisions are made for easy removal of the fan as a complete unit, for removal of the first stage only, and for removal of individual 1st-stage blades. To remove the first stage, the spanner nut and the tiebolt nuts are removed. Four ½-inch diameter holes are provided in the front hub to insert a fixture which presses against the 2nd-stage hub. The 1st-stage hub has a lip provided for

the puller. To remove the entire fan assembly, the spanner nut is removed and a puller mounted on the front hub. Tapped holes are provided in the 1st-stage fan shroud for fastening the fan assembly fixture. This fixture pilots on the bore of the 1st-stage disk with a soft metal or plastic liner, to protect the disk bore, and the bolts to the extensions of the rotor tiebolts. Bolts retaining the 1st-stage vanes to the intermediate case flange are removed, and the entire assembly is drawn off with a hub puller. An important advantage of this configuration is that the fan assembly may be removed without disturbing the front bearing, and the low rotor shaft is still supported when the fan has been removed.

The fan is initially assembled and balanced as a unit before installation on the engine. Each disk and blade assembly is statically (single plane) balanced by pairing blades of equal moment class weight 180 degrees apart, and by installing balance weights on the flange provided on the front of the first stage and on the rear of the second stage disks. The rotor is then built around the 1st-stage stator and shroud assembly. The rotor is mounted on the balance arbor, which pilots on the 1st-stage disk hub front and rear snap diameters. A fixture mount is provided on the 1st-stage outer shroud to locate the stationary parts during balance. The rotor is dynamically balanced to 0.20 ounce-inch maximum unbalance by riveting correction weights to the flanges on the disks.

Different correction weight types are used for the dynamic rotor balance than are used for disk and blade assembly balance. This provides easy identification and ensures that the individual stage balance will not be lost if the rotor is torn down and reassembled. To minimize the number of balance weights used, and to provide a check against excessive eccentricities, an initial maximum unbalance of 10 ounce-inches is required on the rotor prior to adding correction weights. To achieve the initial minimum unbalance, the first stage may be rotated relative to the second stage to place points of maximum unbalance on each disk 180° apart. When the best position has been determined, a dowel pin is pressed into one of the 16 holes on the 2nd-stage hub and aligned with the single hole on the 1st-stage hub. Thereafter, the rotor is disassembled and assembled in the same relative position. All pilot diameters are closely controlled for concentricity, and stackup is controlled for squareness to control unbalance. The rotor has enough wheel base between the hub snap diameters to assure squareness of the rotor on the shaft. The hub-to-shaft fits are designed to ensure that the hubs will remain tight at all running conditions. It is permissible to remove and replace the 1st-stage blades in pairs of equal moment weights located 180 degrees apart without rebalancing the rotor assembly. To replace 2nd-stage blades, the entire fan rotor is removed from the engine and rebalancing is recommended. The first stage may be removed from the engine, have blades replaced, be statically balanced, and be reinstalled without rebalancing the rotor assembly, provided the same disk is used. Trim balance of the rotor on the engine is permitted by the addition of weights on the 1st-stage disk flange. Each blade will be marked with the applicable moment class. In the event a damaged blade requires replacement, a comparison of the damaged blade moment class with the moment class of the blade 180 degrees apart provides a correction value. Disk rim static balance weights are then added to obtain the necessary correction.

The rotor is secured by 16 3/4-in. tiebolts of PWA 1010 (Inco 718) material. These bolts maintain the axial pinch load on the rim spacer. Having a higher thermal expansion than the titanium disks, the bolt load is reduced as the temperature increases. Preloading these bolts to 90% of the 0.2% yield at assembly will ensure the required load of 468,000 lb at the cruise condition.

To achieve a reliable preload, the bolts are designed to permit hydraulic stretching by means of a special fixture. The bolts are stretched to a predetermined preload and the nuts are tightened to a low torque level to ensure seating, then the bolts are released. Hydraulic stretching of the bolts is the most accurate means of establishing bolt preload. This allows the bolt to have a high tensile load without torsional stress.

(3) Disks

The design of the fan disks must meet the following criteria at maximum engine operating conditions.

1. Yield margin - 13% over maximum deteriorated speed
2. Yield margin - 3% over maximum overspeed due to malfunction
3. Burst margin - 20% over maximum deteriorated speed
4. Radial stress in the disk web is restricted to the average tangential stress. Radial stress between bolt holes will not exceed 90% of the average tangential stress. Allowable tangential stress is equal to yield factor times 0.2% yield strength of the material divided by the yield margin squared.
5. Low cycle fatigue for disks (LCF) - 20,000 engine acceleration cycles or 12,000 engine thermal cycles.

The limiting disk criteria were found to be the 13% yield margin requirement at maximum normal operating speed plus production control and deterioration tolerances. Figure 9 shows the resultant yield and burst margins as a function of rotor speed and inlet temperature. Vibratory stiffness requirements established cone thicknesses and rim widths. Refer to Volume III, Report B, Section II, paragraph H (Structural/Mechanical).

The yield margins imposed are calculated to prevent growth in excess of 0.1% during control malfunction overspeed. This growth restriction protects the engine from excess blade rubbing and ensures that adequate fits between disks, hubs, spacers, etc., will be maintained. These fits are required to ensure rotor balance and to keep engine vibration to a minimum.

Yield growth is limited to 0.1% during disk life. The following equation, incorporating a yield factor and the 0.2% yield strength, is used to predict the average tangential stress (and hence the rpm) at which the disk will grow 0.1%.

Disk average tangential stress at 0.1% growth =
yield factor x 0.2% yield strength

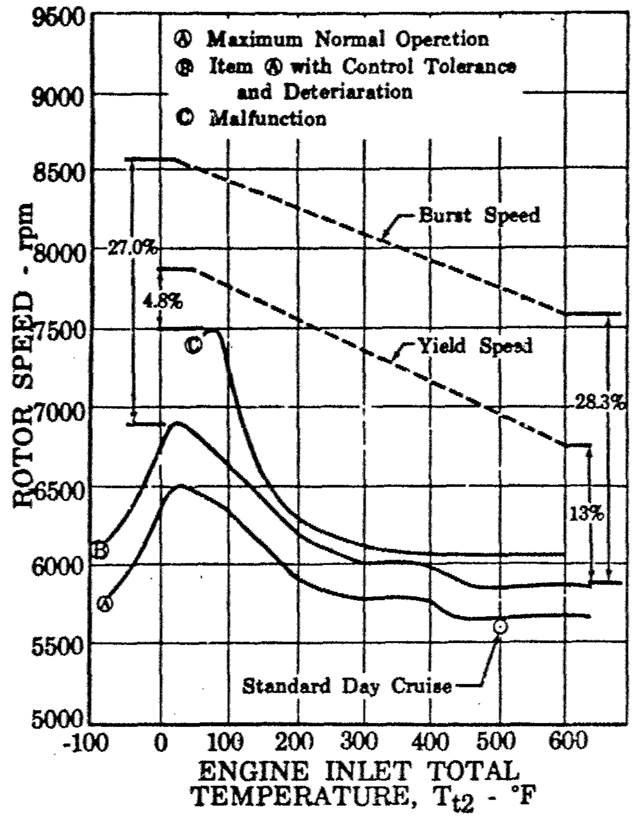

Figure 9. Fan Speed Margin FD 17509 IIA

The yield factor for compressor disks is 0.9 which was obtained from actual whirl pit testing in which disks were spun at progressively increasing speeds until 0.1% plastic growth was measured. Table 4 shows the results of some of the whirl tests which compare actual yield speed to predicted yield speed. It is evident from this table that, of the materials tested, the elevated temperature yield tests gave identical results with room temperature tests. However, all new materials are given the elevated temperature test to substantiate this general material tendency.

Table 4. Yield Factors for Disks

Engine/Stage		Material	Temperature	Actual Yield Speed / Predicted Yield Speed
JT3	16th	AMS 6415	70	1.03
JT3	14th	AMS 6304	70	1.07
JT4	16th	AMS 6304	70	1.01
JT8D	11th	AMS 4928	70	1.03
JT3	15th	PWA 1003	70	1.12
JT3	14th	AMS 6304	70	1.05
JT3	15th	PWA 1003	950	1.12
JT4	16th	AMS 5735	70	1.03
JT4	2nd	AMS 4928	70	1.03
JT4	9th	AMS 6415	610	1.03
JT4	15th	PWA 759	70	1.04
JT3	14th	AMS 6304	70	1.07
JT3	4th	AMS 6415	70	0.99
JT8D	2nd	AMS 4928	70	1.04
JT4	8th	AMS 6415	70	1.05
JT8D	12th	PWA 1002	690	0.99
JT3	16th	AMS 6304	850	1.12

The disk cross section shapes were determined primarily by disk and blade vibration requirements and by weight optimization. For the first stage, a compact thick cross section for rim torsional stiffness is required. The truss system has been carefully designed to balance out all forces and moments to assure there is no rotation of the disk rims. This is achieved by making the 1st-stage disk cross section slightly asymmetrical and by careful location of the axial spacer on the second stage.

A further advantage of the trussed cone design is ability to remove bolt holes from critical areas of the design. Experience has shown that low cycle fatigue (LCF) life is greatly reduced by the stress concentrations caused by bolt holes. The cone disk concept avoids this problem by moving the bolt holes from the central structure to the disk support cone, which operates at a low stress level. LCF is adversely affected at a disk bore or rim when high thermal gradients occur. The relatively compact configuration of the fan disks results in low thermal gradients. See figure 10 for 1st-stage fan disk temperature gradients and stresses, and figure 11 for 2nd-stage. The low expansion coefficient and modulus of titanium further decrease thermal stresses, thus improving LCF.

All surfaces of the disks are glass-bead peened to improve fatigue resistance. The 1st-stage hub spline, and the dovetail slots of both disks are treated per specification PWA 60 graphite varnish to prevent surface galling and to ease assembly. The bolt holes in the hubs are polished per PWA 99 to improve fatigue resistance. The fan drive spline on the 1st-stage hub is designed to a bearing stress of 11,000 psi at maximum sea level condition, and 3500 psi at cruise, with a predicted cruise temperature of 550°F. P&WA experience has shown that the non-working splines with double pilots will operate at 15,000 psi.

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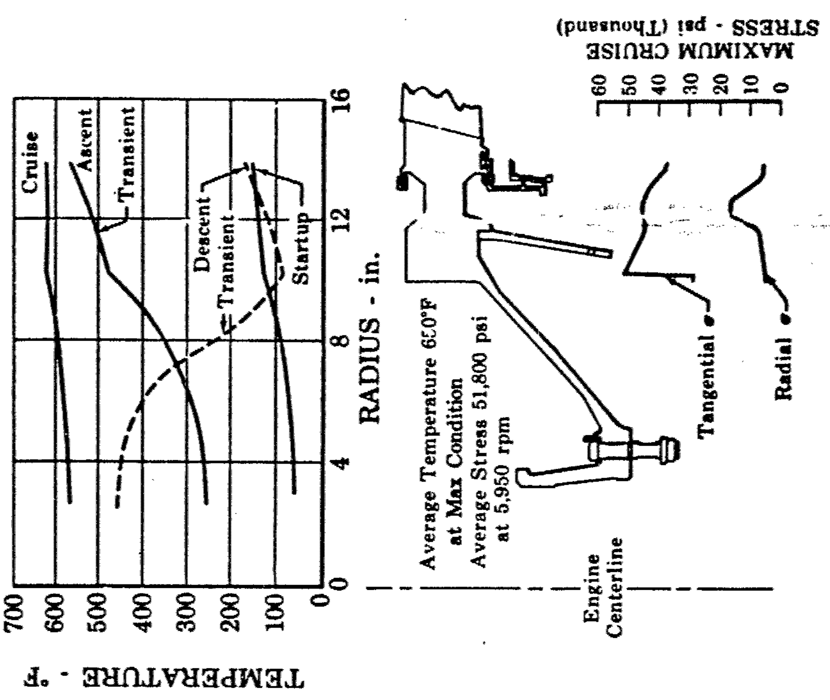

Figure 11. JTF17 2nd-Stage Fan Disk FD 16281 IIA

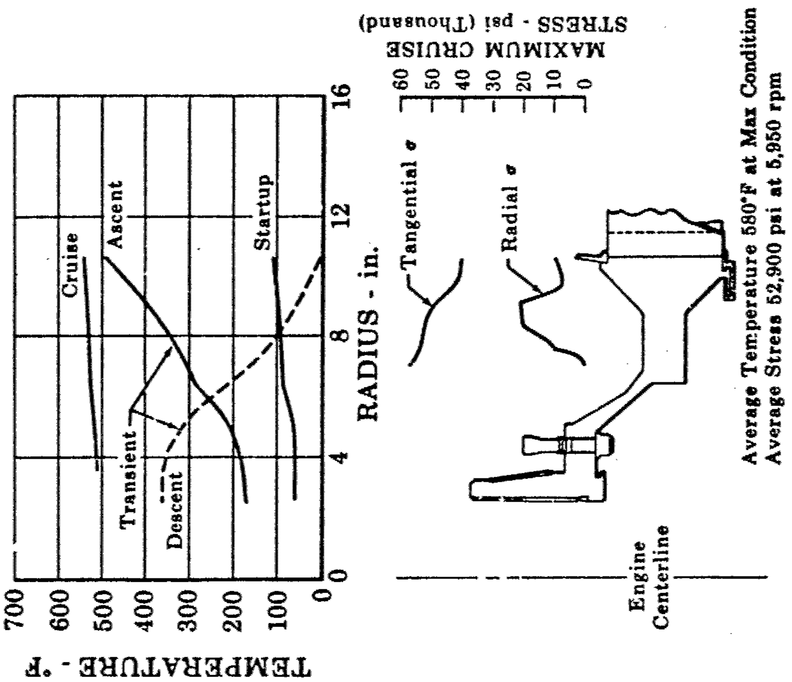

Figure 10. JTF17 1st-Stage Fan Disk FD 16282 IIA

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(4) Rotor Blades

(a) Airfoils

Both stages of blades are designed with two part-span shrouds for good structural integrity and resistance to foreign object damage. Experience obtained from the JT3D, JT8D, and TF30 fan development effort, and from a number of fan research stages, has shown that rotor blades must be designed to withstand self-excited vibrations at high rotor speeds, self-excited vibrations at stalled flow conditions (flutter), and standing wave resonant vibration excited by inlet distortion. The fan is designed for freedom from low order resonances and flutter over the entire normal operating range. Double shrouds are used for torsional stiffness and blade incidence control, which are important in eliminating blade flutter. The use of double shrouds has a further advantage in that it improves the FOD ingestion capability. The forces resulting from ingestion of large objects are distributed to adjacent blades by the shrouds, thereby limiting the resulting stresses. The total untwist moments caused by centrifugal loading on the blading are distributed between the two shrouds to produce torsional restraint and damping during vibration. Experience within the distortion conditions from both subsonic and supersonic inlets indicates that the highest stresses from vibration excitation occur in the lowest harmonic orders. Typical distortions from subsonic and supersonic inlets are shown in figure 12. The blade-disk systems are designed with 10% frequency margin over the frequency resulting from two impulses per revolution to allow for blade and disk tolerances. Resonances for low orders of excitation at the normal operating range are eliminated by this design. The resultant resonant characteristics, shown in figures 13 and 14, are achieved by combining blades designed for flutter resistance with nonrigid disks. The fan resonance characteristics are satisfactory for the imposed excitations. Past experience shows insignificant excitation from the higher order resonance points remaining in the operating range.

All fan stages have been analyzed with respect to the torsional stall flutter. This was accomplished by a nodal analysis of the results of the blade disk calculations. Torsional and bending contents of the vibratory modes are related to the aerodynamic excitation to predict the flutter resistance of each stage. The flutter criteria for making predictions (figures 15 and 16) are based on the accumulated experience from the J58, JT3D, JT8D, and other production engines. The 1st-stage fan blades of all current Pratt & Whitney Aircraft turbofan engines are part-span shrouded to allow the use of lightweight, high-aspect-ratio blading. As the blade tends to straighten under centrifugal load, normal forces are developed on the shroud contact surfaces. Friction between rubbing surfaces provides damping that resists the vibration motion of each blade at the shroud radius. This increases blade torsional frequency and maintains proper air incidence angle, providing the required flutter resistance. The fan stages on the production engine will have ample stall flutter margin as indicated in figure 16.

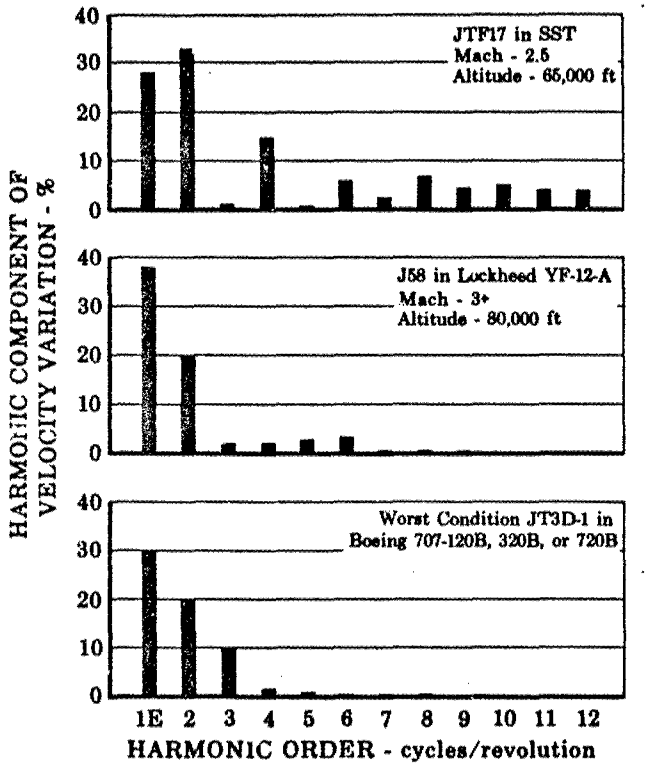

Figure 12. Harmonic Analyses of Typical Inlet Distortion FD 16230 IIA

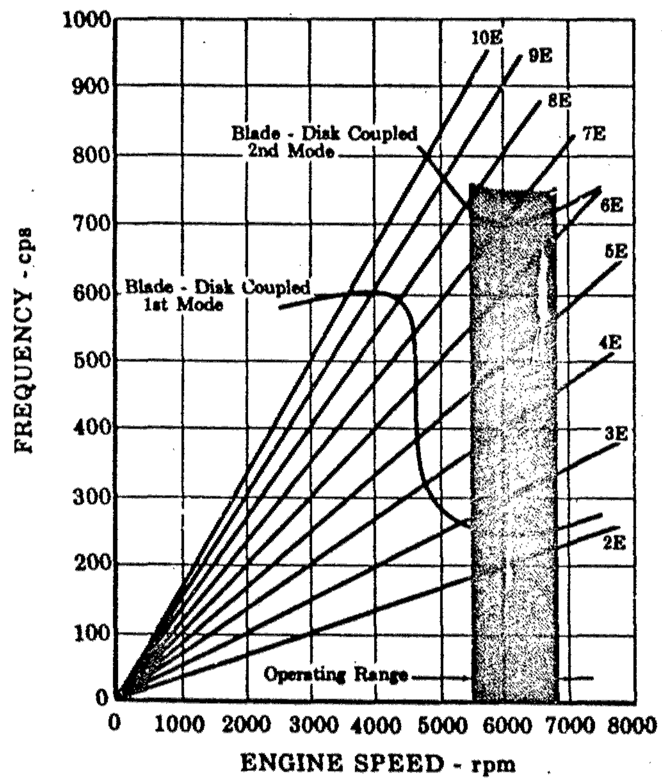

Figure 13. 1st-Stage Fan Rotor Resonance Diagram FD 16229 IIA

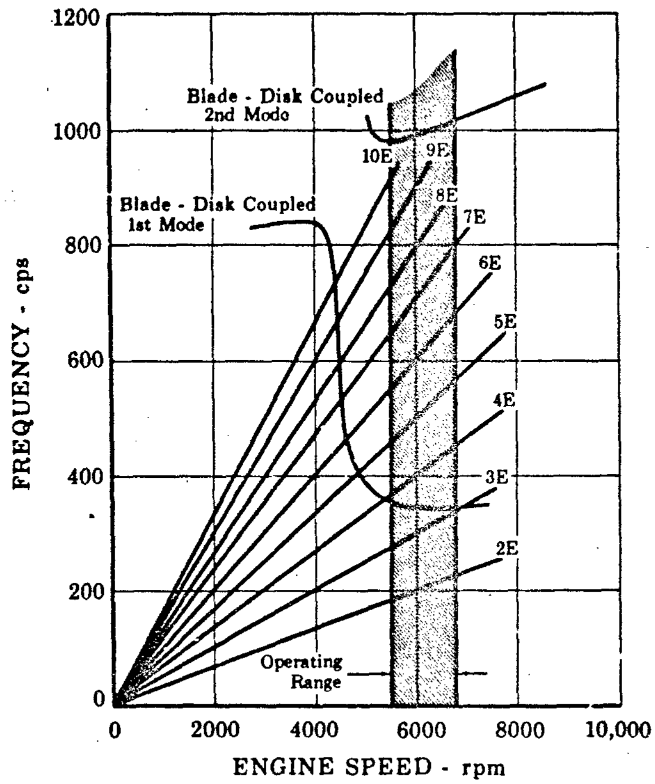

Figure 14. 2nd-Stage Fan Rotor Resonance Diagram FD 16228 IIA

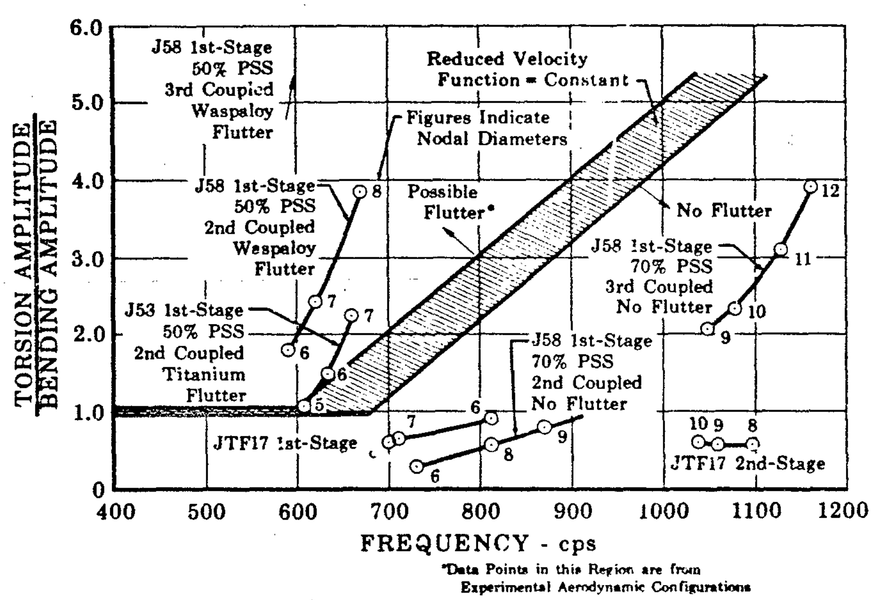

Figure 15. JTF17 Fan Flutter Study FD 16210 IIA

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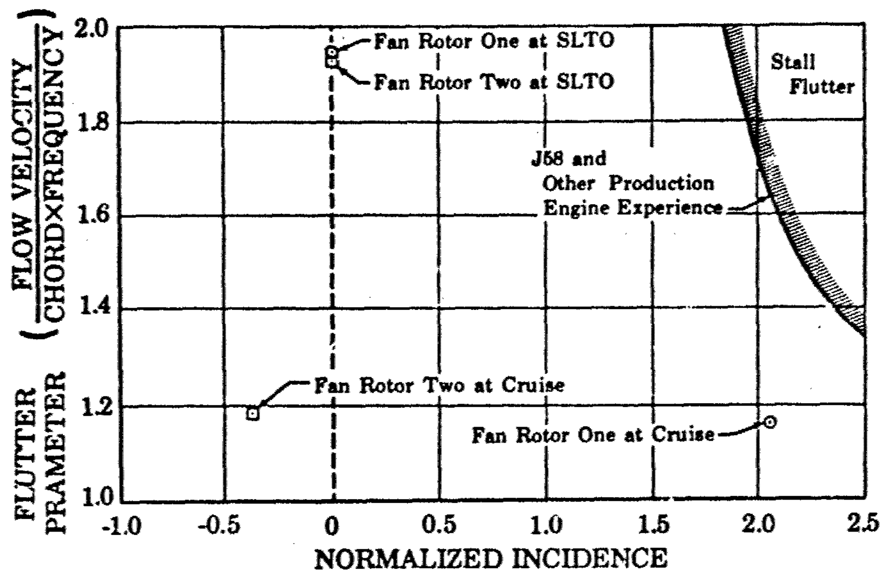

Figure 16. Fan Stall Flutter Characteristics FD 16227 IIA

Foreign object ingestion capability is a direct funtion of the overall blade strength and a correlation based on past experience (figure 17) in which blade bending energy is related to the blade root strength. Based on this criterion, the 1st-stage fan of the JTF17 engine will have ingesting capability superior to the other production Pratt & Whitney Aircraft engines. In addition to making the blade strength adequate for foreign object ingestion, it is also necessary to provide sufficient blade-to-vane spacing to ensure clearance when the rotor blades deflect. The part-span shroud contact angles are set to prevent the blades from disengaging. By doing this, the load will be distributed among the several adjacent blades, thereby limiting the total deflection of the system.

The design criteria that must be satisfied in the design of the part-span shroud are (1) minimum aerodynamic drag, which is attained by keeping the critical Mach number of the section high, (2) shingling resistance, which is accomplished by maintaining a shroud thickness as dictated by an empirical correlation of past P&WA experience, and (3) sufficient wear resistance, which is accomplished by flame-plating the wear surfaces of the shroud with tungsten carbide and maintaining low bearing stresses. The shingling parameters, in comparison with the experience of the other P&WA production engines, are shown in table 5 and the bearing stress limitation is shown in table 6. (See figure 18.)

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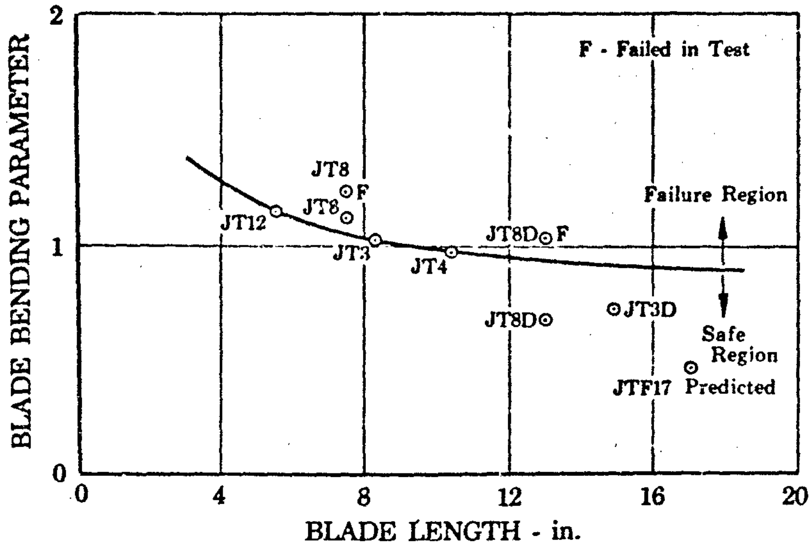

Figure 17. Foreign Object Ingestion Study FD 16214 IIA

Table 5. Part-Span Shroud Shingling Parameter Experience

Engine	Stage	Shingling Parameter (Radians)
JTF17	1 Inboard Shroud	0.0566
	1 Outboard Shroud	0.0518
	2 Inboard Shroud	0.0585
	2 Outboard Shroud	0.0530
JT3D	1	0.0548
JT8D	1	0.0578
TF30	1	0.0423
J58 (Experimental)	4	0.0278 (Shingled at low rpm)
JTF14C (Experimental)	1	0.0168 (Shingled at low rpm)
JTF14F (Experimental)	1 Inboard	0.0242
	1 Outboard	0.0245

Table 6. Part-Span Shroud Bearing Stress Summary

Engine	Stage	Shroud Location	Allowable Bearing Stress (psi)	Actual Bearing Stress (psi)
Initial Experimental Engine	1	Inboard	5000	4000
		Outboard		
	2	Inboard	5000	3900
		Outboard	5000	3470
JTF17	1	Inboard	5000	4840
		Outboard	5000	1450
	2	Inboard	5000	5000
		Outboard	5000	4190

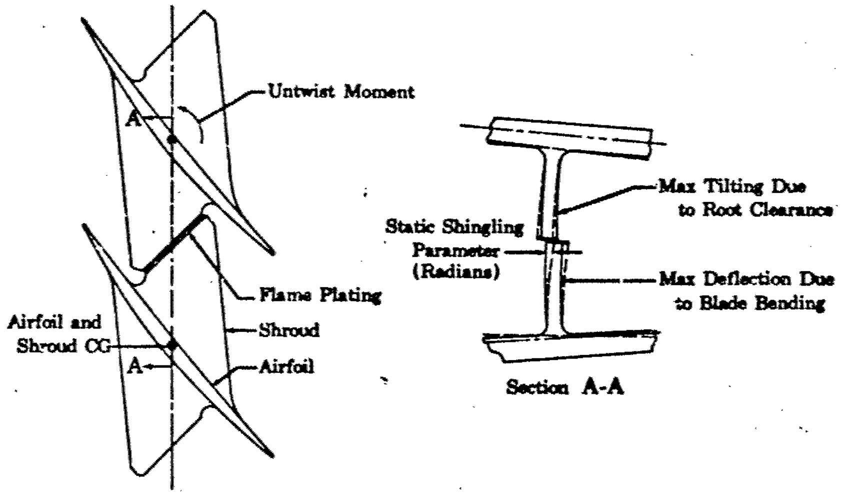

Figure 18. Airfoil Shroud Arrangement FD 16209 IIA

(b) Blade Attachments

The blades are mounted in a single dovetail slot in the disks and are retained axially by the locking rings. This method of retention is more reliable than the conventional tablocks. The 1st-stage blade is retained axially in the disk by the rim spacer in the rear and by a lock ring in the front. The 2nd-stage blade is retained by a lock ring in the rear, which engages in a tang on each blade to prevent fore and aft movement. Both 1st- and 2nd-stage lock rings (figure 19) work on a breech-lock principle. Both the disk and the lock ring have a series of tangs that intermesh and engage when the ring is rotated. The lock rings are each locked to the disk in the engaged position by six rivets. The advantages of this locking system are: (1) all blades are locked simultaneously, (2) no loads are placed on the retaining rivets, and (3) there is redundancy in the locking because all the rivets must fail before the ring can disengage. Tubular rivets are used for easy installation and removal, thus preventing damage to the disk.

The blade attachment dovetails, as shown in figure 20, are designed to transmit the radial forces, axial and tangential bending moments, and torsion into the disk rim. Table 7 shows the design criteria used for the JTF17 engine. All root designs are further verified by extensive laboratory photoelastic tests and analysis. See figures 21 and 22 for stress comparison with other P&WA engines.

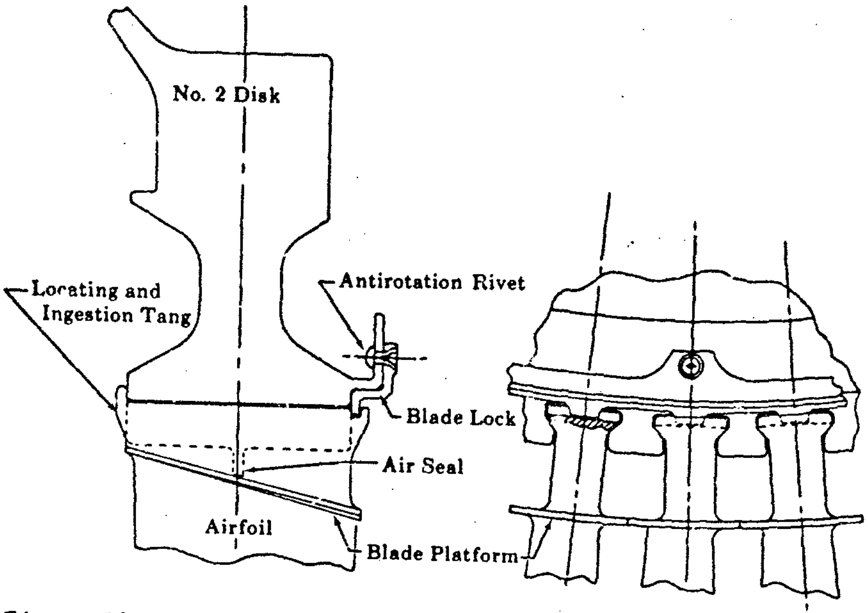

Figure 19. 2nd-Stage Blade Lock

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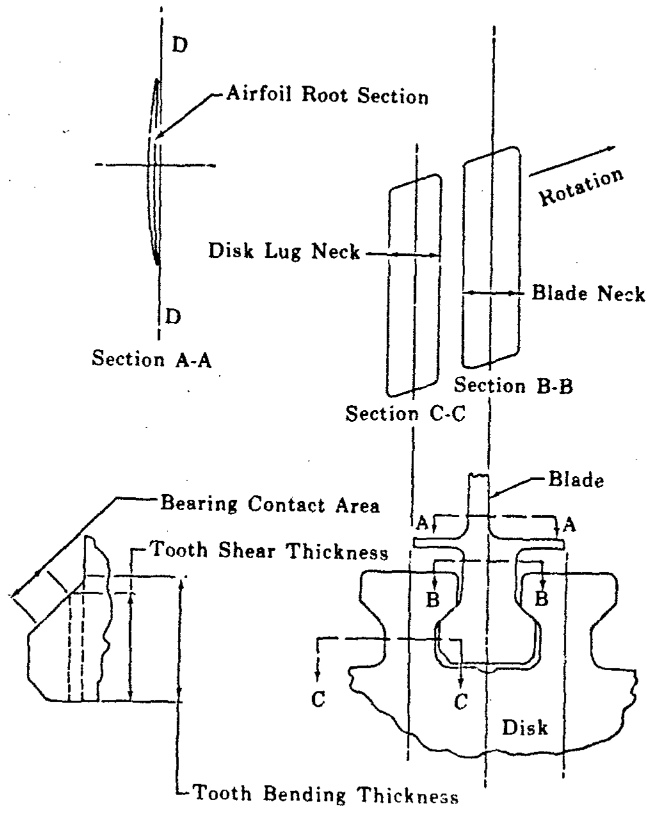

Figure 20. Typical Blade Attachment

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Table 7. Blade to Disk Attachment Stresses (See figure 20.)

	First Stage JTF17	Second Stage JTF17	Maximum Allowable Stress Ratios**
Speed - RPM	5800	5800	
Blade Information			
Temperature, °F	596	676	
Material	AMS 4928	AMS 4928	
0.2% Yield Strength	71,000	69,000	
Blade Stresses			
Neck Tensile, psi	21,400	14,680	
Stress Ratio	0.301	0.212	0.30
Bearing, psi	64,400	55,080	
Stress Ratio	0.907	0.798	0.90
Bending, psi	17,120	12,290	
Stress Ratio	0.241	0.178	0.50
Shear, psi *	14,590	10,250	
Stress Ratio	0.205	0.148	0.25
Combined, psi	38,600	26,920	
Combined Allowable, psi	46,000	41,200	
Disk Information			
Material	AMS 4928	PWA 1202	
Temperature, °F	580	660	
0.2% Yield Strength	72,000	73,200	
Disk Stresses			
Lug Tensile	24,850	15,200	
Stress Ratio	0.360	0.208	0.45
Bending	14,750	21,550	
Stress Ratio	0.205	0.299	0.50
Torsion	5,160	6,350	
Stress Ratio	0.072	0.087	0.20
Combined	39,600	36,900	
Stress Ratio	0.550	0.505	0.55
Geometric Limits			Minimum Limits
Lug Z Ratio	3.640	5.809	1.45
Blade Root Z Ratio	4.014	4.035	4.00
Neck Ratio	0.854	1.056	0.8
B/2 Limit	1.022	0.974	0.5
R Factor	0.818	0.734	0.3

* Stress Ratio = $\frac{\text{Actual Stress}}{\text{0.2\% Yield Strength}}$

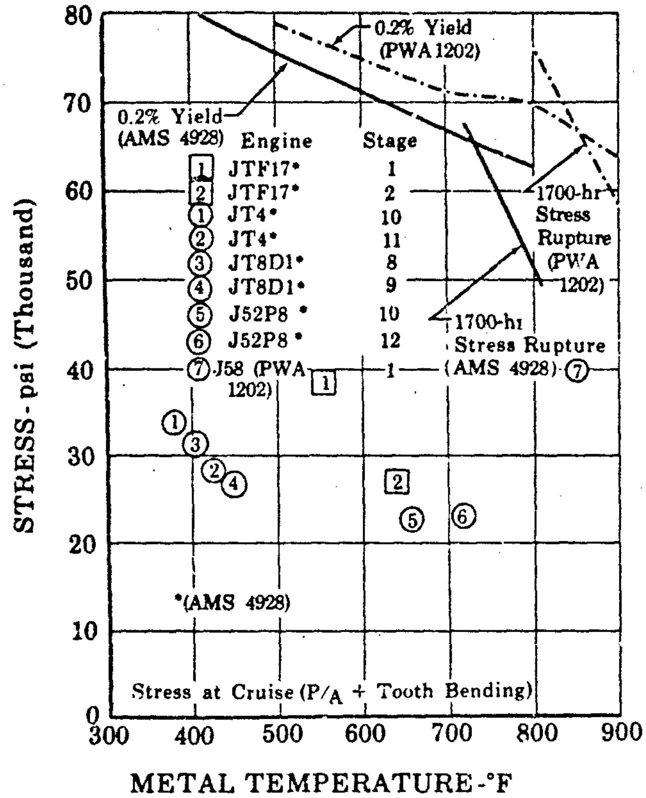


Figure 21. Blade Dovetail Combined Stress FD 16304 IIA

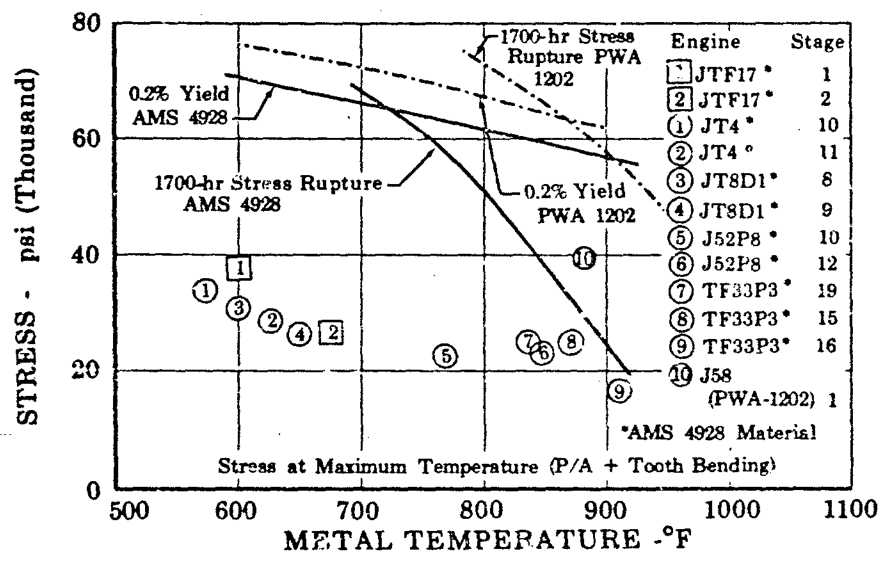


Figure 22. Blade Dovetail Combined Stress FD 16303 IIA

Allowable blade attachment stresses and geometry limits are based on empirical formulas developed by experience on such engines as the JT3D, JT8D, JT4 and J58. The geometric limits established are as follows:

1. The root attachment length must be equal to at least 1/2 of the axial chord of the blade to provide optimum airfoil root support. This relationship is termed the "B/2 Limit".
2. The "Root Z Ratio" requires the bending strength of the blade attachment (Z Root) to be four times greater than that of the airfoil root (Z foil) in order to prevent root fatigue failure.

$$\text{Root Z Ratio} = \frac{(\text{Z Root})}{(\text{Z Foil})} \geq 4.00$$

3. The "Lug Z Ratio" requires the bending strength of the disk lug (Z Lug) to be 1.45 times greater than that of the airfoil root (Z Foil) in order to prevent lug fatigue failure.

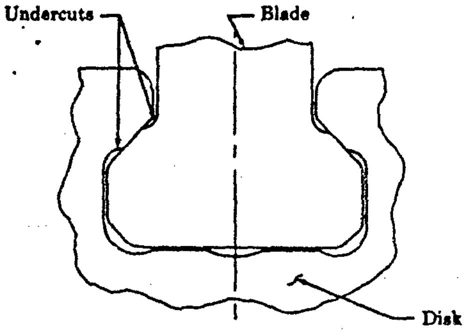
$$\text{Lug Z Ratio} = \frac{(\text{Z Lug})}{(\text{Z Foil})} \geq 1.45$$

4. The "Neck Ratio" controls the relative size of the blade neck with that of the disk lug neck and requires that the disk neck thickness with better forging properties due to its configuration be at least 80% of the blade neck thickness.

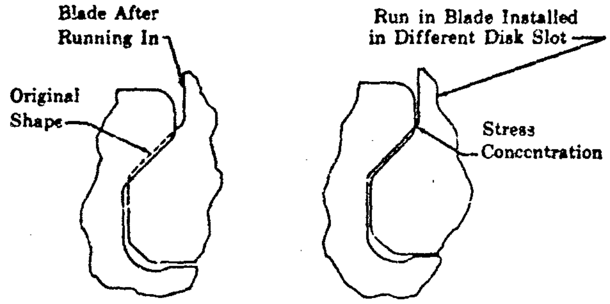
$$\text{Neck Ratio} = \frac{\text{Disk Lug Neck}}{\text{Blade Neck}} \geq 0.80$$

5. The "R" Factor: controls the load distribution due to the airfoil root angle on the attachment so that proper loading of the acute angle surfaces of the blade dovetail is assured and excessive stresses on the obtuse angle surfaces are prevented. This uneven stress distribution is the combined result of gas bending moment plus the torsional moment caused by the dovetail centerplane being other than a right angle.

Root attachments for titanium parts require the use of radius undercuts at each end of the bearing face. (See figure 23.) Based on J58 Mach 3+ experience, the undercuts greatly increase the life of root attachments by eliminating the highpoint contact loads that can occur when the corner radii of the disk and blade interfere. If wear, local yielding, or redistribution occurs in the root attachment, the undercut also prevents the buildup of a high point on the neck radius. If the blade is assembled in a new disk or in a different slot in the same disk, the high point can interfere and cause a stress concentration.



Root with Undercuts Typical 1st and 2nd Stage Blade



Root Attachment Without Undercuts

Figure 23. Root Details FD 16213
IIA

(5) Vanes and Cases

Since the useful maximum temperature range of titanium is from 850° to 950°F and is above the highest operational fan temperature, it can be used liberally throughout the fan section for lightweight design.

The parts life requirements shown below are established on the basis of the general engine parts life requirements described in Section I. (Introduction to Report B).

	Required	Actual
Front Mount Case	50,000 hours	>50,000 hours
Fan Stators	10,000 hours	>50,000 hours

The 1st-stage fan stator is full length and the 2nd-stage stator incorporates an annulus splitter, separating primary engine flow from the duct flow. (See figure 1.) The vanes are mechanically attached to the supporting cases to obtain beneficial damping and facilitate replacement. The stators behind each blade row are axially spaced to minimize this source of blade-to-vane passing frequency noise. Inlet guide vanes forward of the fan rotor were not used, eliminating one source of blade-to-vane passing frequency noise.

Airfoils are forged and riveted at both the inner diameter and outer diameter of the 1st-stage stator to permit easy replacement of individual vanes. The 2nd-stage stators are also forged and mechanically attached

to obtain beneficial damping. An inner platform seal ring at the stator ID and a knife edge seal on the first stator retard recirculation leakage around the stator. The seal lands are serrated construction to provide better sealing than conventional solid lands because of the tighter running clearance permissible.

Commercial engine experience has indicated that 1st-stage blades and vanes are susceptible to foreign object damage in service. For this reason, it is required that the JTF17 design permit horizontal removal -- without disturbing the bearings or disassembling the front mount case -- of individual 1st-stage blades, the 1st-stage rotor assembly, or the fan module, which consists of both rotor stages and the 1st-stage stator assembly.

The detachable fan splitter nose and forged foot-bolted construction of the 2nd-stage stator, and the fan exit guide vane permit ready access to this area for individual vane replacement. In addition, the vane in the duct is slotted to improve stator efficiency by proved means of boundary layer control. Pratt & Whitney Aircraft's experience in the STF200 and research rigs have proved the aerodynamic feasibility of slots. (See figure 24.)

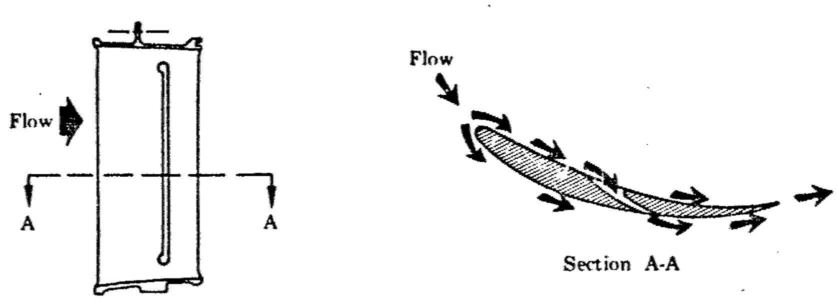


Figure 24. Slotted Duct Exit Guide Vane FD 16231 IIA

The fan is designed so that the required maneuver loads shall not cause any permanent deformation. (See figures 6 and 7 for Boeing, and figure 8 for Lockheed.) (The primary limiting design modes for each major part are shown later in tables 9 and 10 in the Materials Summary; paragraph (9), following.) Because of the low temperatures of the fan, all structural limits are based on yield strength (0.2% YS), deflection, or buckling (1.3 margin).

The ability to maintain minimum blade tip clearances is self-explanatory criteria. High local radial deflection in the front engine mount case is isolated from the tip clearance region by the double-wall construction.

The wall thicknesses of the fan cases in the JTF17 have been designed to ensure blade containment. Containment is provided by all material external to the blade. Thus, the fan containment case consists of the inner shroud and the outer case combined.

Prior to the use of fans, the criteria employed to ensure containment equated the potential energy absorption capacity of the case to the kinetic energy of the blades released. While this method yielded generally acceptable results for high compressors and turbines (figures 25 and 26), tests revealed that the energy absorption was not sufficient for the larger diameter fans. The blade impact was determined to produce intense local compressive stresses that caused a portion of the case to be sheared before the energy could be absorbed.

A technique has been developed to provide a type of containment analysis that employs the shape of the failed blade and the dynamic shear strength of a localized section of the engine case. This method has provided consistent results in tests using actual blades and engine cases. From many tests, a factor has evolved that relates the actual blade velocity at impact to the total velocity of the same weight blade required to shear the case. Experience defines this factor. (See figure 27.)

In the JTF17, shear criteria has been used to size the fan case. The shear criteria was also used for the compressor and turbine sections as well as the energy criteria. Table 8 shows the actual containment factors, shear and energy, provided in the engine.

Containment Capability of cases is judged by a "containment factor," that is empirically obtained from test.

This containment factor (CF) relates case energy absorption potential (PE) to blade kinetic energy (KE)

$$CF = \frac{\text{KE (All Blades Instage)}}{\text{PE (Of all cases surrounding stage)}}$$

For unshrouded blades CF = 0.25
Shrouded blades CF = 0.12
(such as turbine stage)

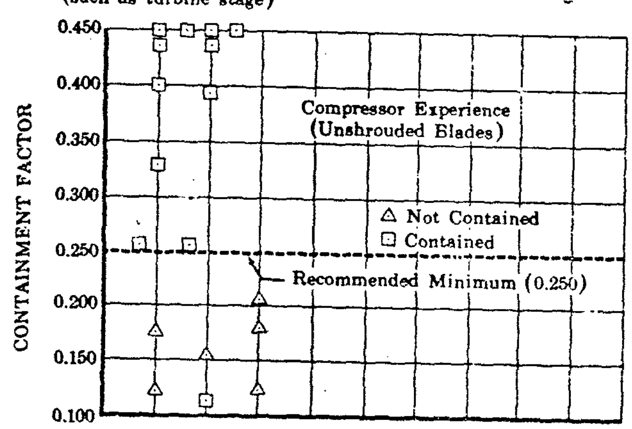


Figure 25. Containment Experience

FD 16232
IIA

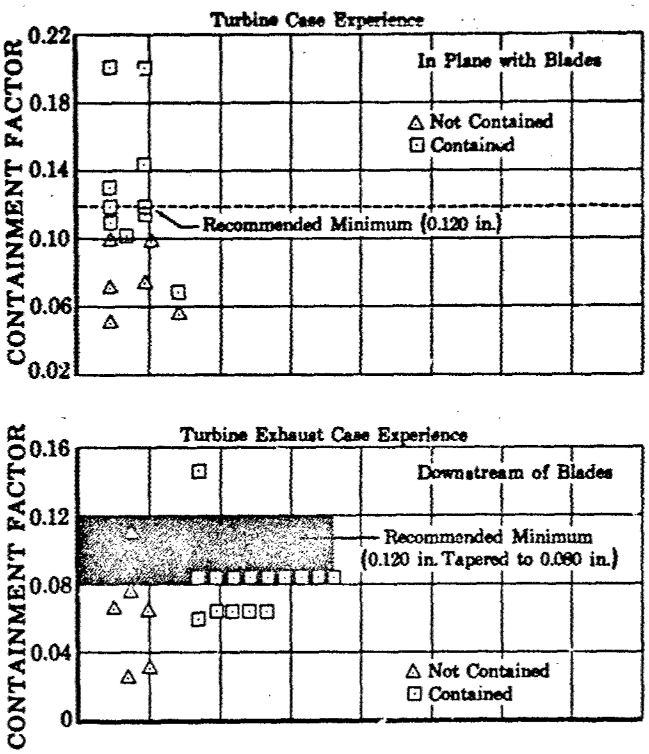


Figure 26. Containment Experience

FD 16234
IIA

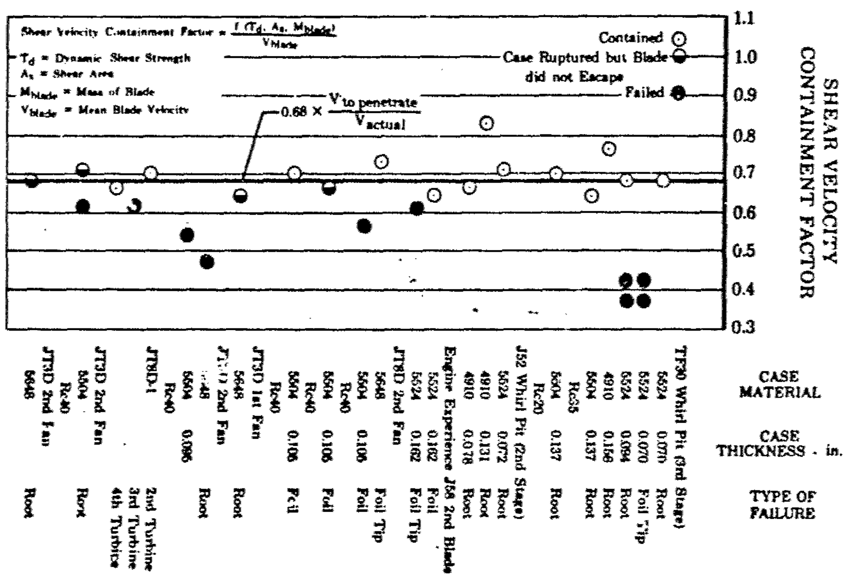


Figure 27. Containment Experience Based on Shear Penetration Testing

FD 16233
IIA

Table 8. JTF17 Blade Containment

Stage	Shear Criteria (See figure 27.)		Energy Criteria (Sec figure 25.)	
	Factor	Allowable Min	Factor	Allowable Min
First (Fan)	0.68	0.68	11.3	8.6
Second (Fan)	0.68	0.68	12.6	8.6
Third (Compressor)	0.772	0.68	0.25	0.25
Fourth (Compressor)	0.772	0.68	0.263	0.25
Fifth (Compressor)	0.772	0.68	0.322	0.25
Sixth (Compressor)	0.772	0.68	0.264	0.25
Seventh (Compressor)	0.772	0.68	0.379	0.25
Eighth (Compressor)	0.772	0.68	0.475	0.25
First (Turbine)	0.890	0.68	0.250	0.250
Second (Turbine)	0.755	0.68	0.120	0.120
Third (Turbine)	0.702	0.68	0.120	0.120

A containment failure involves an impulsive load of infinitely short duration that results in an extremely high strain rate (approximately 100,000,000 times greater than standard tensile test strain rates). High strain research indicates that the true energy absorption and shear capability of a material varies with strain rate, generally increasing substantially with increased strain rate.

Material dynamic shear factors have been obtained by Pratt & Whitney Aircraft through ballistic testing of materials at velocities related to blade impact and subsequently verified by whirl pit testing with actual blades and cases. Figure 28 shows the relationship of dynamic shear factor to static tensile for a variety of materials.

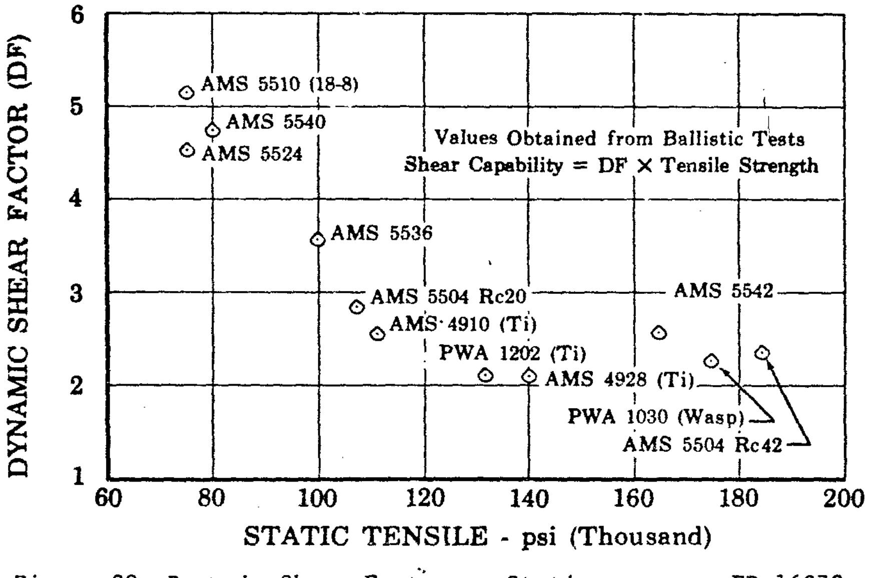


Figure 28. Dynamic Shear Factor vs Static Tensile Strength

FD 16279
IIA

(6) Intermediate Case

The intermediate case, which is the section between the fan and the compressor, is the support for the No. 1 and No. 2 bearings and also the major support for the gas generator.

The intermediate case is designed to carry the radial and thrust loads of the No. 1 and No. 2 bearings, thrust load, shear torsional load of the gas generator, and the main engine maneuver loads. In addition to these loads, the intermediate case must withstand the unbalance forces caused by blade loss and supplement the 2nd-stage fan OD liner in providing blade containment.

The basic structure of the intermediate case is a titanium weldment utilizing AMS 4966 (A-110) forgings and AMS 4910 (A-110) sheet. Butt welded joints will be used on all attachments by welding to integral, machined, contoured standups in the wall forgings. (See figure 29.) This construction has the decided advantage of having all welds loaded in simple tension or compression. Areas with combined bending stresses will occur in parent material away from the welds and heat-affected zone.

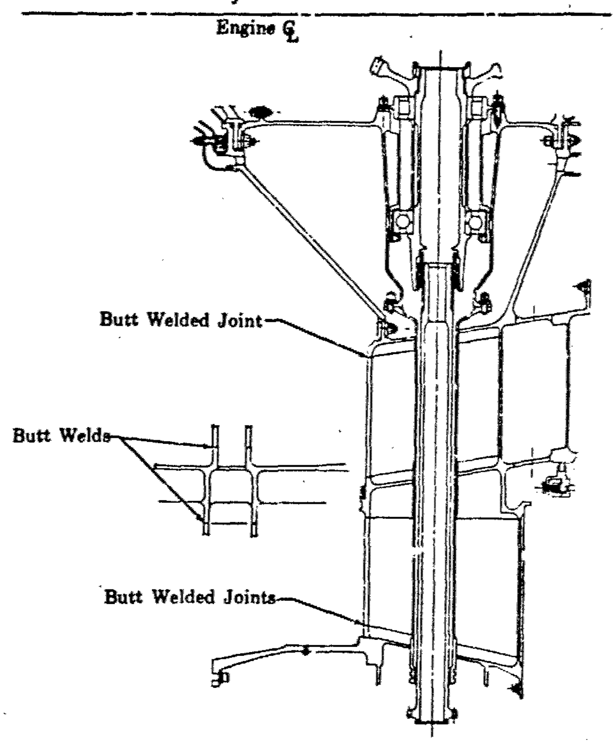


Figure 29. Intermediate Case Basic Structure FD 17666 IIA

The struts are fabricated from a combination of sheet stock and forgings and are butt welded to airfoil contoured platforms that are an integral part of the wall rings.

Vane to shroud cracking experienced in service on the JT3D inlet case has led to the standup foot butt weld design. Cases such as the JT3D

diffuser case, having overlap joints, which are either fillet - or resistance-welded, have proved generally much less reliable in field service than butt welds.

The repairability of this intermediate case, therefore, must be judged on the following:

1. The butt-welded joint, removed from the corner stress concentrations, results in a more reliable case having fewer cracks.
2. When and if cracks do occur, they are much easier to rout out and clean preparatory to welding, since the joint is accessible, thus ensuring successful repairs. A crack in an overlap joint is inherently impossible to clean.

Of the eight struts in the primary engine stream, four are increased in airfoil thickness to house the towershafts and the lubricating oil drain line. The struts are oriented and spaced as shown in figure 30.

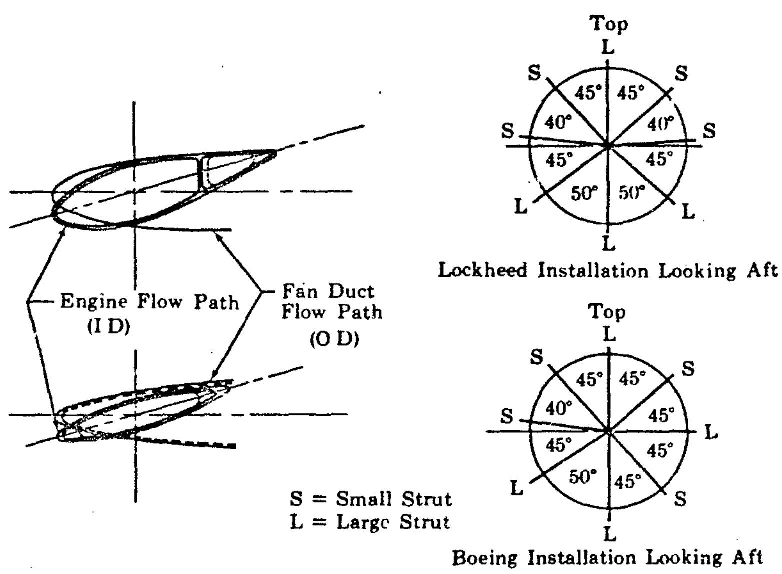


Figure 30. Intermediate Case Strut Orientation FD 16283 IIA

The large struts were sized to accept a towershaft enclosed in a sleeve to maintain a nominal air space around the sleeve to insulate against oil coking. The sleeves connect the bearing compartment with the gearboxes and carry oil or breather air. The bottom strut has the sleeve only, and is used to drain oil from the bearing compartment to the sump on the outside of the engine. This strut is also used as an overboard drain in the event of oil collection in the vent system.

The small struts are used for oil supply, seal air supply and seal air overboard vent. The oil supply line furnishes pressure oil to cool and lubricate the No. 1 and No. 2 bearings, the gears, and towershaft bearings, and to cool the rotating shaft seal plates.

Seal pressurizing air from fan discharge is supplied to the hydrostatic seals. The strut is used to carry the air from the fan discharge duct of the engine to the inner cavity. At the inner end, the air is picked up and ducted to the seals.

The remaining two small struts are used to vent the seals to ambient overboard pressures. The cavity around the bearing compartment, and all of the struts, comprise a plenum that is at ambient pressures. The two vent struts have flanged connections at the outside to connect to the airframe overboard vent lines. (See figure 31.)

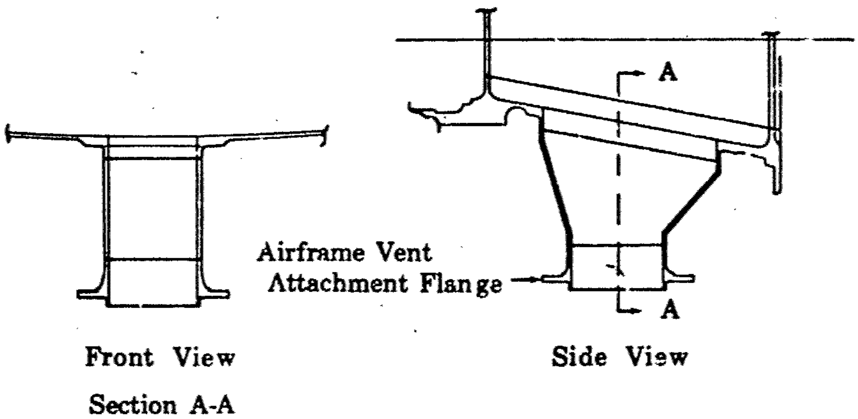


Figure 31. Intermediate Case Overboard Access for Seal Vent FD 16284 IIA

The bearing compartment has been designed as a separate and removable part with the rotor bearing supports independent of either the compartment or the intermediate case. This simplifies the intermediate case structure and permits the removal of close toleranced bearing housing and/or supports if weld repair is required on the case. It also makes those parts most subject to damage replaceable.

Replaceable bushings or liners will be used at all points where wear might occur and cause damage to the main case.

The intermediate case was structurally analyzed for the combination of engine loads, maneuver loads, and aerodynamic loads. The outer shroud forward of the struts is limited in thickness by buckling. The ring behind the inner struts, which carries the front of the gas generator, is also limited by buckling.

The loads from the bearings and gas generator are carried to the outer ring and shroud through the struts. The strut loads are distributed into the case through integral rings that are stress limited.

The inner conical supports for the bearing compartment are deflection limited.

The low rotor speed will be indicated by a signal from inductive pickups installed in the intermediate case approximately on the center-

line of the 2nd-stage fan blades. (See figure 32.) The pickup will be excited by the passing frequency of titanium fan blade tips. This signal, by means of an amplifier, will be intensified by a ratio of 1000:1.

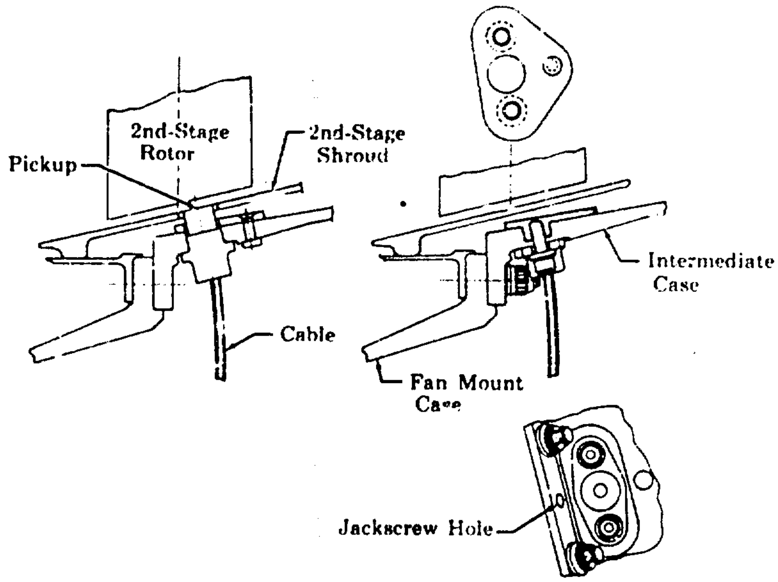


Figure 32. N_1 Speed Pickup

FD 17501
IIA

The inductive pickup on the fan blades was selected to be used as it has proved to be rugged, lightweight, and easy to install.

The pickup will be replaceable from the outside of the engine while installed in the airframe.

(7) No. 1 and No. 2 Bearings Compartment

The compartment which contains the No. 1 and No. 2 main bearings is located aft of the fan section and housed inside the intermediate case. This compartment houses the main engine thrust bearings and seals and the gear trains for the three accessory gearbox drives. Figures 33 (Boeing) and 34 (Lockheed) show an end view of the drive arrangements.

The No. 1 bearing takes forward thrust for the low (N_1) rotor shaft. The No. 2 bearing is the front support for the compressor (N_2) rotor and takes rearward thrust. These bearings incorporate flanged races offering the advantage of providing a postive anti-rotation attachment and weight saving. Previous bearing retention practice, using a nut and lockwasher, had to rely on high assembly torques and in some cases left hand threads, depending on the direction of rotation, to preclude race skidding. These high torques often require special hydraulic tool wrenches and are more sensitive to obtaining the required preload than multi-bolted flanged bearings. The flanges also cure a possible misassembly problem existing with non-flanged bearings.

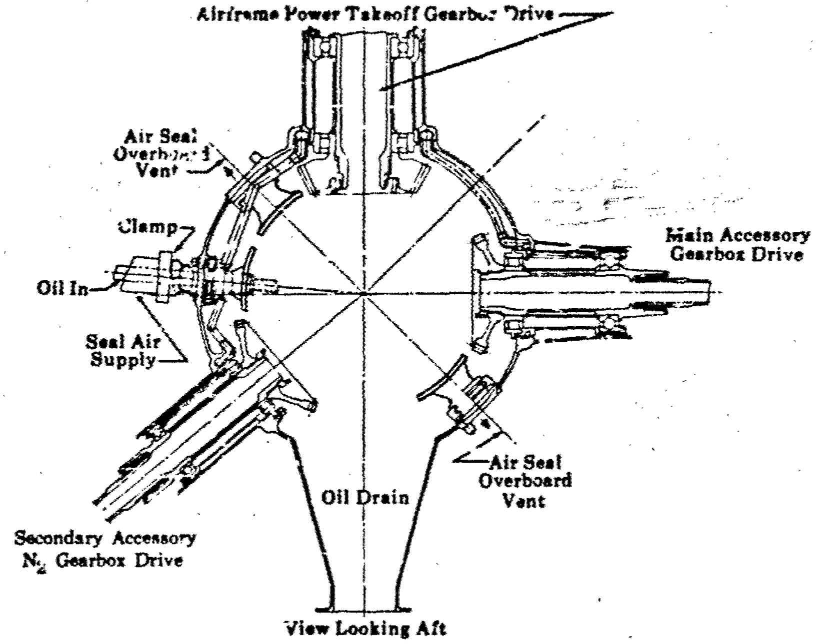


Figure 33. Bearing Compartment - Boeing FD 16439 IIA

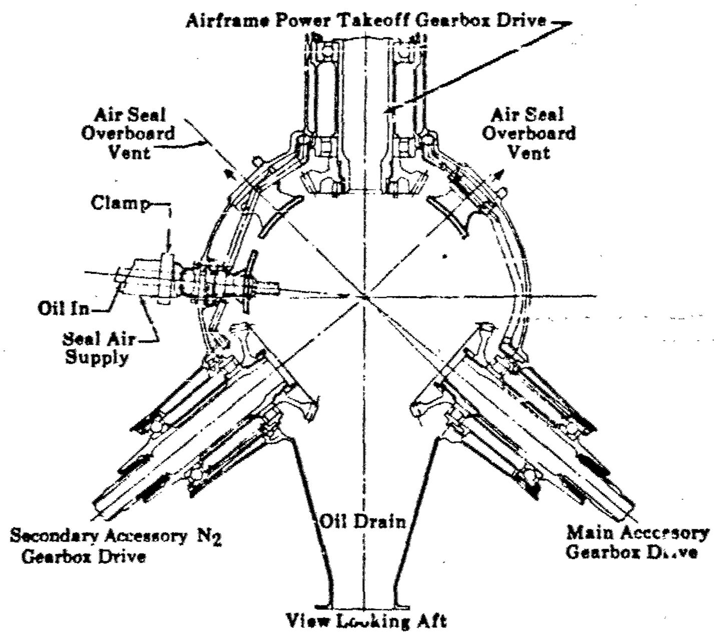


Figure 34. Bearing Compartment - Lockheed FD 16438 IIA

The No. 1 and 2 thrust bearings are each supplied with oil through the inner race. This oil supplies the lubrication and cooling of the inner race for maximum life of the bearings. Oil holes in rotating parts are drilled at a minimum angle of 1° 30' with the rotational axis to provide centrifugal pumping to keep the passage clear or washed of dirt or sludge accumulations. This assures maximum heat transfer from the oil to the clean passage and prevents eventual clogging due to stagnant sludge.

The AMS 5667 (Inconel) bearing supports are removable from the supporting intermediate case. This meets both the requirements of "bearing removal from the front" and "independence from distortion on critical mounting surfaces due to large weldment repairs".

The No. 1 bearing is located close to the fan rotor center of gravity to provide the required low rotor critical speed margin. Considerations of fan assembly removal and bearing failure modes and effects analysis resulted in the JTF17 configuration, which places the No. 1 bearing on a sleeve, isolated from the direct torque path between the turbine and compressor. This design approach meets the reliability criteria of compressor-turbine torque drive integrity.

In protecting against turbine-compressor separation resulting from bearing failure, the criteria applied to thrust bearing placement is to create a heat dam such that the torque driving shaft is protected until a "slow down" by energy absorption is accomplished.

This bearing compartment contains four hydrostatic main shaft seals. The concept of the hydrostatic static seal, while covered in more detail in Volume III, Report B, Section II, paragraph G (Seals and Bearings), is briefly as follows. Basically a modified conventional carbon seal and piston ring configuration, the hydrostatic differs in that a thin film of air separates the stationary carbon seal interface from the rotating seal plate. In order to maintain the air film, air must be supplied to the interface at a pressure higher than either the oil compartment or the environment external to the compartment. This air then flows in both directions, into the compartment and outboard external to the compartment. (See figure 35.)

As added protection, the air which flows outboard of the interface is collected in a manifold formed between the hydrostatic carbon and a secondary labyrinth seal. The manifolding serves two functions - it controls the seal thrust balance to a minimum pressure level, thus requiring supply air from a cooler source, and it serves as a backup chamber through which to exhaust any possible oil vapors that might get past the main seal.

There are two main seals for each rotor shaft (N_1 & N_2). The seal pressure air is supplied from fan discharge air extracted off the duct passage inner wall. (See figure 36.)

The seal leakage air collected in the manifolds is directed out through the intermediate case to be ducted to overboard ambient. (See figure 37.)

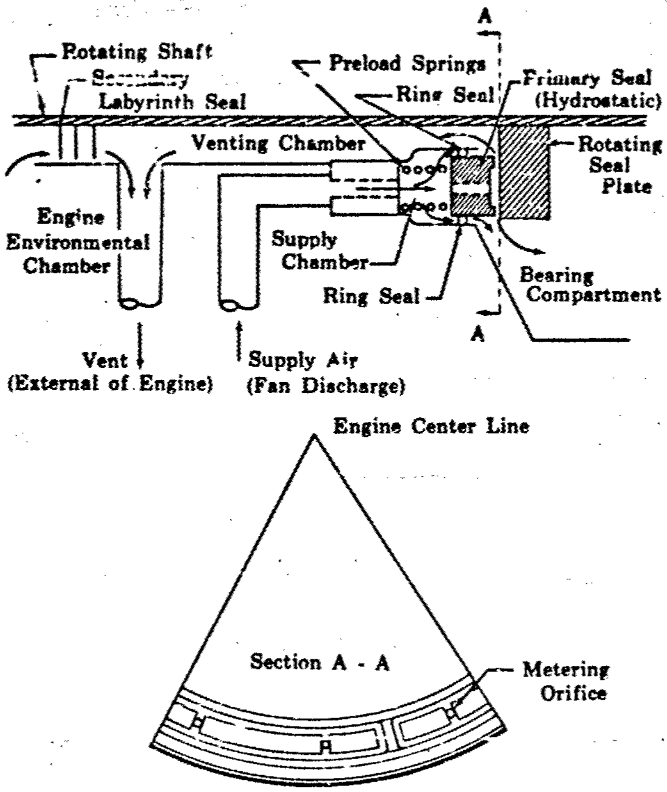


Figure 35. Hydrostatic Seal System FD 16568 IIA

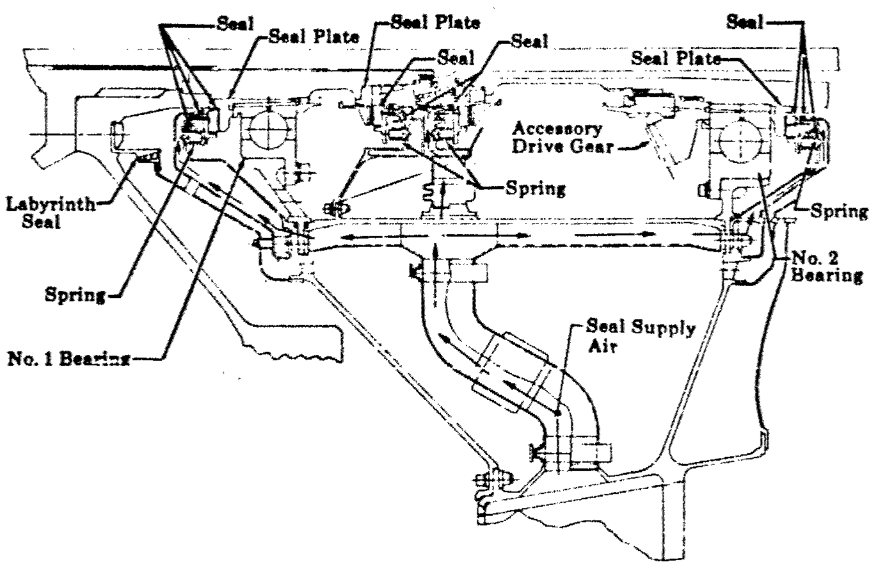


Figure 36. No. 1 and 2 Bearing Compartment Seal Supply Air System FD 16428 IIA

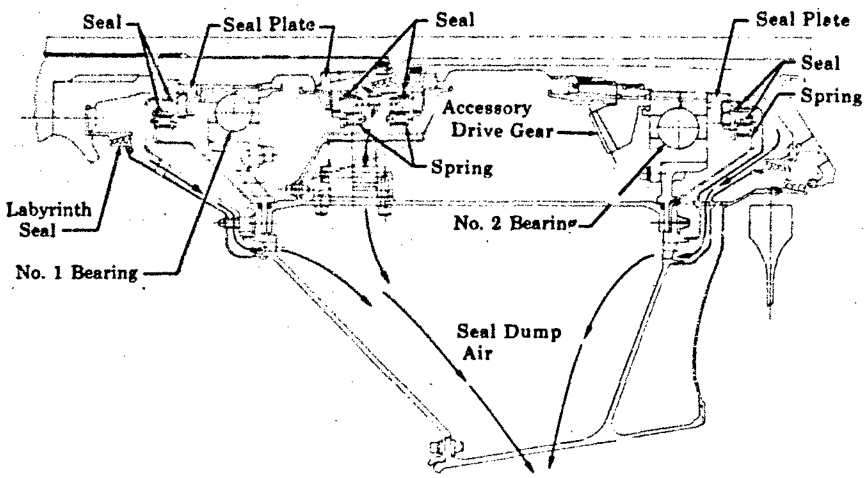


Figure 37. No. 1 and 2 Bearing Compartment Dump Air System FD 16429 IIA

As the source of seal pressure air (fan discharge) is internal, additional pressure is required to keep the seals closed during start and shutdown. This pressure is obtained from several coil springs reacting with a total of 10 pounds for each seal. This spring force also acts as a backup in the event the pressurizing air is not available to the carbon interface due to windmill braked engine condition or malfunction in the supply lines. The rotating seal plates receive cooling oil from the bearing inner races for a source of heat rejection which is needed only when the air film breaks down due to one of the above conditions.

Each seal assembly is prevented from rotating by three anti-torque pins. Development on the J58 resulted in the replaceable pin sleeve concept used on the JTF17. This AMS 5759 (L-605) sleeve, by means of its square periphery, also affords longer initial wear by utilizing area rather than line contact.

The No. 1 and 2 bearings compartment also houses the main engine bevel gear train, which supplies power from the N_2 rotor to drive the main accessory gearbox, airframe power takeoff gearbox, and the secondary accessory gearbox drive. The main drive gear is attached to the high rotor shaft with a double piloted spline and secured with a nut and lock-washer. The main drive gear drives three bevel pinion gears, which in turn drive the various required accessories. Each pinion drive is made as a package unit that is installed into the intermediate case. Each of the pinion drive assemblies has one roller bearing and one ball bearing. The roller bearing is located as close to the gear as possible to transmit the radial loads. The ball bearing transmits the thrust loads. All the accessory bearings incorporated anti-torque features built into the bearing to preclude the possibility of race spinning.

The main pinion shaft is designed to incorporate a removable pinion gear. (See figure 38.) This has a dual purpose because it allows the gear to be replaced without replacing the whole shaft; however, the

biggest advantage is the ability to use smaller bearings for the power takeoff drives, thus reducing the weight. Because the size of the tower-shaft is determined by critical speed, the one-piece pinion gear shaft made it necessary to size the bearing so that the ID would pass over the towershaft; however, the split pinion gear shaft eliminates this and provides the lightest weight design.

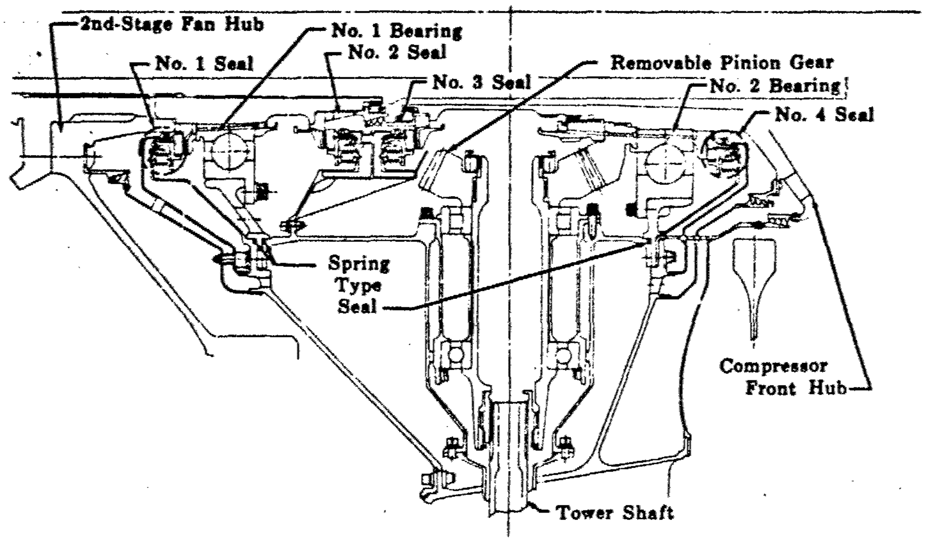


Figure 38. No. 1 and 2 Bearing Compartment Hydrostatic Seal Arrangement FD 16431 IIA

Gear alignment and tooth mesh control is accomplished by controlling the tolerances. Ground spacers were eliminated because they can be incorrectly assembled and cause costly repairs. Experience on the J58 engine, using the controlled tolerance methods for aligning the gears, has been very successful.

Lubricating oil for the No. 1 and 2 bearings compartment is piped through the intermediate case into the compartment. The oil is then distributed inside the bearing compartment to the main engine thrust bearings, seal face plates, gears, and accessory bearings. Each lubricated part of the compartment is supplied by a metering orifice. The oil supplied is filtered through a "last chance" filter located outside of the intermediate case for ease of maintainability. The scavenge oil is returned by gravity through the bottom intermediate case strut to a sump, outside of the intermediate case. The scavenge system is designed to provide adequate scavenging of the oil for 30 seconds at the following attitudes; nose up 30 degrees, nose down 25 degrees, and roll ± 45 degrees.

The No. 1 and 2 bearings compartment will breathe through the three towershafts into the three gearboxes, and in addition through the oil drain passage. This design eliminates the need for separate breather lines.

All the retaining spanner nuts will use a specially designed lock washer that is installed after the nut is torqued and then secured by a locking ring. This is superior to locking devices that are installed

behind the nut. Previous experience has indicated that damage to the locking tabs occurred during the torquing of the nut. This improved locking device eliminates any chance of damage to the locking tangs and ensures a positive lock.

Maintainability was a primary consideration in the design of the No. 1 and 2 bearings compartment. Various designs were studied to obtain one permitting removal of the bearings, seals and bevel drive gear train without further engine disassembly.

By redesigning the forward fan hub it is now possible to remove the fan tiebolts without removing the No. 1 thrust bearing, as was necessary in the initial test engine. This allows the turbine shaft to be supported by the No. 1 thrust bearing in the event that only fan work is required. The No. 1 thrust bearing can now be removed by simply removing the forward cover from the intermediate case and removing the bearing. The whole center section of the compartment can be removed as a unit by simply disengaging the towershafts with a retractable mechanism on the three gearboxes so that gearbox removal is not required. By disconnecting the seal joints and the bolts that attach the center portion of the compartment to the intermediate case, the entire center body of the compartment can be removed as a unit for repair or inspection.

With the center section removed, the main engine drive gear and No. 2 bearing can be removed along with the rear seal assembly. The removal of the No. 2 bearing from the front of the engine without removing the intermediate case results in the following estimated time savings for replacement.

	Elapsed time	Man-hours
Initial experimental engine	242.0	1936.0
Redesign	26.5	95.0

The split intermediate case offers the advantage of bearing compartment inspection without disassembly of the compartment. In addition, it is possible to replace portions of the intermediate case if failures occur.

A review of the Pratt & Whitney Aircraft inflight shutdown records for commercial jet engines now in operation shows that some inflight shutdowns were attributed to bearing failures. The primary failure modes for bearings are fatigue overheating (due to poor lubrication and misalignment), and spinning of the bearing races. The twin-spool turbofan design uses four main engine bearings rather than the conventional seven-bearing configuration. This decrease in the number of main bearings reduces the complexity of the engine, reduces the number of vibration modes, eliminates complicated bearing alignment problems, and reduces significantly the number of critical parts. Fifty percent of the inflight shutdown in the JT3D engines were attributable to intershaft bearings that were eliminated in the JTF17 engine.

Both thrust bearings are located in the forward section of the engine where it is relatively cool.

The primary reason for bearing failure is fatigue. This problem has been improved by using bearings made from consumable electrode vacuum melt.

The functions of the bearing oil supply are first, to cool, and, second, to lubricate. This design minimizes the heat generated in the bearing due to the churning of oil by supplying the major portion of the oil through the inner race for more efficient cooling.

(8) Product Assurance Considerations

Maintainability

1. Provisions are made for easy removal of the fan as a complete unit, for removal of the first stage, and for removal of individual 1st-stage blades. Both thrust bearings are also easily removable from the front while the engine remains mounted.
2. The number of parts, bolted joints and sliding joints are minimized.
3. Standard tools and fasteners are used wherever possible.
4. Configuration of joints consider minimizing risk of incorrect assembly.

Safety

1. Fan cases have been designed to contain failed blades, thereby preventing catastrophic engine failures.
2. The fan rotor is designed to withstand rotor overspeed due to duct nozzle failure.

Growth Potential

1. The twin spool turbofan engine lends itself to increases in engine thrust with increases in fan blade tip diameter.

Reliability

1. Both disks are mounted on double piloted hubs and are not dependent on the tiebolts alone to maintain the integrity of the rotor.
2. Disks have adequate burst and yield margins independently of the axial rim spacer.
3. Blade lock rings must lose all rivets and rotate before a blade may come loose.
4. The fan is designed to withstand blade loss without losing the rotor integrity.
5. The double-shrouded blades can withstand surges and bird ingestion.
6. There are no identical snap diameters in the rotor which might permit misassembly.
7. All rotor retention features, rivets, tablocks, etc., except the 2nd-stage blade lock ring, are visible for inspection from the front of the engine.

8. Blades cannot be installed backward.
9. Fabrication methods, welding, brazing, and riveting techniques are evaluated for all parts to ensure that good inspection methods can be used. Where forging grainflow is required, the consideration is made to integrate functional shape to good forging practice. Material selection also gives consideration to quality predictability. Past experience has resulted in the source control method by Pratt & Whitney Aircraft engineering on certain material processing and manufacturing techniques.

Value Engineering

1. The rotor has a minimum number of parts. Hubs and disks are integral, thereby reducing the number of forgings required which are a large portion of the total cost.
2. Blade material is AMS 4928 and is less costly than PWA 1202 which was used on the initial experimental engine.

(9) Materials Summary

Tables 9 and 10 list the material selected for the major components of the fan and intermediate case. Also included is a representative critical metal temperature and the primary design criteria. In general, titanium alloys are used wherever possible to take advantage of the high strength-to-weight ratio. J58 experience has established the maximum metal temperatures and stress levels for this material although an aggressive development program of new alloys is also in process.

A very important advantage in the use of titanium for disks in a supersonic engine is that the low thermal expansion and modulus of a material increase its capability to sustain greater temperature gradients or improved LCF life for a given gradient. A detailed discussion of this appears in the Compressor Design Approach in subsection 2.

Many of the other materials selected for this engine are based on good performance and fabricability in the J58 Mach 3+ environment. Figures 21 and 22 present experience with titanium on other P&WA engines.

Table 9. Fan Stator Material

Part Name	Material	Typical Standard Day Cruise Temperature, °F	Limiting Mode
Mount Ring and Fan Case	AMS 4966 (A-110)	625	Stress, Containment, and Buckling
1st-Stage Shroud	AMS 5613 (AISI 410)	540	Vibration
1st-Stage Stator	AMS 4928 (6Al-4V)	575	Aerodynamics/ Stress
2nd-Stage Shroud	AMS 5613 (AISI 410)	625	Manufacturing
Duct Exit Guide Vane	AMS 4928 (6Al-4V)	655	Aerodynamics, Manufacturing
Splitter Nose	AMS 4966 (A-110)	655	Aerodynamics
2nd-State Stator	AMS 4928 (6Al-4V)	655	Aerodynamics, Manufacturing

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Table 10. Fan Rotor Material

Part Name	Material	Typical Standard Day Cruise Temperature, °F	Limiting Mode
1st-Stage Disk and Hub	AMS 4928 (6Al-4V)	530	Yield
2nd-Stage Disk and Hub	PWA 1202 (8Al-1 MO-1V)	620	Yield
Rim Spacer	PWA 1202 (8Al-1 MO-IV)	560	Yield
Rotor Tiebolts	PWA 1010 (INCO 718)	540	Yield
Shaft Spanner Nut	PWA 1010 (INCO 718)	500	Deflection
No. 1 Bearing Compartment Primary Labyrinth Seal	AMS 5754 (Hastelloy X)	560	Machining
1st-Stage Fan Blade	AMS 4928 (6Al-4V)	550	Ingestion
2nd-Stage Fan Blade	AMS 4928 (6Al-4V)	630	Vibration
1st-Stage Blade Lock Ring	AMS 4928 (6Al-4V)	530	Yield (local bending)
2nd-Stage Blade Lock Ring	AMS 4928 (6Al-4V)	610	Yield (local bending)
1st-Stage Rear KE Seal	AMS 4928 (6Al-4V)	550	Machining

2. Compressor

a. Design Description

The basic function of the compressor is to continue the compression of 298.7 lb/sec of the 687 lb/sec fan air to an overall engine pressure ratio of 12.97 (takeoff conditions) for the gas generator portion of engine. The major portion of this compressor air is directed to the primary combustor; however, small amounts are extracted for airframe cabin pressurization, anti-icing, engine duct heater fuel pump drive power and compressor and turbine cooling.

The compressor (figure 39) consists of a variable inlet guide vane and six stages of compression. The compressor rotor is on a separate, high speed spool relative to the fan. This permits the fan and the compressor to rotate independently at the most efficient speed relationship for each rotor system. It is thus possible to match the widely varying operational flow requirements of the engine without use of variable compressor geometry downstream of the compressor inlet guide vanes. A variable inlet guide vane is, however, provided at the compressor inlet to serve the dual function of braking the high rotor and to provide some compressor matching over the operating range.

The compressor rotor consists of six bladed disks and a front and rear hub which are held together by axial tiebolts. Disk spacers are an integral part of the 1st, 3rd and 5th compressor disks (3rd, 5th and 7th disks are observed from the front of the engine). Compressor disk cooling is provided to minimize low cycle fatigue gradient effects. The cooling air is introduced to the inside of the rotor by anti-vortex tubes between the second and third disks. The first compressor blades incorporate a mid-span shroud for torsional stiffness, and to damp any possible first bending mode vibration excited by the widely varying inlet condition. The remaining blades are unshrouded.

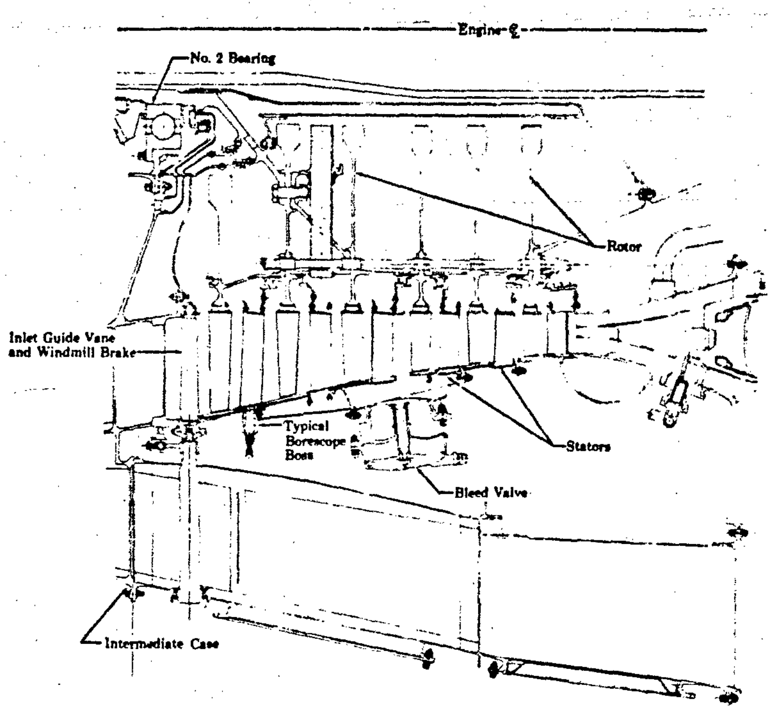

Figure 39. Compressor Assembly FD 17668 IIA

The compressor vanes are retained and supported by shrouds at both the inner and outer diameters. This double (guided cantilever) support provides positive retention of damaged blades and avoids vibrational problems of single ended vane support. All compressor cases are continuous rings as airline service experience has shown that split cases warp and are difficult to assemble after overhaul with attendant increases in blade tip clearance and performance losses. Circumferential knife edged seals supported from the rotating disk spacers form a low clearance air seal with a ring secured to the inner stator shroud to prevent circulation of air around the stator ends.

The cases surrounding the compressor (and fan) are designed to retain failed blades.

An interstage bleed at the discharge of the 3rd compressor stage is provided to assist in start and idle operation.

Turbine power is transmitted through the compressor to drive shafts in the front intermediate case, which in turn furnish power for accessory drives. The compressor-low turbine rotor (N_2) rotates at 8220 rpm at sea level takeoff.

b. Design Objectives and Requirements

The design objectives and requirements for the compressor and consistent with those for the engine as stated in Section I of this report. Specific objectives for the design of the compressor are:

1. High performance, stable operation, light weight
2. Structural integrity and safety
3. Ease of maintenance

4. Life consistent with 50,000 hours airframe life

a.	Front and Rear Compressor Hubs	50,000 hours
b.	Compressor Disks	20,000 hours
c.	Compressor Zee Spacer	50,000 hours
d.	Compressor Blades	50,000 hours
e.	Compressor Stators	50,000 hours
f.	Variable Inlet Guide Vanes	50,000 hours

The disk life based on the detailed design is summarized later in table 18. All other major components exceeded the 50,000 hour life requirements.

c. Design Approach

(1) Compressor Aerodynamic Design

The six stage axial flow compressor is capable of supplying the pressure ratios and efficiencies described in table 11 as well as supplying the transient and steady-state surge margin and distortion tolerance defined in the Performance section of this proposal. It is designed to accomplish this objective with the lightest weight and within a length consistent with the four bearing twin spool concept. The blading operates at Mach numbers and aerodynamic loading levels based on advanced technology rigs and engines as well as the SST initial experimental engine.

Engine matching studies indicated that two points on the flight envelope would be critical to the JTF17 engine. These are the efficiency and flow capacity of the compressor at cruise and the flow capacity of the compressor at the takeoff point. These design requirements are also listed in table 11.

Table 11. JTF17 Compressor Performance

	SLTO	Cruise
Airflow absolute - lb/sec	298.7	118.4
Corrected Airflow - lb/sec	130.3	98.38
RPM - Absolute	8220.	8274
RPM - Corrected	7034	5695
Adiabatic Efficiency - %	85.9	86.8
Flow per Unit Area - lb/sec/ft^2	36.46	27.53
Inlet Total Temperature - ^{o}F	248	634
Inlet Total Pressure - psia	39.4	25.71
Inlet Axial Mach Number (Actual Area)	0.496	0.340
Inlet Axial Mach Number (Effective Area)	0.540	0.367
Exit Axial Mach Number (Actual Area)	0.347	0.409
Exit Axial Mach Number (Effective Area)	0.380	0.471
Tip Relative Mach Number (rotor 3)	0.990	0.693
Tip Diameter (Inlet) - ft	3.162	3.162

Preliminary design work during Phase II on the compressor narrowed the choice of configurations to a large diameter 5-stage compressor and a 6-stage smaller diameter compressor. The 4-stage compressor was

eliminated because it required wheel speed beyond the structural state-of-the-art, whereas the 7-stage compressor reduced the efficiency potential at cruise due to stage mismatching. Aerodynamic and mechanical designs of both a five and six-stage unit were carried through to design layouts, so that a complete comparison could be made of both concepts. See figure 40. The six-stage compressor yielded diameters which are more compatible with the elevation desired by a burner-turbine combination and is 200 lb lighter than the equivalent 5-stage compressor. As a consequence, six stages were selected as the best design for this engine.

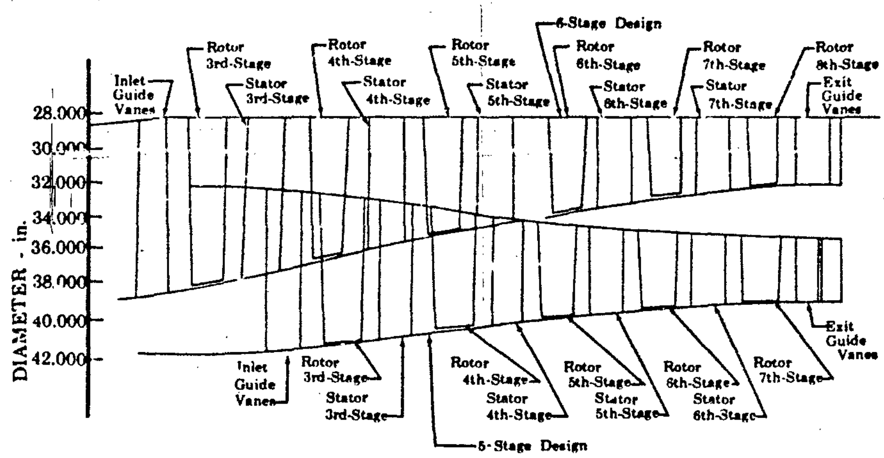

Figure 40. Compressor Comparison Study FD 17506 IIA

The effect of the compressor upon the engine capabilities was continually reviewed as the design progressed to attach a relative level of importance to the various design parameters. A good example of this is the selection of inlet specific airflow. A relatively conservative value of 37 lb/sec ft^2 was selected because frontal area is not as important in this location. However, an aggressive approach was taken in the selection of axial length since engine weight is a strong function of this variable. The chord lengths chosen for this compressor are based primarily on the analysis of the data resulting from the compressor rig program supporting the initial experimental engine. The compressor designed and tested for the Phase II-C engines was intentionally designed with the shortest possible chord lengths to obtain the lightest and shortest overall configuration. Based on this preliminary testing, some adjustments in selective stages will be made. By using this development, the lightest and shortest final configuration will result.

The detailed design of the compressor is essentially a three step process. The three steps are as follows:

1. Selecting an internal flowpath
2. Selecting airfoil geometry capable of supplying the pressure flow capacity desired.
3. Evaluating all the critical "off-design" operating conditions.

This process is iterated upon with mathematical models until best overall performance is attained.

The parameters required to establish the internal flowpath are:

1. Select axial Mach number distribution
2. Select stage pressure ratio distribution
3. Select stage type (reaction)
4. Select radial work and vortex distribution.

The axial Mach number at the inlet is established at 0.54 by the selection of inlet specific flow. The exit value is established as a result of minimizing the length and weight of the compressor and main combustor diffuser. The compressor-diffuser combination is required to deliver airflow to the ram induction burner at a 0.25 Mach number which is dictated by the allowable burner pressure loss. This can be accomplished with a long diffuser and a compressor exit Mach number or by a short diffuser and a low compressor discharge Mach number. To maintain the design level of aerodynamic loading the number of compressor stages must vary inversely with exit Mach number. For this engine the best configuration occurred with a 0.38 compressor exit Mach number. (See figure 39.)

The stage pressure ratio distribution is shown in table 12. The high flight Mach number cruise requirements require a stagewise distribution of pressure ratio different from subsonic engine practice. (See figure 41.) The relative unloading of the front stages results in a compressor having efficient operational capability at the cruise conditions as well as enhancing the distortion attenuation characteristics of the compressor.

Table 12. JTF17 Compressor Performance

STAGE	STAGE PRESSURE RATIO	STAGE TEMPERATURE RISE-°F
3	1.373	81
4	1.347	78
5	1.318	77
6	1.288	76
7	1.255	73
8	1.220	68

The stage type (reaction level) and choice of radial flow pattern are selected by balancing the range capability of each cascade against the range requirement in going from sea level static to cruise. The stage type and flow pattern are largely dictated by the allowable local aerodynamic loadings and Mach numbers. These values are listed in table 13. The design limitations are as established by the NACA Research Memorandum RME 561303 and by Pratt & Whitney experience with advanced technology compressors.

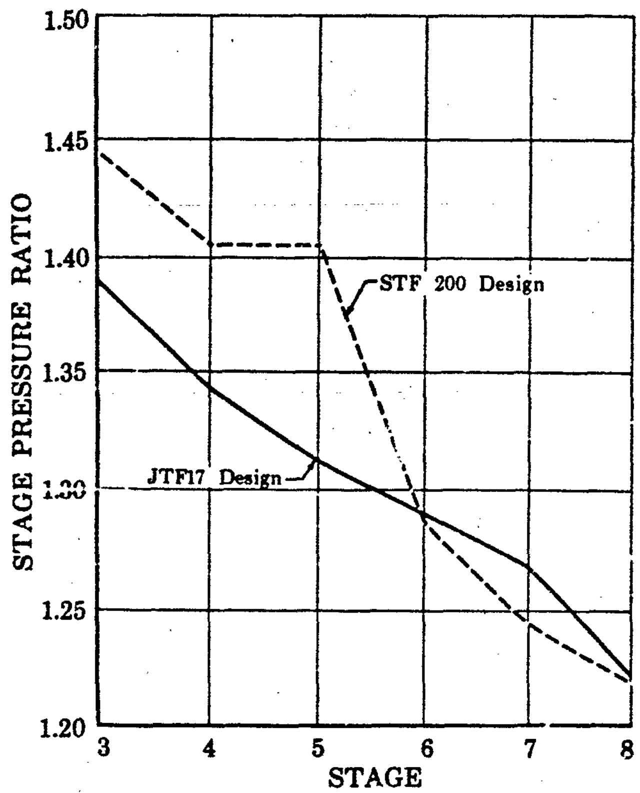

Figure 41. JTF17 Compressor Stage Pressure Ratios

FD 17198
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Table 13. JTF17 Compressor Aerodynamic Loading

	D-FACTOR			ΔP/Q		
ROTOR	ROOT	MEAN	TIP	ROOT	MEAN	TIP
3	0.434	0.419	0.420	0.474	0.480	0.478
4	0.419	0.410	0.425	0.437	0.446	0.456
5	0.402	0.402	0.417	0.433	0.447	0.465
6	0.382	0.397	0.429	0.419	0.442	0.470
7	0.373	0.387	0.421	0.424	0.443	0.472
8	0.374	0.382	0.402	0.447	0.459	0.474
STATOR						
3	0.420	0.408	0.414	0.447	0.416	0.385
4	0.400	0.402	0.417	0.396	0.386	0.363
5	0.407	0.422	0.455	0.421	0.431	0.434
6	0.381	0.396	0.427	0.394	0.398	0.394
7	0.406	0.414	0.438	0.413	0.409	0.392
8	0.464	0.472	0.493	0.426	0.405	0.375

The second major step in the design of this compressor is the selection of airfoil geometry. This again is established in three steps.

1. Select best airfoil shape
2. Select solidity and thickness ratios
3. Select camber levels

The selection of airfoil shape or series is directly related to the relative Mach number and section angle of attack range requirements. (See figure 42.) The selection in the Pratt & Whitney system is based on over 40,000 hours of two-dimensional systematic testing of cascades of airfoils. These data, when combined with the empirical design limits, allow the choice of the best airfoil shape for a given requirement.

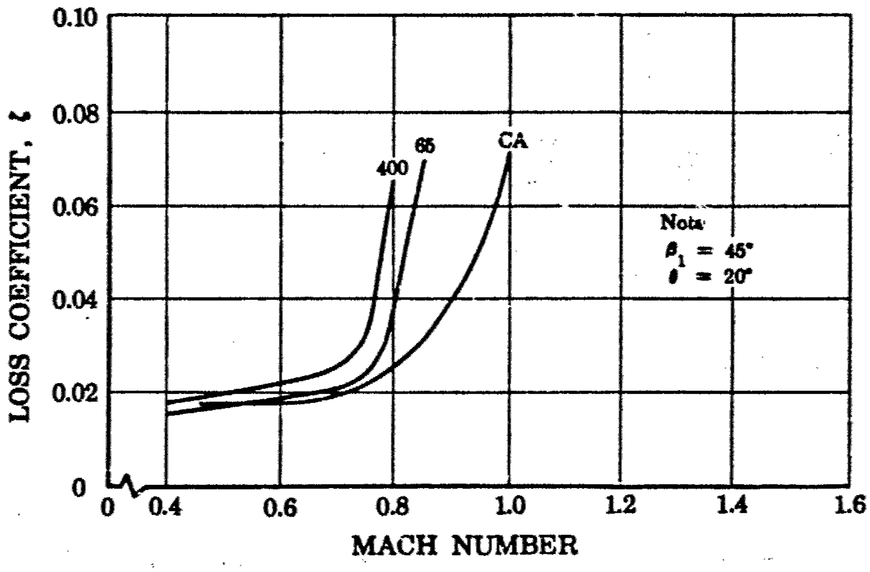

Figure 42. Series Airfoils - Loss Characteristics

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The selection of solidity and thickness ratio is closely interrelated with the structural and aeroelastic criteria. In general solidity is selected high enough to give $D_{factors}$ $\left(\text{as defined in NACA RM 56 1303, } D = (1 - \frac{V_2}{V_1}) + \frac{\Delta C_u}{2V_1}\, \gamma/b\right)$ less than 0.5 and low enough to maintain at least 5% margin from aerodynamic choking to ensure low loss levels. Thickness ratios are set as low as possible consistent with the required structural integrity except at rotor tip sections in the rear of the compressor. Higher thickness ratios than the structural minimum are necessary to supply the necessary range of efficient cascade operation. The values of gap/chord ratio and thickness ratios are listed in table 14.

The selection of camber constitutes a major part of the overall design process. It involves an initial selection, an evaluation of the range capabilities of that selection, and then a comparison of the capabilities

against the range requirements at the critical flight conditions (see fan aerodynamic description for more detailed explanation).

Table 14. JTF17 Compressor Geometric Parameters

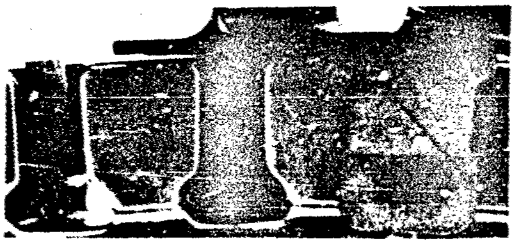
	Gap Chord Ratio			Thickness Ratio		
Rotor	Root	Mean	Tip	Root	Mean	Tip
3	0.700	0.812	0.909	0.079	0.059	0.041
4	0.705	0.808	0.899	0.087	0.078	0.043
5	0.680	0.759	0.831	0.088	0.064	0.042
6	0.679	0.744	0.804	0.088	0.064	0.042
7	0.650	0.702	0.751	0.088	0.064	0.042
8	0.650	0.696	0.739	0.088	0.064	0.042
Stator						
3	0.646	0.748	0.838	0.051	0.072	0.091
4	0.661	0.746	0.822	0.090	0.090	0.090
5	0.657	0.727	0.790	0.090	0.090	0.090
6	0.659	0.717	0.770	0.090	0.090	0.090
7	0.656	0.704	0.750	0.090	0.090	0.090
8	0.657	0.701	0.743	0.090	0.090	0.090

The compressor designed and tested in the initial experimental engine program is described in the Performance section of this proposal. In general, the compressor met its flow, efficiency, and pressure rise requirements at the cruise condition. At the high corrected speed conditions it has surpassed the flow requirements and is significantly higher in efficiency (3 - 5 pts.) than other highly loaded short-chord compressors at the same stage of development, but still 4% lower than the design goal. Surge margin also requires improvement.

There are several significant differences between the compressor designed for the prototype and production engines and the compressor used in the initial experimental engine program.

1. Elimination of deep roots and the attendant root leakage. Extended roots were used in the initial experimental engine to allow engine running before vibration-free aerodynamic performance had been developed. The reduction in performance was accepted as an expedient to allow successful mechanical operation. (See figure 43.)
2. Elimination of variable geometry except for the inlet guide vane (which has shown very low pressure losses).
3. Increase in chord length in areas proven critical in the initial experimental engine compressor rig program.
4. Average running tip clearance is reduced as a result of running cooler disk operating temperature.
5. Camber modifications indicated by the analyses of the multistage rig, by analyses of NASA program (NAS3-7603) and by other P&W Research rigs.

SHALLOW ROOT PROTOYPE ENGINE



DEEP ROOT INITIAL EXPERIMENTAL ENGINE

Figure 43. Revised Blade Attachment With Reduced Leakage — FD 16562 IIA

In the first item listed above the mechanical design of the prototype engine has resulted in reducing the leakage flow to approximately 30% of that in the initial experimental engine. Experience in the TF30 high compressor indicates 7% increase in pressure rise and 2% in efficiency is obtainable by minimizing root leakage. It is estimated that prototype pressure rise capability will be improved by 7% and the efficiency by 2%.

Elimination of the variable stator vanes will result in 1% efficiency improvement. Experience with a TF30 compressor indicates that this improvement can be as high as 3%. (See figure 44.)

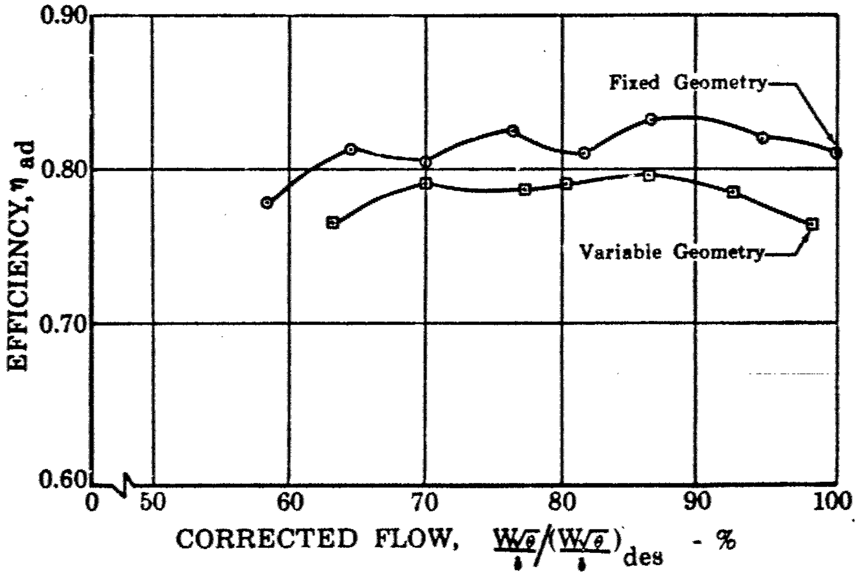


Figure 44. Difference in Overall Efficiency Between Fixed and Variable Geometry from TF30 HPC — FD 17502 IIA

Increasing the chord lengths in those areas proven critical in the initial experimental engine program will result in 5% improvement in pressure rise capability. This is substantiated by a Pratt & Whitney Aircraft 3-stage rig devoted to the evaluation of short chord compressor stages and by detailed evaluation of the Phase II-C compressor rig data.

Camber modifications are necessary to improve efficiency in a short-chord design to allow for a shift in the angular location of minimum loss range and air turning capability. This will result in 2% efficiency improvement and 3% in pressure rise capability.

The combined effects of the improvements listed above will give a compressor capable of operating at a pressure ratio of 4.84 and an efficiency of 86%.

In addition to the performance improvements inherently "built in" to the prototype high pressure compressor design other approaches are available for compressor performance growth. The first of these is the use of slotted airfoils. A systematic program aimed at the development of a design system for the use of slotted airfoils was started under NASA contract NAS3-7603. Data from this program available at the present time indicate a potential efficiency improvement of 3 to 6 points when applied to the JTF17 compressor.

Slotted airfoil concepts that provide suction surface boundary layer control and off-design life improvement for operational aircraft have been applied with an encouraging degree of success to advanced axial flow compressor design. Analytical and experimental effort associated with the J58, JT8D, and TF30 advanced development programs involving slotted vanes and rotor blades indicated a potential for slotted blading to reduce profile loss at high loading levels and to increase operating range. In addition to continued company supported effort to explore this potential in two-dimensional cascade and full-scale engine compressor rigs, P&WA is currently working on a systematic evaluation of slotted vanes and rotor blades under NASA contract NAS3-7603.

An annular cascade investigation of slotted stator vanes was conducted to evaluate slot geometry parameters and slot location to establish preliminary slot design criteria for rotating rig blading. Configurations were tested at slot locations of approximately 55 and 75% chord. Significant loss reduction resulted, (figure 45) when the slots were located at the 55% chord point.

The single stage rotating rig part of the NASA program involves tests of three slotted rotor configurations designed with tip D-factors of 0.46, 0.51, and 0.61 and three stator configurations designed with hub D-factors of 0.52, 0.60, and 0.70. The best slotted rotor and stator configuration as determined from the above tests will be combined in a single stage test to evaluate interaction effects. To date, two slotted rotor configurations and one slotted stator configuration have been tested. A direct comparison of the same blade design with and without a slot is made in figure 45. A nominal loss coefficient reduction of 50% can be seen. Loading level (D-factor) and deviation at the same incidence angle are not appreciably affected by the slot; however, the decrease in loss permits operation over

a larger incidence range which can be used to advantage in one or more of several ways: (1) increase surge margin, (2) increase distortion tolerance, (3) increase loading level, (4) increase efficiency.

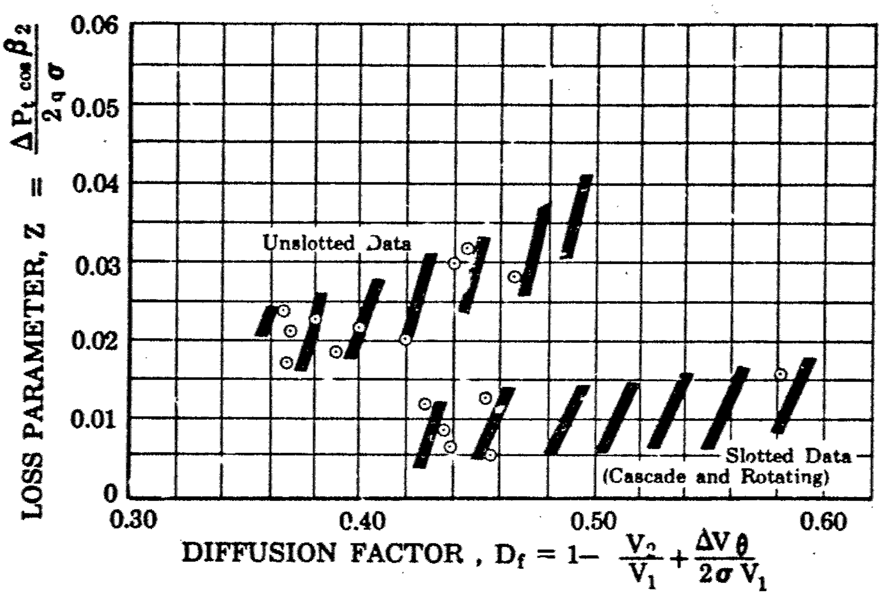


Figure 45. Slotted Airfoil Loss Characteristics FD 17505 IIA

Similar test results have been obtained with the slotted stator configuration having a design hub D-factor of 0.60 (39° camber). The highly loaded slotted stator indicates a lower loss than the unslotted stator that has less loading. The measured D-factor level for the slotted stator is higher than the design value. Stator loss parameter values and loading levels are compared in figure 45 with the annular cascade stator loss data and NASA stator loss data.

If these data are applied to the SST compressor, significant efficiency and/or pressure rise improvements are obtainable. Continuous P&WA effort is directed toward maturing this concept for both structural and aerodynamic application to commercial power plants.

(2) Windmill Brake - Variable Inlet Guide Vane (See figure 46.)

(a) Introduction

With no windmill brake the JTF17 engine will rotate at approximately 80% of normal operating speed on the high pressure spool and 60% on the low pressure spool when windmilling at the supersonic cruise conditions.

The variable inlet guide vane to the high compressor has been selected to provide the windmill brake function because it provides large reductions in windmill speeds for minimum engine weight and complexity.

Failure analyses completed for the JTF17 engine indicate that an aerodynamic windmill brake on the high pressure compressor will permit continuous supersonic cruise operation comparable to present commercial subsonic engine-out operation providing fuel is circulated as a heat sink. If it is necessary to shut off the fuel, an emergency descent to a lower flight Mach number is required to prevent the oil coking and bearing damage which would result from the high ambient temperature. At the subsonic flight conditions the JTF17 will windmill at significantly lower rotor speeds with the windmill brake activated than current commercial engines.

At the request of the airlines, further studies are being conducted in conjunction with the airframe companies to evaluate engine weight and complexity required for further improvements in engine-out performance. These studies to further reduce low rotor speed will include variable 1st stage fan stators, and variable fan exit vanes in conjunction with the high compressor variable IGV, or variable fan second stage vanes. These studies could result in a system which would effectively stop both shafts at subsonic conditions if increases in engine weight and complexity are not excessive.

The present IGV-windmill brake will reduce the low rotor windmill speed to approximately 1400 rpm at cruise conditions and 500 rpm at subsonic conditions, while the high rotor is reduced to approximately 100 rpm at cruise conditions and effectively stopped at subsonic conditions.

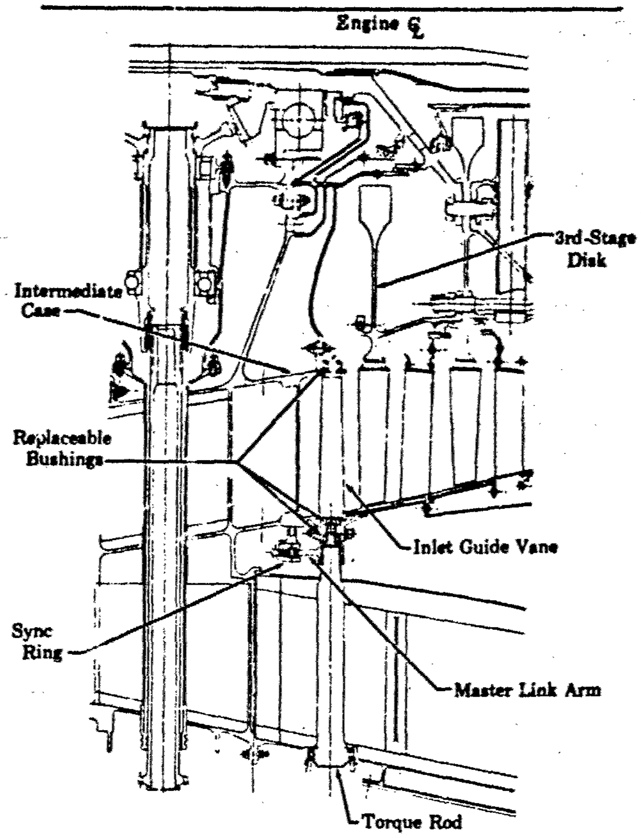


Figure 46. Compressor Inlet Guide Vane Windmill Brake FD 17667 IIA

(b) Design Criteria

The windmill brake must be capable of operating to brake position, on pilot signal, at any point in the flight envelope. The windmill brake must also be capable of movement to a restart position at any point in the flight envelope that is above the minimum windmilling restart envelope.

Control interlocks must be provided to ensure that primary combustor fuel is shut off before actuation of the inlet guide vane to the brake position. Continuing to supply power to the turbine, as the majority of the compressor load is removed, would result in overspeeding, turbine overheating, or both. Similarly, the fuel must be supplied for a restart when the windmill brake is moved to restart position.

(c) Starting

An intermediate vane position, between brake and SLTO, matches the inlet guide vane for higher cruise performance and aids in starting.

(See figure 47.) With only two aerodynamic positions, a two-position hydraulic actuator system may be used. The third position (brake) is obtained by over-riding the hydraulic system with a two position pneumatic actuator system.

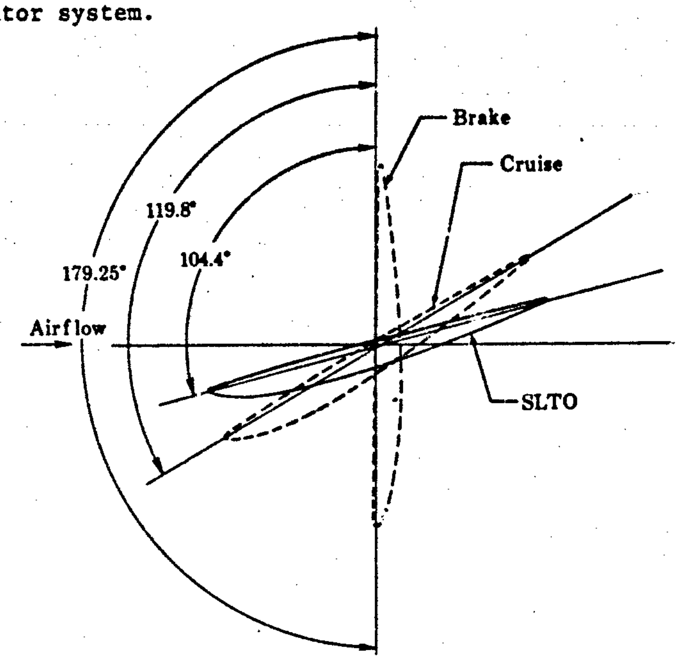


Figure 47. Inlet Guide Vane Windmill Brake Positions FD 16348 IIA

The start-cruise position on the inlet guide vane also has the advantage of reducing the total flow area required for the interstage bleed system by 40 in.^{2} from that required if SLTO inlet guide vane position had been selected for the start position. (See figure 48.)

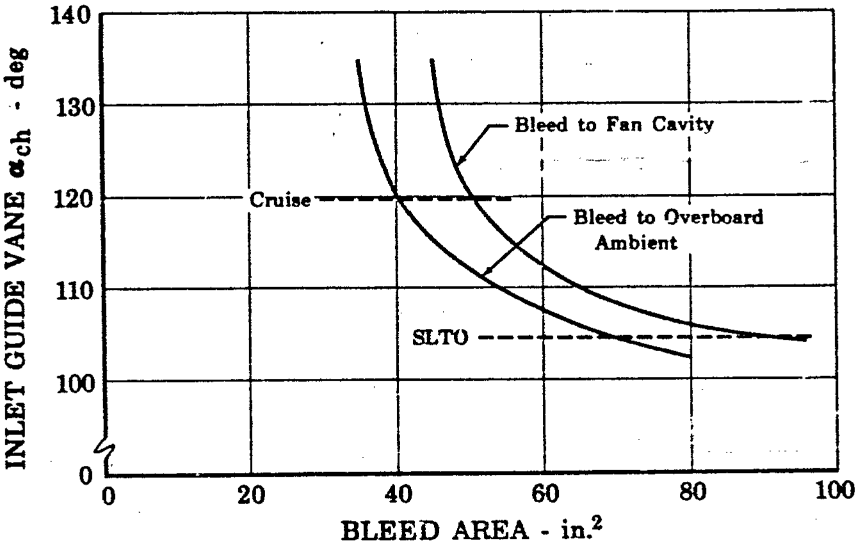


Figure 48. Bleed Area Required at the 5th Stator Exit as a Function of Inlet Guide Vane α_{ch} FD 16350

(d) Failure Condition

Failure mode and analysis studies indicated that a "fail shut" design using a midchord vane pivot was most desirable for the following reasons:

1. Selection of a midchord pivot point results in minimum stress in the foils and attachments, minimizing torque requirements to actuate (close and open), minimum end gap clearances in normal running positions.

2. Due to the bearing and rotor unbalance problems that could result from a windmilling rotor, particularly at supersonic speeds, it is advantageous to have a failed inlet guide vane go to the brake position since the most probable cause of the inlet guide vane failure would have been an engine mode failure that called for brake position. This failure mode probability is further substantiated by the normal operating stress conditions presented in (1) above and (3) below.

3. Built-in safety factors result because the maximum torques required in the normal operating range with the midchord pivot are less than 1/3 the required design torque. Design torque is established by forces required to unbrake through the vane stall angle. Maximum torque required to maintain the vane angle setting at takeoff is less than 1/10 the design torque. (See figure 49.)

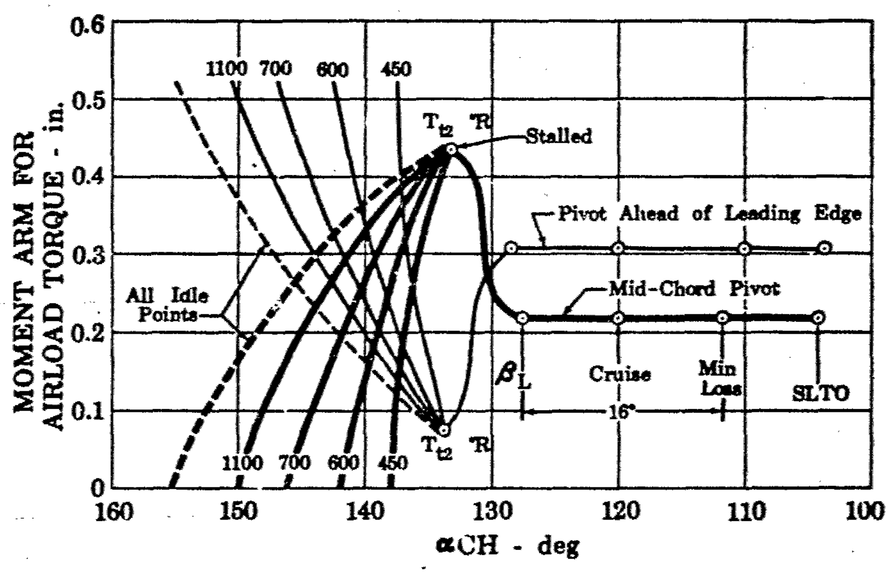


Figure 49. Moment Arm vs Inlet Guide Vane α Chord

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4. Additional fail-safe features include actuation system stops to prevent rotation to brake in event of hydraulic failure unless the brake position is selected.

(e) Structural Loads

The method of determining airfoil stresses and actuator load requirements is discussed in the following paragraphs. The airfoil stresses and actuator load requirements are summarized in tables 15 and 16, respectively.

Table 15. Representative Foil Stresses at Maximum Transient Conditions

(Based on Lockheed Envelope)

Vane Angle (α_{ch}) (Degrees)	Foil Temp (^{o}F)	Foil Bending Stress (psi)	Foil Torsional Stress (psi)	PWA 1202 Allowable Stress (psi)
104.4 (SLTO)	450			82,000
119.8 (Cruise)	700	4800 at ID	1630 at OD	72,000
133 (Stall)	700	37,000 at OD	26,500 at OD	72,000
179.25 (Brake)	700	62,000 at OD*	0	
		46,800 at 40% from ID***	0	100,000**
		74,500 at ID*	0	

Notes: 1. Stresses in the normal operating envelope are 35% lower and temperature 50 degrees less than values shown.
2. *Denotes maximum values - occurs at the maximum pressure, maximum temperature corner on the transient flight envelope (M = 2.9, Altitude = 58,000 ft).
3. **Allowable for titanium in pure bending is 1.4 times 0.2% yield.
4. ***Maximum deflection point = 0.107 inch.

Table 16. Maximum Torque Tube Torques vs Vane Angle

(Two Torque Tubes Sharing Load)

Vane Angle (α_{ch}) (Degrees)	Actuation Direction	Maximum or Minimum Friction Torque (in.-lb)	Maximum Airload Torque (in.-lb)	Maximum Total Torque (in.-lb)
104.4 (SLTO)	Toward Brake	0 (Min)	-102	-102
	Toward SLTO	113 (max)	102	215
119.8 (Cruise)	Toward Brake	0 (Min)	-265	-265
	Toward SLTO	335 (Max)	265	600
133 (Stall)	Toward Brake	0 (Min)	-1340	-1340
	Toward SLTO	760 (Max)	1340	2100
179.25 (Brake)	Toward Brake	0 (Min)	0	0
	Toward SLTO	1150 (Max)	0	1150

From inlet pressure and airfoil parameters of airflow, life and moment arm versus temperature and chord angle, the vane torques and loads normal to the chord line are calculated for SLTO and cruise vane angles. These calculations are made for idle and 100% power ratings at limiting points of the flight envelope.

As chord angle (α_{ch}) increases through the stall angle, the center of pressure moves, as a function of inlet temperature, from approximately 25% chord to the nose of the vane, and then rapidly falls to 50% chord. The pressure center coincides with the rotation center from α_{ch} = 150 degrees to 180 degrees (brake), and the drive torque required is a function of the frictional forces. (See figure 49.)

The maximum bending stress vane load occurs with the vane in the brake position. In this condition the foil aerodynamics collapse, the vane loads are calculated by using idle flow P_{t3} for fan side pressure, and P_{amb} plus 1/4 p_{ram} for downstream pressure. Because the vane twist is less than 5 degrees, the loading across the vane is considered a function of chord length. The section moment of inertia of the foil at the inner end is only 1/10 the section at the outer end. Therefore, the foil is analyzed as a varying section beam with a varying load, and with consideration for the initial end slopes caused by looseness of the end trunnions in the bushings. PWA 1202 was selected for the vane material because it provides the best strength-to-weight ratio of the materials meeting the strength requirements. The maximum outer fiber foil stress occurs at brake condition at the maximum transient Mach number and minimum altitude corner of the operating envelope. (See table 15.) Normal flight envelope maximums will be 35% lower in stress and 50 degrees less temperature.

(f) Inlet Guide Vane and Synchronizing Ring Description

The inlet guide vane aerodynamic brake assembly (figure 46) consisting of the inner and outer support rings, bushing, vanes, linkages, pins, and synchronizing ring is installed or removed as a unit. The synchronizing ring centering supports, employing carbon skids, are attached to the synchronizing ring after the assembly is attached to the intermediate case. The AMS 4926 (A-110) titanium inner support ring and the inner rear intermediate case cover utilize alternate bolts in the bolt pattern so that either may be removed independently. The AMS 4926 (A-110) outer ring is split in the radial plane for vane assembly. Countersunk screws attaching the two rings trap the vanes as a subassembly. The outer rings are later trapped between the intermediate and 3rd-stage case flanges. Snap diameters and jackscrew holes are provided at inner and outer flanges. The compressor assembly is attached or removed without disturbing the inlet guide vane assembly.

Carbon bushings backed with AMS 5616 sleeves are used to support the vane at the inner and outer trunnions. These bushings provide minimum friction and maximum tolerance against grit contamination. Similar bushings are used in the initial experimental engines and have been successful in the J58 variable inlet case. The outer bushings are designed to restrain the vanes radially. Upon possible foil breakage, the inner portion of the foil is prevented from entering the compressor by a safety button attached to the end of the inner trunnion. This button has sufficient clearance for tolerances and thermal transients.

Since the individual vane torque loads are low, simple pin joints are used to attach the link arms to the synchronizing ring. This method is successfully employed in the variable inlet guide vane of the J58. Because long time wear is a possibility, AMS 5616 bushings are provided in the pin holes of AMS 4926 (A-110) titanium links. Loss of the vane set due to wear is then prevented by replacing the bushing. The AMS 5616 pins attaching the link arms to the AMS 4926 (A-110) titanium synchronizing ring are trapped by an AMS 4910 (A-110) titanium sheet metal ring bolted to the synchronizing ring.

(g) Torque Tube Drive Description

Two AMS 5616 master links employ an internal spline for engagement with two AMS 5616 torque tubes. These torque tubes are speared through the appropriate outer duct struts after all major components are assembled. A dry-lubricated stainless steel uniball mounted in the master link arm is used to transmit the driving force into the synchronizing ring.

The torque tube and drive arm is one piece to further reduce angular tolerance error. Thus, the vane chord angle relative to the torque tube angle is accomplished with minimum angle of deviation. The vane angle is verified for any setting by a position indicator on the outer end of the torque tube and duct case. Adjustment to the precise SLTO setting is accomplished in the actuator rod adjustment. Torque tubes are supported at the outer end by ball bearings mounted on the tubes and at the inner end by master link bushings. The ball bearings are installed over the spline end and retained by snap rings. The snap ring is prevented from disengaging by a shoulder on the duct case boss.

(h) Actuation System

The system was designed utilizing a hydraulic-pneumatic system, hydraulic for engine functions, pneumatic override for brake with air pressure being supplied from airframe cabin bleed system.

(i) Hydraulic-Pneumatic System

The actuation system for the inlet guide vane windmill brake is illustrated in figure 50. The actuators consist of tandem hydraulic-pneumatic sections to accomplish inlet guide vane brake requirements as previously stated. Figure 51 illustrates the actuator. The hydraulic pressure is supplied from the fuel pump discharge by means of the inlet guide vane servo located in the fuel control. This supply is selected to allow cooling flow through the actuator while in the brake position. Pneumatic pressure is supplied for emergency (no hydraulic pressure) by the airframe cabin air bleed (CAB) manifold. The CAB manifold supply is produced by the engines remaining in operation.

The actuator hydraulic section piston moves between SLTO and cruise positions. The pneumatic section piston is forced to follow the hydraulic piston toward SLTO by pneumatic pressure on the rod end of the actuator. This pneumatic pressure is always present on the rod end side during normal (SLTO-Cruise) operation. The pneumatic piston is forced by the hydraulic piston toward cruise because hydraulic forces generated are always greater than pneumatic forces.

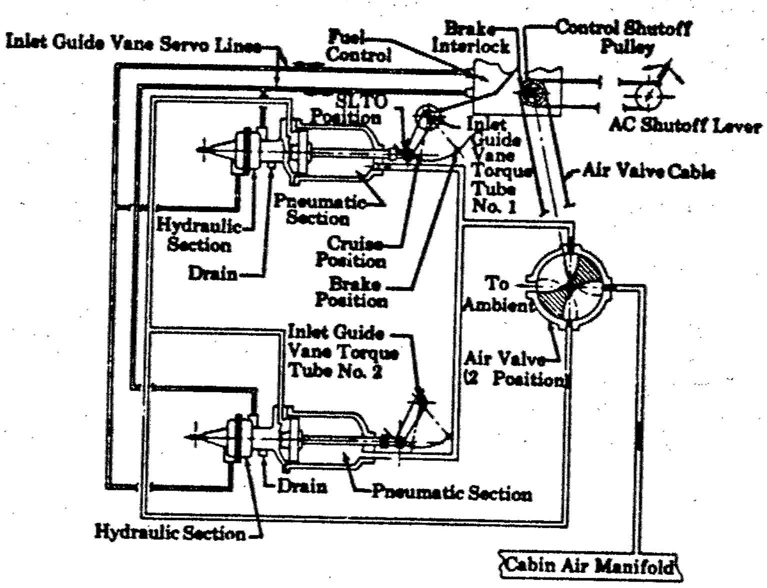

Figure 50. Schematic of Hydraulic - Pneumatic Actuator

FD 16383
IIA

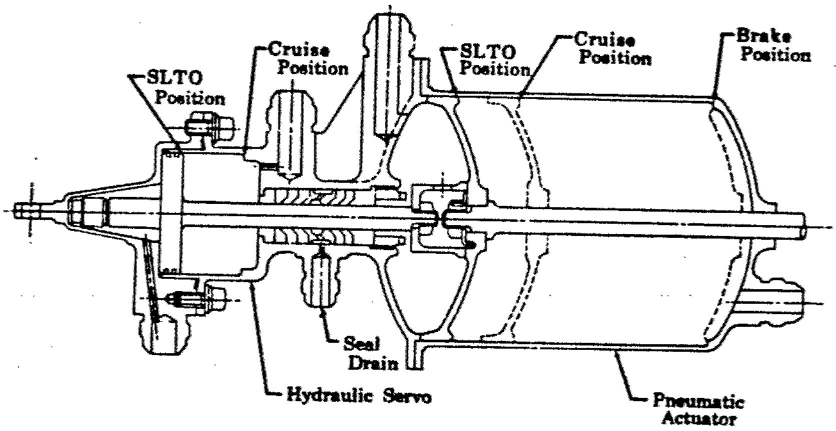

Figure 51. Hydraulic - Pneumatic Actuator for JTF17 Inlet Guide Vane System

FD 16394
IIA

The inlet guide vane remains in SLTO position with a hydraulic failure at SLTO because of the everpresent pneumatic pressure. If a brake position is necessary, a pilot signal causes the fuel control to shut off fuel discharge pressure and the CAB manifold air valve to rotate. The valve rotation reverses the pneumatic piston pressures and forces the piston and connected rod end to brake. The pneumatic piston and rod end separate from the hydraulic piston and rod and reach brake position regardless of the hydraulic piston position or pressures. For a restart under this failure mode, the fuel control discharge must be on for burner supply; thus, hydraulic pressure is again available to move the hydraulic piston to start (cruise vane setting). The air valve is reversed by the restart signal to move the pneumatic piston and rod end toward the normal SLTO-cruise range. Should the original engine shutdown be due to a fuel line failure, meaning no hydraulic pressure is available to accomplish the restart, then a restart is not desirable because of the fuel leakage.

The inlet guide vane remains in cruise position with a hydraulic failure at cruise in spite of the existing rod end pressure by means of a built-in lock. This lock operates through a linkage system controlled by the position of the hydraulic piston. The braking and restart is accomplished as previously discussed.

In the event of an in-flight engine shutdown with fuel cutoff at the firewall, it is desirable to stop both rotors to minimize the possibility of parts damage.

Stopping the rotors is also desirable in the case of an in-flight shutdown due to indications of internal engine damage or for oil loss, where fuel flow would be maintained.

Rotor speed reduction can be accomplished by positioning the high compressor inlet guide vane to the "brake" position. However, to lower rotor speeds to a level which would further reduce the possibility of damage at high Mach number flight, and to completely stop the shafts for subsonic flight, additional braking capability is desirable.

Several methods for increasing rotor braking capability are currently under investigation. Trade studies to determine engine weight and complexity effects of these systems are also being conducted.

The use of fan variable exit guide vanes in conjunction with the high compressor variable IGV is typical of these studies. If an in-flight shutdown was necessary at cruise (Mn 2.7 at 65,000 feet) the positioning of these two variable vanes at the "brake" position would reduce rotor speeds to a level that would prevent damage during descent to subsonic flight speed. The compressor variable IGV alone would reduce rotor speeds to a safe level but not to the speed that would preclude possible parts damage. The two variable vanes would effectively stop both shafts at subsonic condition.

Further studies will be conducted to evaluate all feasible braking systems and to determine their effects upon the engine.

Fuel contamination of the CAB manifold is avoided by means of an overboard vent between the hydraulic and pneumatic systems.

(3) Compressor Rotor

The rotor portion of the compressor has six stages of disk and blade assemblies that are separated and spaced by integral axial extensions on alternate disks. (See figure 52.) Axial tie rods are used to retain the disks and hubs, carry the separation loads, and transmit torque to the adjoining disks. Pilot diameters provide concentricity between successive stages. They are long enough to keep each stage seated on assembly until the final tie bolt torque is applied, assuring positive seating and excellent balance control.

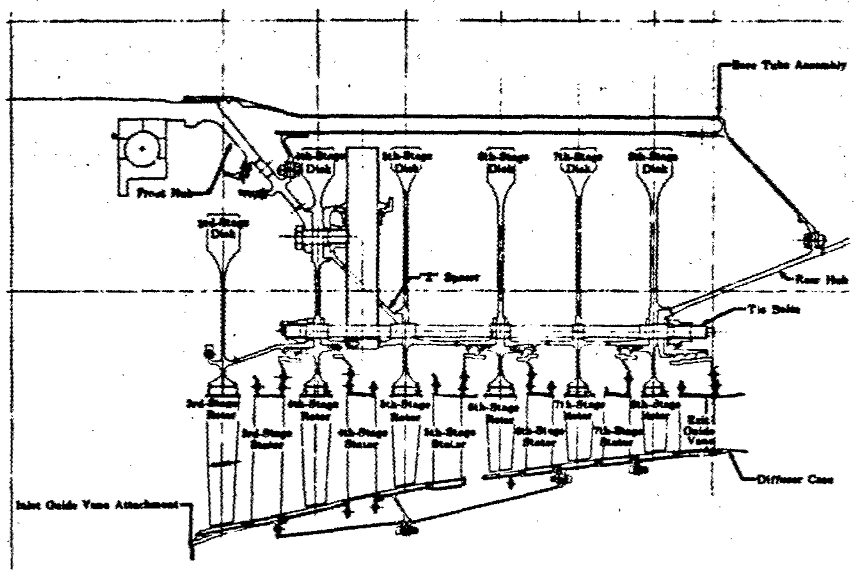

Figure 52. Compressor Rotor and Stator FD 16421
IIA

The Z-spacer between the 4th- and 5th-stage disks provides support for the antivortex tubes and transmits the torque and maneuver loads from the rotor to the front hub.

The blades of all stages are retained radially by a dovetail root section engaged in a matching slot in the disk rim. Titanium blade dovetails (stages three through five) are undercut similar to the fan blades for bearing surface control. The Waspaloy blades (stages six through eight) do not require this undercut due to the very high toughness of Waspaloy.

The blades are retained axially by a double locking device. The aft load which would result in the event of objects passing through the compressor is resisted by a tang made integral with the blade; the forward movement due to the air load is prevented by a wire blade lock incorporating four shear areas. The tang and blade lock are assisted by interlocking blade platforms. In this way, a single blade impact will receive additional support from adjacent blades. This lock replaces the troublesome sheet metal tab locks that are prone to straighten after long periods at elevated temperatures. Figure 53 shows a typical high rotor dovetail and wire blade lock.

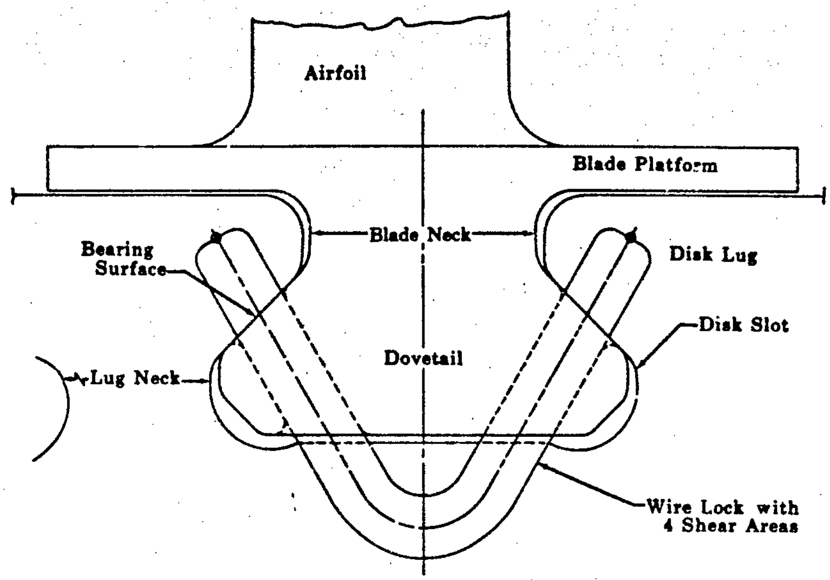

Figure 53. Typical Compressor Blade Attachment FD 16341 IIA

The interstage rotating knife edge seals are restrained to prevent axial movement, and torque is transmitted through tangs to prevent rotation and wear.

Antivortex tubes are used between the 4th- and 5th-stage disks to provide cooling airflow to the disk bores. These tubes prevent the formation of a free vortex within the compressor rotor, which would result in reduced static pressures in the region of the disk bores and low cooling airflow. Without the tubes it would be necessary to tap off a higher compressor stage to achieve the required cooling air pressure, resulting in higher cooling air temperatures. The design calculation procedure used for these tubes is explained in a later paragraph on Low Cycle Fatigue of Disks.

To achieve good balance characteristics, a minimum number of joints are used in the main rotor structures and long pilot engagements have been incorporated on all rotating parts. Detail balancing by removal of material is incorporated on all parts where rebalance is not required. These parts are the front hub, the rear hub, and the bore tube assembly. Dynamic balance of the rotor is accomplished by the addition of weights to mounting flanges located on the front of the third disk and on aft side of the rear seal. To minimize unbalance, the bore tube is made of welded construction incorporating closely controlled wall thickness, concentricity, and pilot diameters.

Rotor tie bolts are sized to maintain rotor integrity despite 10% blade loss and to maintain axial tightness under maneuver loading and for torque transmission. The compressor tie bolts have been designed to the same criteria as described for fan section tie bolts.

To preclude replacement of expensive parts having other functions, the knife edge seals are not made integral with the spacers. The seals were eliminated from the rotor aligning joints as a precaution against balance problems caused by unnecessary flanges. Figure 54 shows an enlarged view of a typical knife edge seal.

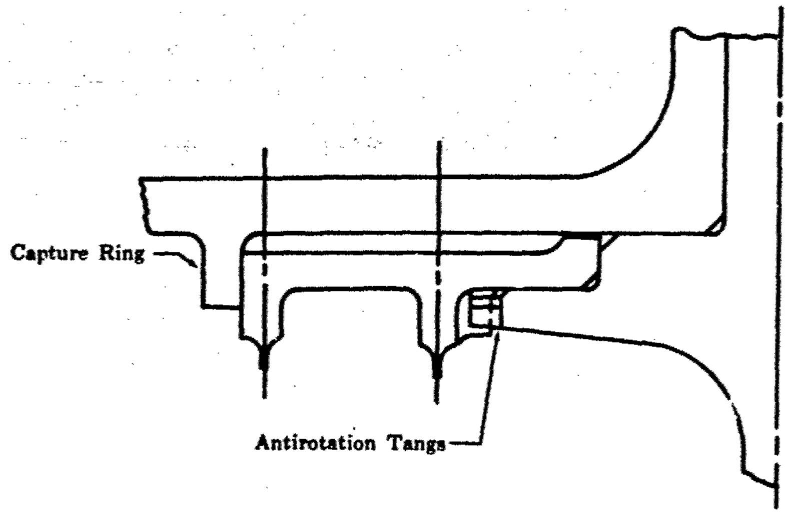

Figure 54. Typical Knife Edge Seal FD 16343 IIA

Foolproof assembly means are provided by incorporating a step on the rear snap of the 6th-stage disk. If assembly of either the 6th- or 7th-stage disk and blade assemblies are attempted backwards, the 0.050-inch radial step will prevent engagement of the pilots. Special configurations and similar snaps on other stages provide similar misassembly protection.

The compressor blades are designed with the ingestion and locating tangs on the forward side. The attachment lengths and radial heights of the blades differ for each stage, thus providing a positive check on misassembly.

Replaceable rings have been inserted on the front hub and Z-spacer to prevent replacement of expensive parts if a wear problem occurred.

Roots of the third through the fifth stages and the disk slots are coated with graphite varnish to prevent galling. This method has been proved on titanium blade roots of the JT4 and JT3D engines. These blades are also shot peened on the root attachment and glass bead peened all over to increase fatigue strength and resistance to stress corrosion.

The nickel base alloy blade roots on the last three stages of the compressor are shot peened to improve fatigue strength. Silver plating of the roots is based on experience gained in the J58 engine, and prevents galling of the bearing surfaces between the dovetail and disk slot.

(a) Disks

The disks for the high compressor are made of forged PWA 1016 (modified) Waspaloy material. The PWA 1016 disks used on the J58 engine demonstrated very high toughness and resistance to the thermal cycling experienced with supersonic engine transient thermal gradients. AMS 5735 (Tinidur) disks were considered in view of lower cost but were rejected in favor of PWA 1016 (modified) which is better by 50% in smooth LCF data and 15% in notched (bolt holes) LCF data.

After rough shapes are machined they are sonic tested for inclusions and foreign deposits. They are then spin tested in the semifinished configuration at sufficient speed to prove that the disk has the minimum strength required for the material. After machining to the final configuration, the disks are vibratory barrel finished. The disk is vibrated in an abrasive slurry to assure smooth, notch-free surfaces in critical areas, such as bolt holes. The edges of the holes are rounded by the slurry and small imperfections are smoothed to prevent stress concentrations.

The compressor disks were determined to be low cycle fatigue limited at transient conditions. Table 17 shows the actual tangential stresses based on LCF and compares these to yield-limited design allowables. Resulting yield and burst overspeed margins are shown in figure 55.

Table 17. Compressor Disk Stress and Computed LCF Life (Material is PWA 1016 Modified)

Stage	Actual Average Tangential Stress, psi	LCF Cycle Life		Limit		
				Location	Condition	
		Thermal Cycles	Acceleration Cycles		Thermal	Acceleration
3	83,400	12,000	100,000	Rim	Cruise	Sea Level Start-up
4	75,500	12,000	30,000	Bolt Circle	Ascent	Sea Level Start-up
5	78,700	60,000	20,000	Bolt Circle	Ascent	Sea Level Start-up
6	84,300	90,000	20,000	Bolt Circle	Ascent	Sea Level Start-up
7	73,600	12,000	100,000	Bolt Circle	Ascent	Sea Level Start-up
8	78,400	100,000	20,000	Bolt Circle	Ascent	Sea Level Start-up
Yield Limited Design Allowable - 102,000		--	--	--	--	--
LCF Limited Design Allowable		12,000	20,000			

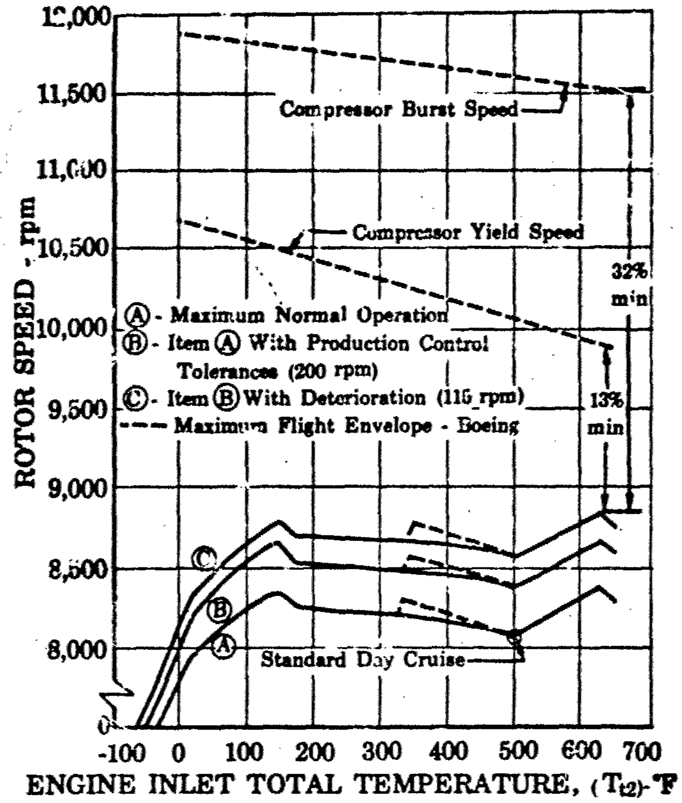

Figure 55. Compressor Speed Margins FD 17507 IIA

The average tangential stress in the compressor is below a yield or burst limit because low cycle fatigue is more limiting.

The compressor disks must be designed to meet the following criteria at the maximum operating conditions:

1. Yield margins - 3% based on maximum normal speed plus production and control tolerances and deterioration.
2. Burst margins - 20% based on maximum normal speed plus production and control tolerances and deterioration.
3. Radial stress in the disk webs should not exceed the allowable average tangential stress.
4. Low cycle fatigue - compressor rotor should be capable of sustaining 20,000 acceleration cycles and 12,000 thermal cycles.
5. Radial stresses between holes in a disk should not exceed 0.9 of the allowable tangential stress.
6. The allowable average tangential stress is 102.000 psi.

Experience on previous P&WA engines is used in establishing the desired material factors and stress ratios of actual stress to allowable stress to produce the high reliability goals. More detail information on the material factors appears in separate discussions on disk yield margins (subsection 1 of this paragraph), disk burst margin and LCF (subsection 2 of this paragraph).

As a standard practice, after the disk is completed the blades are attached and spun at a speed equal to the maximum engine speed. This test opens any small cracks in the surface of the disk, allowing the Zyglo fluid applied during spinning to penetrate. In addition to cracks, the test detects faulty material by control of allowable growth.

(b) Blades

Blade attachment stresses and geometry limits are based on empirical formulas developed by experience on other P&WA engines. This is best explained as allowable stress ratios which have given satisfactory performance on previously tested parts. Refer to table 18 for blade attachment stresses and ratios and table 19 for geometric limits. For description of the geometric limits see page BIIA-24.

Experience on the initial compressor test rig indicated that the 3rd-stage nonshrouded blade was flutter sensitive to the position of the variable vanes. The midspan shrouded 3rd-stage blade allows incidence variation, without flutter, and permits the use of a simplified inlet guide vane actuation system, requiring only two aerodynamic positions. (See figure 56 for shrouds and interlocking platform.) The static shingling parameter for this blade is 0.0645 which is conservative compared to others shown in table 4. The maximum shroud bearing and bending stresses are 4140 psi and 35,800 psi, respectively. Long life allowable stress values are 5000 psi for flameplated contact faces and 54,000 psi for bending.

The 4th-stage blade is also designed to be highly flutter resistant. Although no high flutter stresses were encountered in the fourth stage of the initial experimental engine, the production engine blade was designed with 30% higher torsional frequency to ensure flutter resistance.

Standing wave resonant vibration is one of the most frequently encountered turbine engine vibration problems. Low order excitation from circumferential flow distortion is always present in a compressor and is the result of unsymmetrical airflow, such as might be caused by inlet distortion. Higher order excitation is created by strut blockages in the airstream. The order of excitation equals the number of struts and affects the immediate downstream rotor stages most severely.

The rotor stages of the production engine compressor are designed to avoid all easily excited resonances in the cruise range. The design analysis is accomplished with the methods described previously in the fan vibration bladed-disk analysis.

Figures 57 through 62 are the resonance diagrams for the third through the eighth stages, respectively, of the compressor. As shown in the resonance diagrams, no first, second, or third order resonances are encountered in the operating range; fourth order resonance is avoided at high speed. Disk rims are stiffened to keep second mode, fourth order resonances above the operating range on stages three through six. Blade stiffness properties are adjusted to avoid low order resonances in stages four through six. The 7th- and 8th-stage disks are designed to maintain the first mode of vibration, fourth order resonance below the operating range, and the third order above the operating range. All stages avoid eighth order resonances, which could be excited by the eight intermediate case struts.

Table 18. Blade Attachment Stresses

Item	3rd-Stage PWA 1202 715°F Actual	3rd-Stage PWA 1202 715°F Limit max	3rd-Stage PWA 1205 715°F Actual	3rd-Stage PWA 1205 715°F Limit max	4th-Stage PWA 1202 820°F Actual	4th-Stage PWA 1202 820°F Limit max	4th-Stage PWA 1205 820°F Actual	4th-Stage PWA 1205 820°F Limit max
Blade								
Tensile, psi	11,300	72,000	12,400	79,000	12,200	69,000	13,400	77,000
Ratio	0.16	0.30	0.16	0.30	0.18	0.30	0.17	0.30
Bearing, psi	32,800	72,000	36,000	79,000	38,500	69,000	42,400	77,000
Ratio	0.44	0.90	0.46	0.90	0.56	0.90	0.55	0.90
Bending, psi	18,000	72,000	19,800	79,000	19,500	69,000	21,400	77,000
Ratio	0.25	0.50	0.25	0.50	0.28	0.50	0.28	0.50
Shear, psi	9,500	72,000	10,400	79,000	10,200	69,000	11,300	77,000
Ratio	0.13	0.25	0.13	0.25	0.15	0.25	0.15	0.25
Comb (B+T), psi	29,200	60,000	32,200	68,000	31,700	55,000	34,900	64,000
Ratio	0.49	0.80	0.47	0.80	0.58	0.80	0.54	0.80
Comb (B+T),(scf),(psi)	50,800	60,000	56,100	68,000	55,000	55,000	60,700	64,000
Ratio	0.85	1.20	0.83	1.20	1.00	1.20	0.95	1.20
Disk Lug								
Tensile, psi	17,000	114,000	18,500	114,000	18,000	113,000	19,500	113,000
Ratio	0.15	0.45	0.16	0.45	0.16	0.45	0.19	0.45
Bearing, psi	32,800	114,000	36,100	114,000	38,500	113,000	42,400	113,000
Ratio	0.29	0.90	0.32	0.90	0.34	0.90	0.38	0.90
Bending, psi	15,100	114,000	19,800	114,000	16,300	113,000	18,000	113,000
Ratio	0.13	0.50	0.17	0.53	0.14	0.50	0.16	0.50
Torsion, psi	10,400	114,000	11,400	114,000	14,000	113,000	15,400	113,000
Ratio	0.09	0.40	0.10	0.40	0.12	0.40	0.14	0.40
Comb (B+T)	32,100	114,000	38,300	114,000	34,300	113,000	37,500	113,000
Ratio	0.27	0.65	0.34	0.65	0.30	0.65	0.33	0.65
Blade at Maximum rpm								
Bearing, psi	35,100	72,000	38,600	79,000	41,100	69,000	45,400	77,000
Ratio	0.49	1.10	0.49	1.10	0.60	1.10	0.59	1.10
Comb (B+T),(scf),(psi)	54,400	60,000	60,000	68,000	59,500	55,000	65,000	64,000
Ratio	0.91	1.40	0.88	1.40	1.08	1.40	1.02	1.40

Item	5th-Stage PWA 1202 900°F Actual	5th-Stage PWA 1202 900°F Limit max	5th-Stage PWA 1205 900°F Actual	5th-Stage PWA 1205 900°F Limit max	6th-Stage PWA 1016 990°F Actual	6th-Stage PWA 1016 990°F Limit max	7th-Stage PWA 1016 1060°F Actual	7th-Stage PWA 1016 1060°F Limit max	8th-Stage PWA 1016 1125°F Actual	8th-Stage PWA 1016 1125°F Limit max
Blade										
Tensile, psi	8,600	62,000	9,400	76,000	15,200	110,000	16,600	109,000	13,400	105,000
Ratio	0.14	0.30	0.12	0.30	0.14	0.30	0.15	0.30	0.13	0.30
Bearing, psi	26,200	62,000	28,750	76,000	39,600	110,000	46,400	109,000	40,200	105,000
Ratio	0.42	0.90	0.38	0.90	0.36	0.90	0.43	0.90	0.38	0.90
Bending, psi	12,300	62,000	13,540	76,000	18,000	110,000	19,700	109,000	15,800	105,000
Ratio	0.20	0.50	0.18	0.50	0.16	0.50	0.17	0.50	0.15	0.50
Shear, psi	6,900	62,000	7,560	76,000	11,300	110,000	12,500	109,000	9,900	105,000
Ratio	0.11	0.25	0.10	0.25	0.10	0.25	0.11	0.25	0.09	0.25
Comb (B+T), psi	20,900	38,000	22,950	62,400	33,300	95,100	[illegible]	96,100	29,100	68,000
Ratio	0.55	0.80	0.37	0.80	0.35	0.80	0.38	0.80	0.43	0.80
Comb (B+T),(scf),(psi)	36,400	38,000	40,000	62,400	57,900	95,100	63,000	96,100	50,600	68,000
Ratio	0.96	1.20	0.64	1.20	0.61	1.20	0.66	1.20	0.74	1.20
Disk Lug										
Tensile, psi	11,270	111,000	12,110	111,000	17,500	110,000	21,200	109,000	19,400	105,000
Ratio	0.10	0.45	0.11	0.45	0.16	0.45	[illegible]	0.45	0.18	0.45
Bearing, psi	26,200	111,000	28,750	111,000	39,600	110,000	46,400	109,000	40,200	105,000
Ratio	0.24	0.90	0.26	0.90	0.36	0.90	0.43	0.90	0.38	0.90
Bending, psi	7,640	111,000	8,400	111,000	19,000	110,000	20,700	109,000	16,600	105,000
Ratio	0.07	0.50	0.08	0.50	0.17	0.50	0.19	0.50	0.16	0.50
Torsion, psi	5,530	111,000	6,080	111,000	10,800	110,000	14,400	109,000	15,600	105,000
Ratio	0.05	0.40	0.06	0.40	0.10	0.40	0.13	0.40	0.15	0.40
Comb (B+T)	18,910	111,000	20,510	111,000	35,500	110,000	41,900	109,000	36,000	105,000
Ratio	0.17	0.65	0.18	0.65	0.33	0.65	0.38	0.65	0.34	0.65
Blade at Maximum rpm										
Bearing, psi	28,100	62,000	30,830	76,000	42,400	110,000	49,700	109,000	43,100	105,000
Ratio	0.45	1.10	0.41	1.10	0.39	1.10	0.46	1.10	0.41	1.10
Comb (B+T),(scf),(psi)	39,000	38,000	42,800	62,400	62,100	95,100	67,500	96,100	54,200	68,000
Ratio	1.03	1.40	0.69	1.40	0.65	1.40	0.64	1.40	[illegible]	1.40

Stress limits are either stress rupture or yield allowables.

Control of these variables require an accurate method of prediction for these values. Pratt & Whitney Aircraft has developed such a method based on fundamental aerodynamic and thermodynamic theory and on design limits resulting from more than 500,000 hours of rotating turbomachinery and cascade testing. These systems have been programmed into computing machines so that rapid and accurate solutions to design and development problems may be obtained.

Particular care must be taken in blade design to ensure that no failures result from surge. During compressor surge a few cycles of self-excited bending mode vibration are experienced by each stage as it is driven into a severely stalled flow operating region. The number of cycles of stress (from surge) that accumulate in the life of an engine is too small to be an important consideration, but it is necessary to provide sufficient clearance to prevent blade-to-vane contact in surge. Clearance requirements are based on experience with the JT8D and J58 compressor as well as with the initial experimental JTF17 engine compressor rig.

Table 19. Blade Root Attachment Geometric Limits

	3rd Stage	4th Stage	5th Stage	6th Stage	7th Stage	8th Stage	Min Limits
Neck Ratio	1.005	0.8504	1.184	1.122	0.9339	0.8460	0.800
"Z" Ratio	6.41	4.446*	4.442*	4.446*	4.800	5.575	4.44
Lug "Z"/Foil "Z"	8.77	4.959	8.227	7.1	5.5	5.2	1.45
B/2	0.70	0.547	0.706	0.647	0.500*	0.605	0.500
Ratio "R"	0.255**	0.1572	0.368	0.3202	0.2371	0.1500*	0.150

* Limiting Geometric Factor
** Vibration Limited

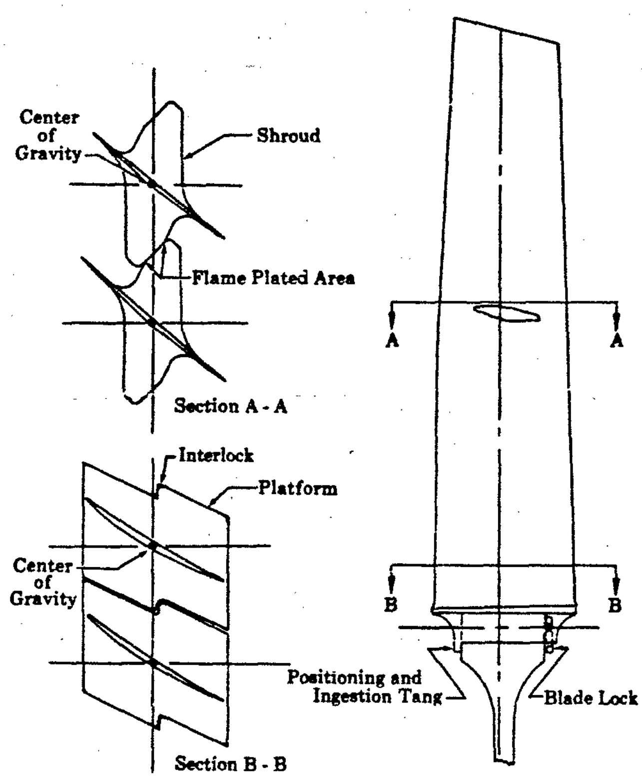

Figure 56. 3rd-Stage Blade Shrouds and Interlocking Platform

FD 16345
IIA

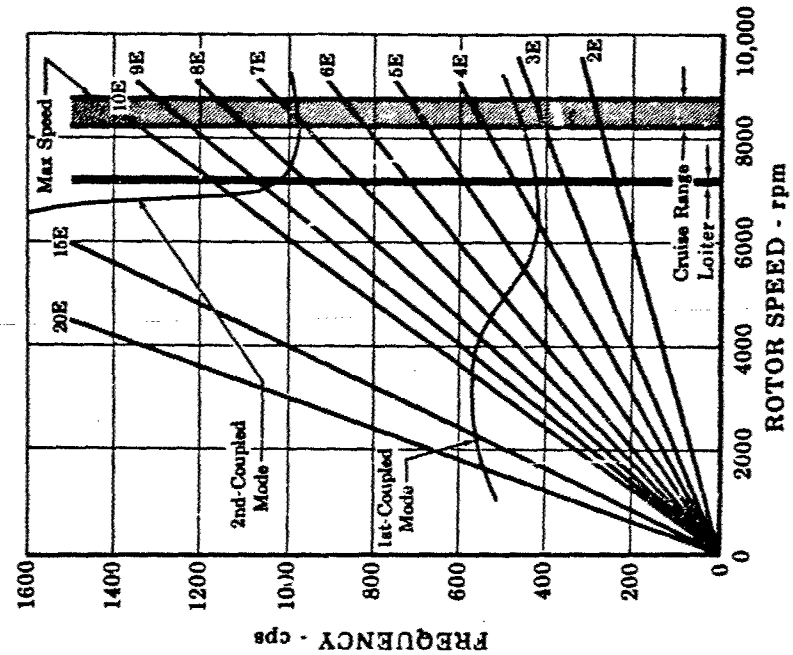

Figure 57. 3rd-Stage Compressor Vibratory Modes FD 16352 IIA

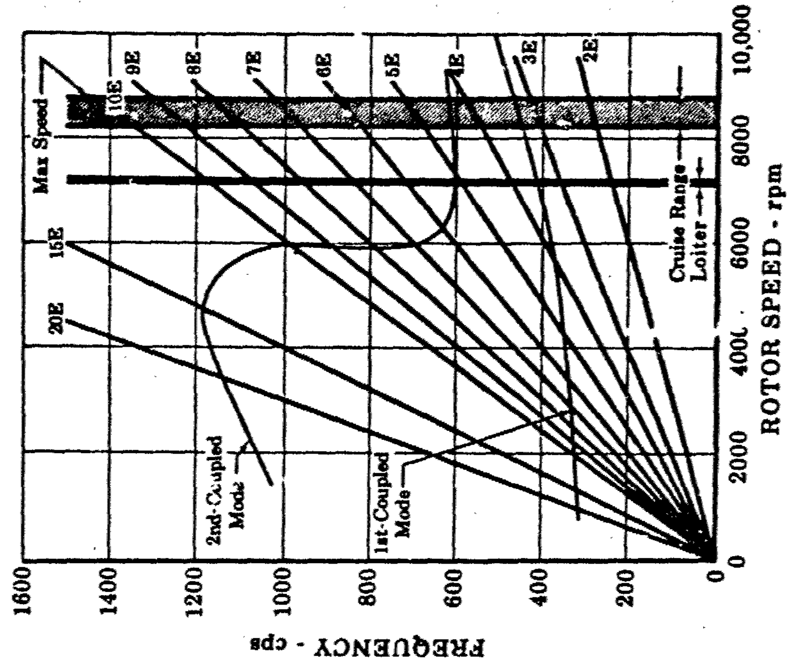

Figure 58. 4th-Stage Compressor Vibratory Modes FD 16353 IIA

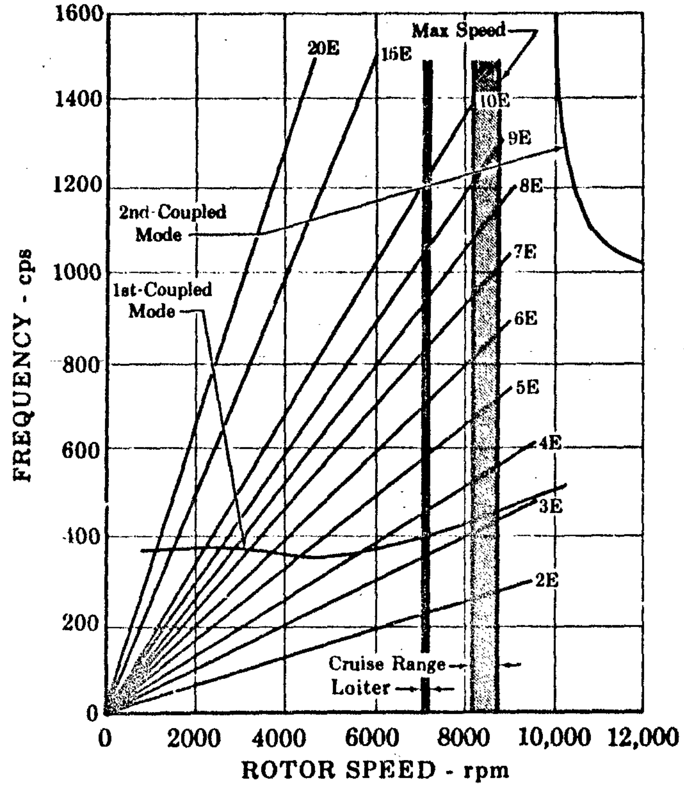

Figure 59. 5th-Stage Compressor Vibratory Modes FD 16354 IIA

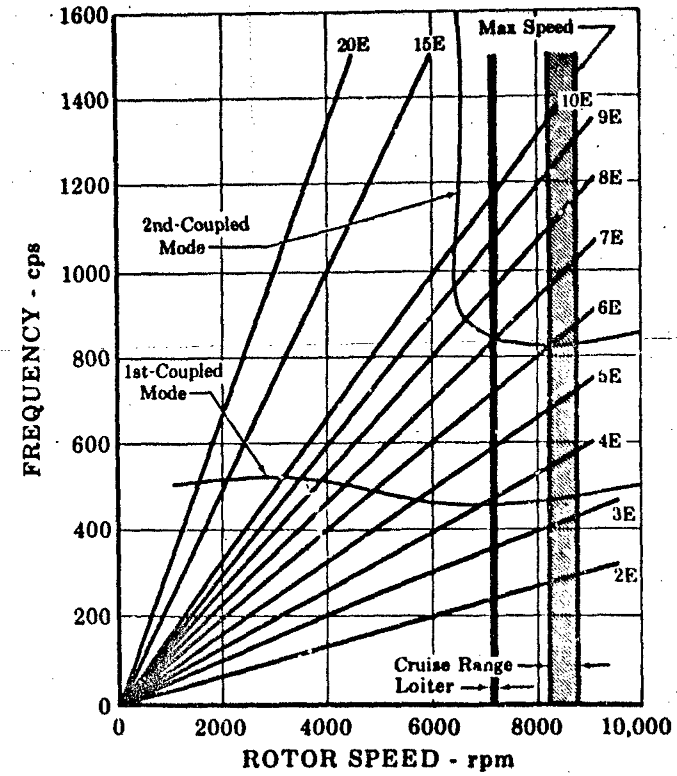

Figure 60. 6th-Stage Compressor Vibratory Modes FD 16355 IIA

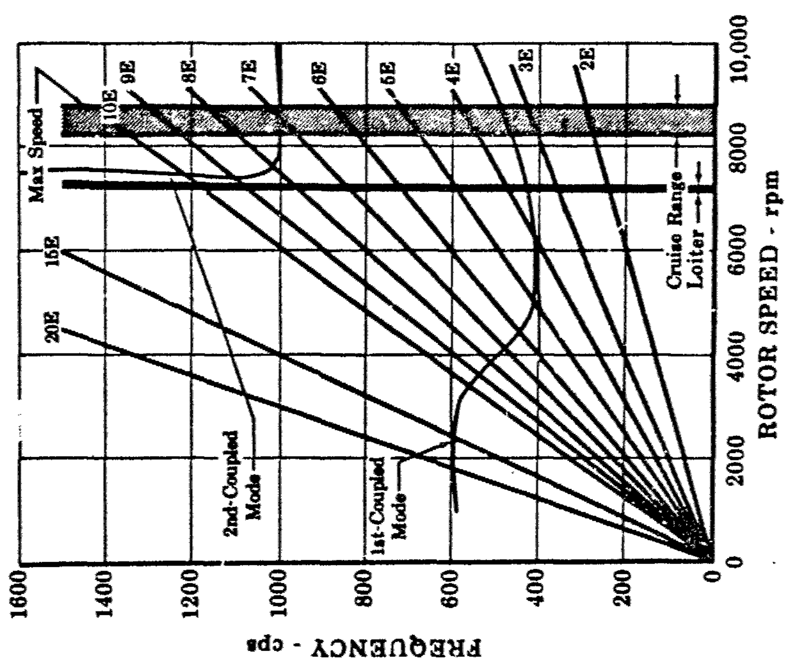

Figure 61. 7th-Stage Compressor Vibratory Modes

FD 16356 IIA

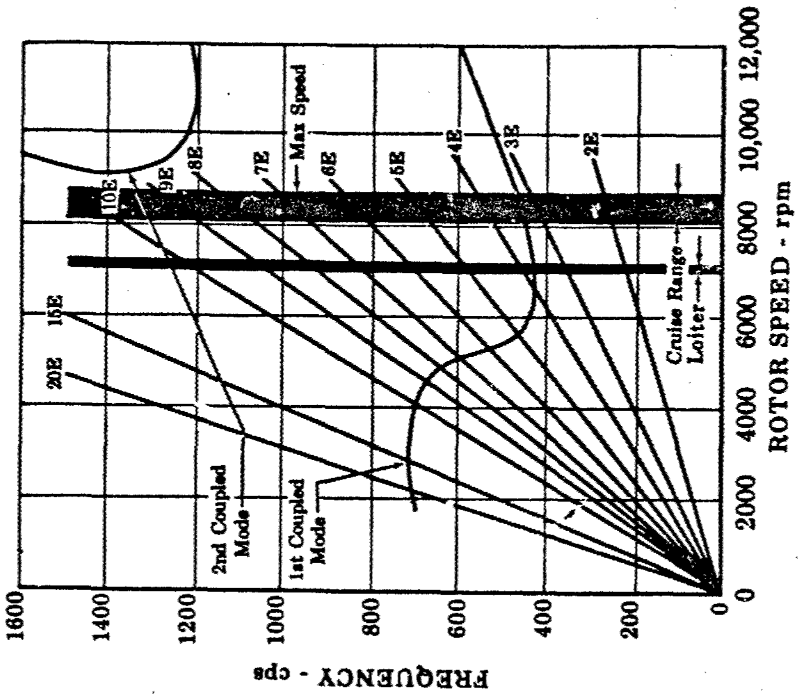

Figure 62. 8th-Stage Compressor Vibratory Modes

FD 16357 IIA

The spacing between blade and vane platforms at the flow path inner wall is gapped for all flight conditions including surge deflections. Using this gap as a base, other spacing between rotating and stationary parts such as the seal lands for the high rotor are made larger. This is done to prohibit simultaneous hard rubs that would increase heat generation and failure potential. A flutter vibration amplitude for each stage is determined from the maximum vibratory stress level that is calculated from the amount of coupling between the blade and disk vibration in surge flutter.

(c) Stackup Techniques Summary

The procedure of assembly is similar to that of other commercial P&WA engines. Tiebolts, in addition to having an angle of turn and stretch measurement capability, have additional thread length permitting the use of hydraulic stretch equipment as used in the JT8D.

Disk snap fits are such that conventional heating and cooling methods will permit easy engagement of each snap. Suitable provisions are made to permit hydraulic jacking of the snaps for disassembly in areas close to the snap diameter, such as additional holes in hub and spacer flanges and the 5th-stage disk.

To achieve good balance characteristics, experience in early JT8D's and the J58 led to the approach of reducing the total number of snap fits by using the integral part concept, combining the cone and disks with spacers, and limiting the number of overhung stages.

Long pilots help to ensure that stackup seating is retained until the final bolt load is applied.

Initial unbalance of the Z-spacer and tubes assembly is controlled similar to individual stage assemblies; that is, tube or blade arrangement capability is possible at the balance assembly build stage. Other balance capabilities of details and the dynamic rotor assembly are described in paragraphs (7)(a)(Maintainability) and (7)(b)(Reliability).

(d) LCF Description

Low cycle fatigue (LCF) is a cracking failure mode, which results from repeated plastic straining of a local area on a part. In a compressor rotor the number of LCF cycles is affected by several factors. These factors fall into two distinct categories: Those controlled by the aircraft environment and those controlled by the local rotor environment.

1. Aircraft Environment Controlled Factors
   a. Engine power setting
   b. Rotor speed acceleration rate
   c. Ascent and descent rate
   d. Engine inlet temperature and pressure.
2. Rotor Environment Controlled Factors
   a. Compressor rotor cooling system
   b. Radial and axial mass distribution of the disk material
   c. Resulting magnitude and shape of disk radial temperature gradient.

To fully evaluate the degree of resistance against LCF failure, the above factors, which constitute the installed engine dynamic and thermal environment, were analyzed. However, the analysis of the number of LCF cycles reduces to a calculation of the local stress level and distribution. The strongest influences on resulting stress were rotor speed and radial temperature distribution.

(e) LCF Design Objectives

The major objective in the design planning of the rotor thermal environment was to obtain the specified number of LCF cycles with minimum rotor weight compatible with engine and airframe life objectives. This had to be accomplished with a high degree of reliability without compromising thrust balance, performance, seal pressurization, or turbine cooling requirements.

The specific objectives relative to these subsystems were:

1. The coldest possible air temperature source for disk cooling
2. A cooling system that resulted in minimum radial flow into the main gas path so as to maximize compressor aerodynamic performance
3. A cooling air flow path in which flow rates could be changed without adversely affecting thrust balance
4. A disk bore temperature environment that could not be affected by air from high temperature stages
5. A rotor configuration in which the temperatures, pressures, and flow rates adjacent to disks could be accurately defined for minimum design risks.

(f) LCF Design Requirements

Commensurate with design criteria listed in the following paragraph (h), the rotor is required to experience 20,000 acceleration cycles and 12,000 mission cycles without failure. The acceleration (idle to sea level static) cyclic requirement is more severe than the mission requirements to account for the number of times an engine is accelerated to rated power in addition to completing a mission.

As previously stated, the aircraft environment plays an important role in determining cyclic life. The rotor acceleration rate and the aircraft ascent and descent rates were the most important aircraft-controlled factors to be considered.

The selection of environmental criteria was based on the current economic ground rules (FAA Report No. 65-7) and on information subsequently supplied by The Boeing Company and Lockheed California Company.

The specific rates used were:

1. Ground rotor acceleration (idle to full power): 4400 to 8220 rpm in 5 seconds
2. Ascent and descent rate to and from cruise Mach number: Engine inlet temperature rate of change of 1.5 degrees per second.

(g) LCF Design Approach

Initial studies considered using engine 3rd stage (compressor 1st stage) air as a source for disk cooling. The temperature level was ideal for rapid disk metal temperature response but the pressure level was too low for the required flow. Subsequent studies led to the selection of 4th stage air. This source, in conjunction with antivortex tubes to compensate for radial inward flow pressure drop, resulted in acceptable bore air temperature and pressure levels.

The next step was to design a flow system in which the bore cooling air would contact all disks and yet be separated from higher temperature compressor discharge air. This was accomplished by incorporporating a bore tube that returns the cooling air to the front of the high compressor where it is discharged through the intermediate case to the fan cavity.

The final cooling air flow path, which resulted in a disk thermal environment that satisfied LCF requirements, is shown in figure 63.

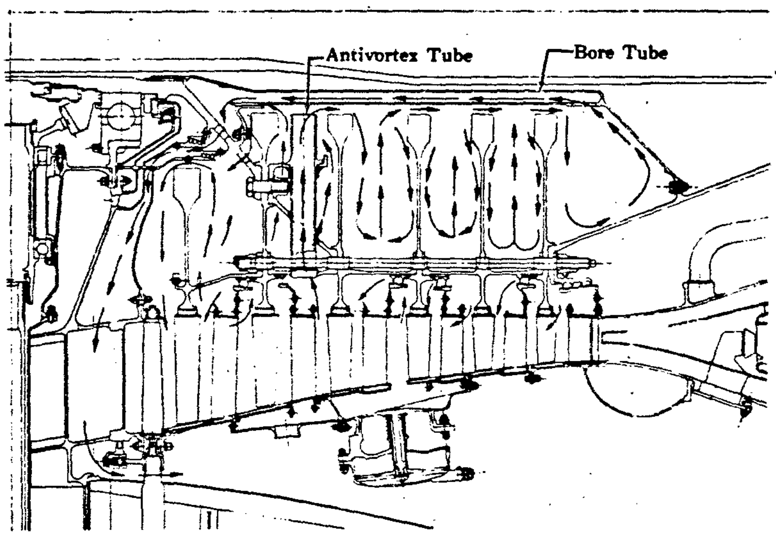

Figure 63. Compressor Disk Cooling System FD 16417 IIA

To evaluate disk transient thermal response, a detailed analysis was performed on the most severe conditions of a typical aircraft mission cycle. This analysis was performed by creating a mathematical model of the entire rotor and programming all pertinent parameters on a digital computer. The computer program, in essence, simulated the total in-flight rotor environment.

To assure that optimum weight had been realized, the disks were first analyzed on a stress-limited basis. This means that the disk configuration was determined first from the standpoint of a noncyclic stress criterion such as yield, burst, and creep. This disk configuration was then adjusted to account for the specific number of dynamic and thermal cycles.

Several supporting studies were conducted in conjunction with the main effort to obtain the basic cooling air flow path. One of these studies involved optimization of the bore cooling air flow rate. Results of this study are shown in figure 64. The curve shows that increasing the flow beyond 0.6% of gas generator flow offered little advantage in decreasing disk temperature gradient.

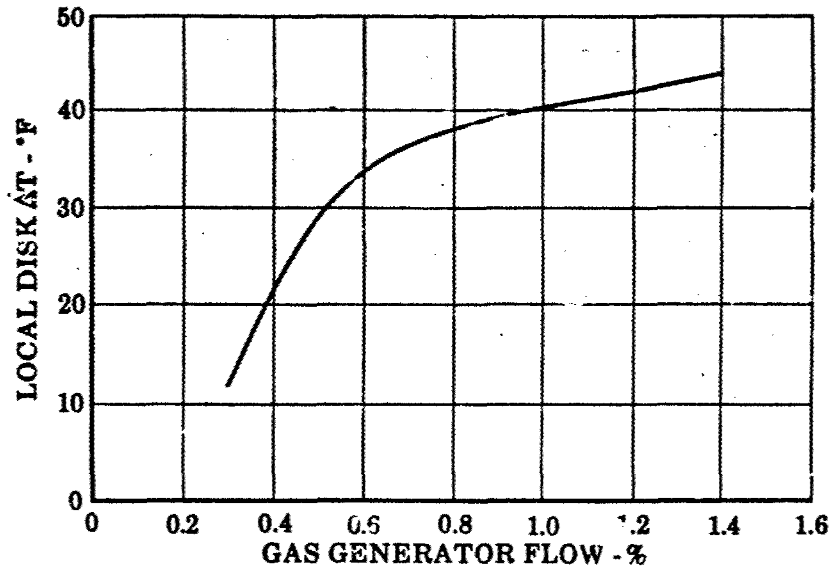

Figure 64. Disk Temperature vs Gas Flow FD 16366 IIA

The substantiation of the need for a design using cold-bore cooling air is illustrated in figure 65. This curve showed the relative weight required to obtain cyclic life with a hot-bore and cold-bore configuration. The hot bore configuration is that used in the initial experimental engine.

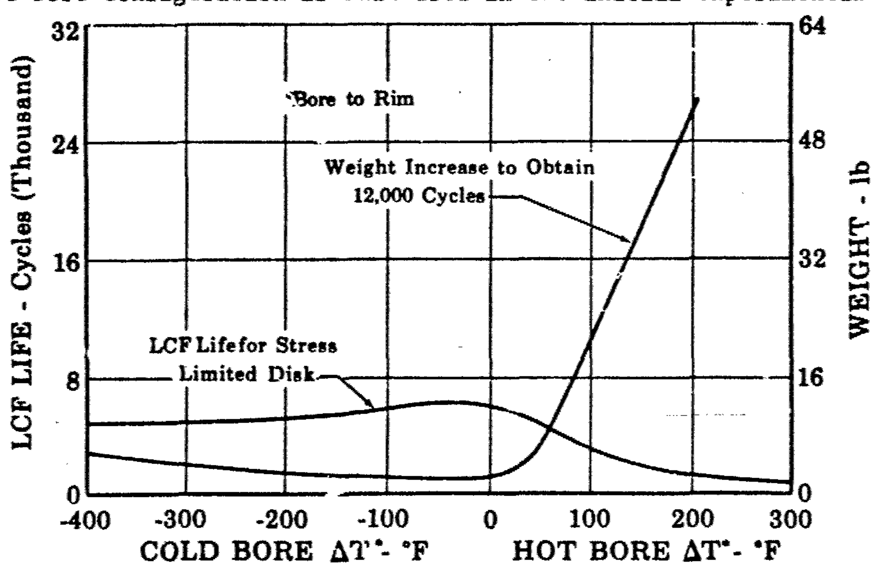

Figure 65. Cold Bore vs Hot Bore LCF Life and Weight Trends FD 16359 IIA

In the course of the J58 engine development program, considerable emphasis was placed on simulated in-flight testing for LCF evaluation.

The high Mach number test programs were instrumental in determining critical flight modes that influenced low cycle fatigue. An example of this was a planned experiment in which a J58 engine descended from high Mach number operation at varying rates of descent. The disk bores and rims were instrumented with thermocouples and the maximum disk transient temperature gradient was recorded as a funtion of descent rate. The results of this study are shown in figure 66.

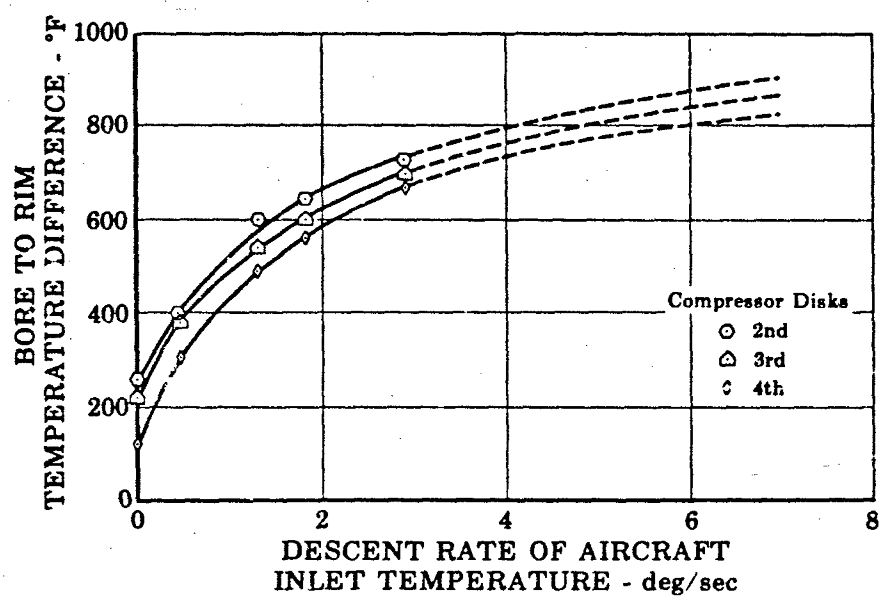

Figure 66. J58 Test Data Compressor Disk ΔT vs Descent Rate — FD 16360 IIA

These J58 engine test results led to a JTF17 study in which the roto: weight required to reach the specified LCF cycles was evaluated as a function of rotor speed and descent rates. Figure 67 illustrates the critical influence of aircraft environment on low cycle fatigue life.

The principal design characteristics that resulted in the attainment of specified LCF cycles in the JTF17 engine were:

1. A cooling configuration that uses cool air bleed from a low compressor stage (this was a critical factor for obtaining minimum weight).
2. The incorporation of a bore tube that is designed so that cooling air passes directly over disk bores at a rate commensurate with desired metal temperature transient response.
3. A single spacer rotor design in which all disk surfaces are scrubbed by cooling air.
4. The use of a free vortex eliminator resulting in maximum pressure potential at lowest temperatures for flowing air through the cooling path.

5. The incorporation of a blade attachment method that has been proved by experience and, therefore, involved minimum design risk. Buried or extended root blade attachments, such as used on the initial experimental engine, and their attendant sealing problems are avoided.
6. A total rotor configuration in which LCF cycles could be determined analytically by techniques proved on the J58 and commercial engine programs.

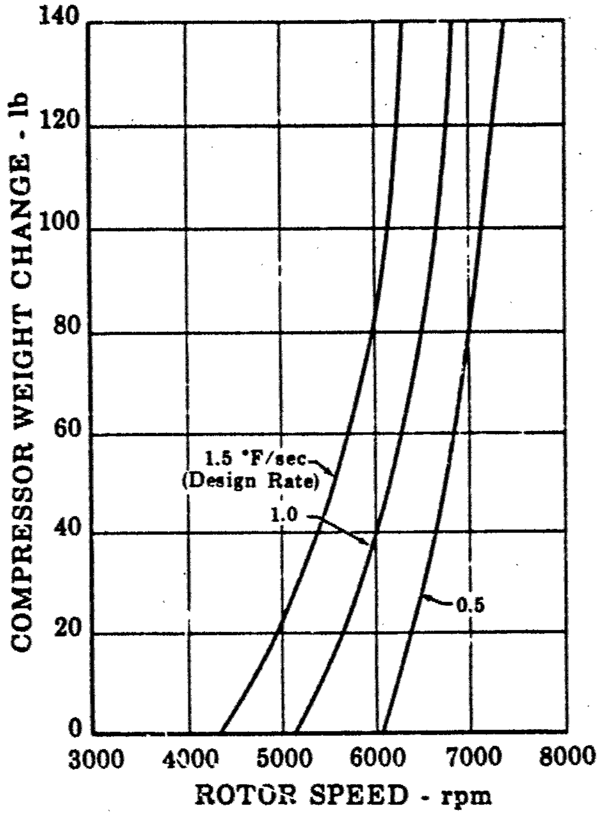

Figure 67. Compressor Rotor Weight Tradeoff as a Function of Descent Rate and Rotor Speed — FD 16361 IIA

The steady-state and transient disk radial temperature distribution is shown for each stage of the compressor in figures 68 through 79.

(h) LCF Design Criteria

To accurately predict disk low cycle fatigue life, an elastic plastic computing deck was developed. This deck analyzes a disk for a complete engine cycle (from startup through transient conditions to shutdown) and then recycle (startup to shutdown, startup to shutdown, etc.) as many times as is required to make an accurate prediction of the disk low cycle fatigue life. Recycling is necessary to redistribute initially high local stresses that are not retained through the entire disk life. The low cycle fatigue life of the disk is based on these redistributed stresses, which are lower than the initial stresses.

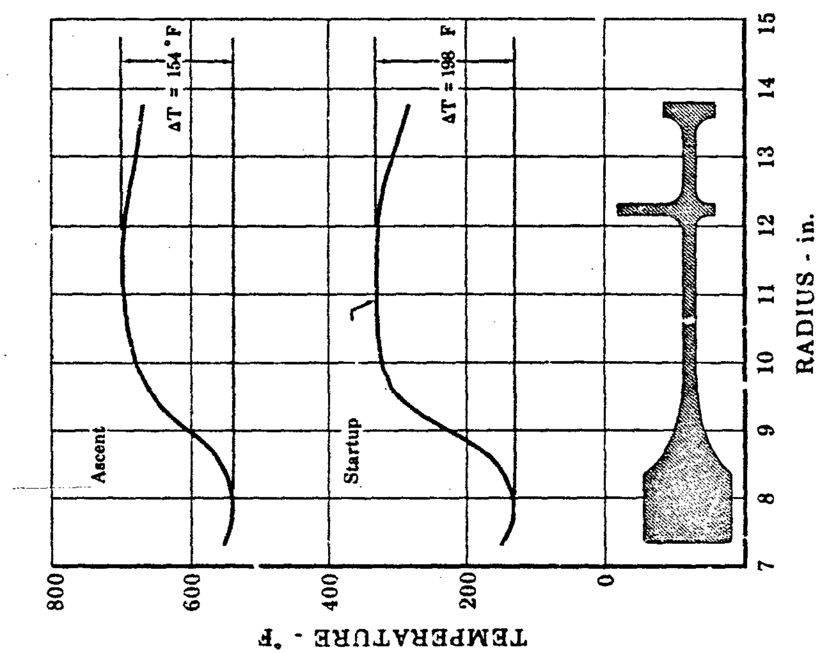

Figure 68. Compressor Disk Temperature - FD 16291
3rd-Stage IIA

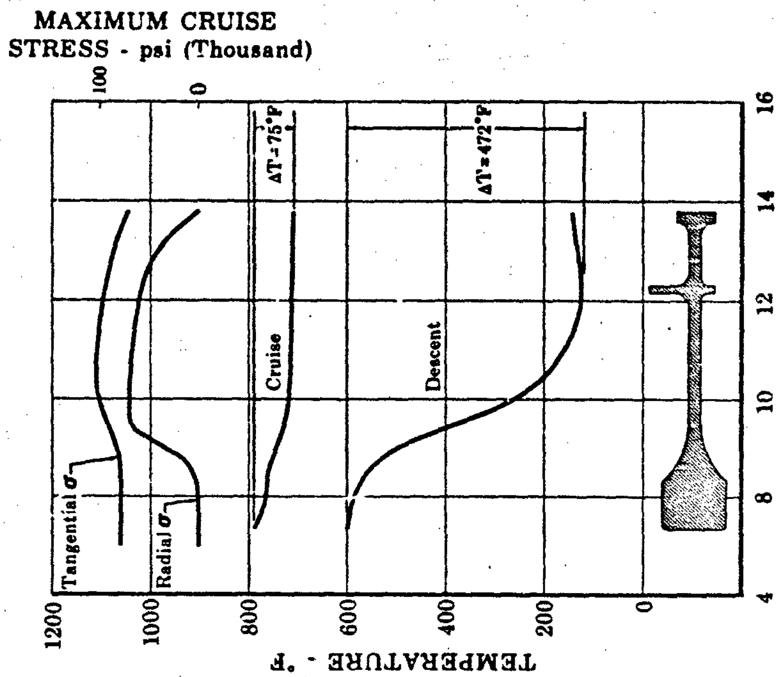

Figure 69. Compressor Disk Temperature - FD 16292
3rd-Stage IIA

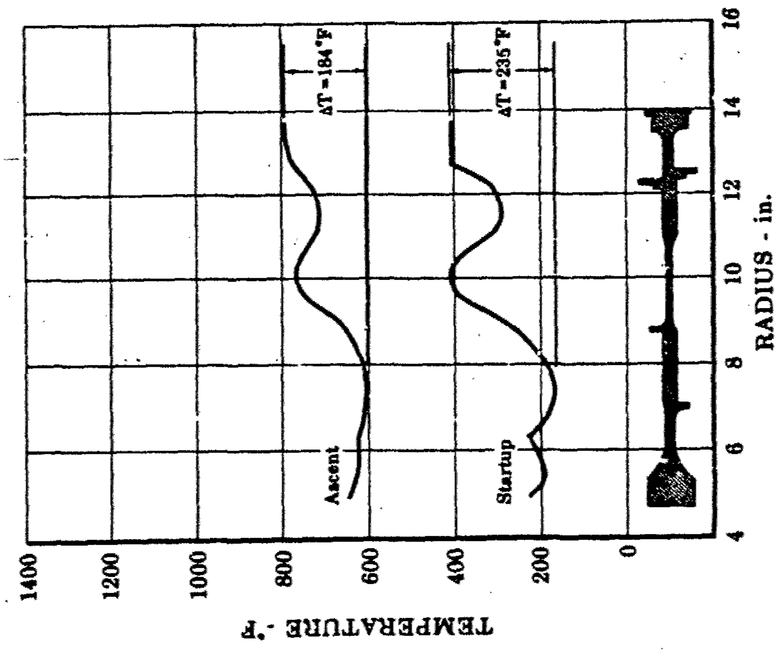

Figure 70. Compressor Disk Temperature - FD 16293
4th-Stage IIA

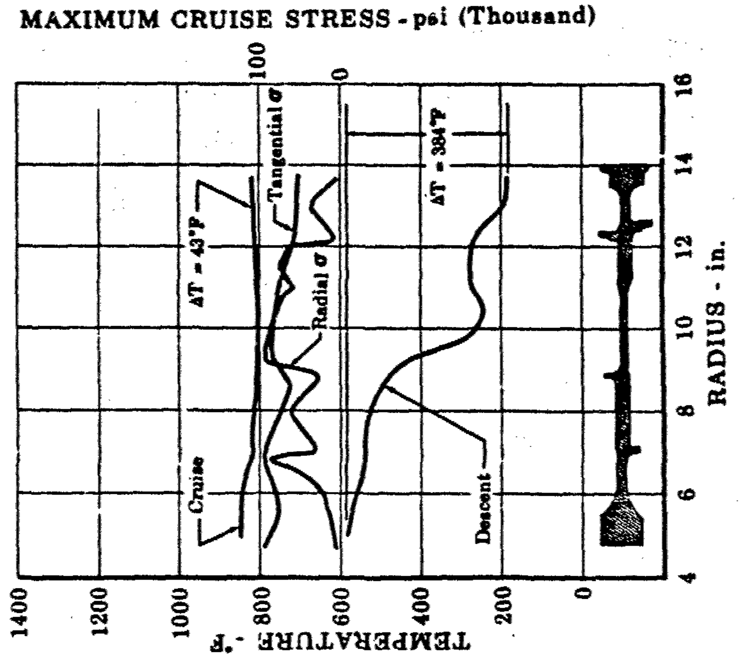

Figure 71. Compressor Disk Temperature - FD 16294
4th-Stage IIA

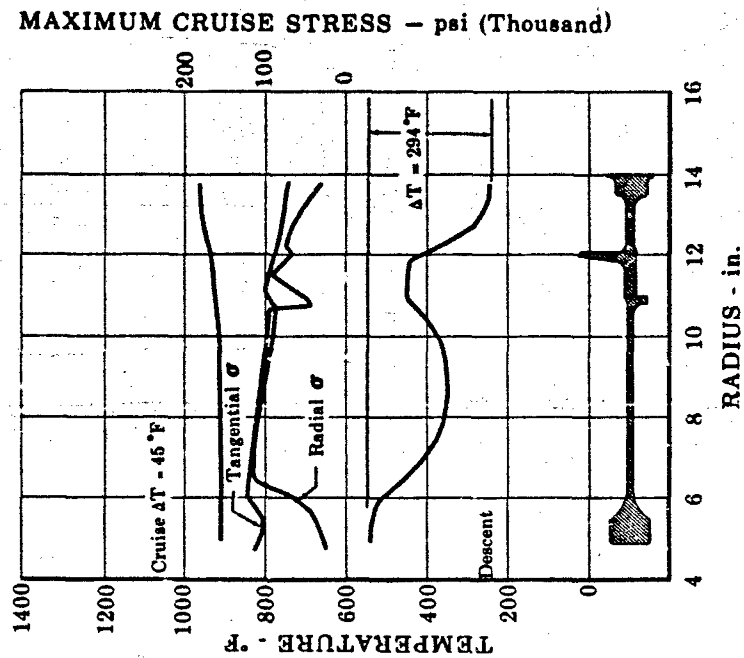

Figure 73. Compressor Disk Temperature - FD 16296 IIA
5th-Stage

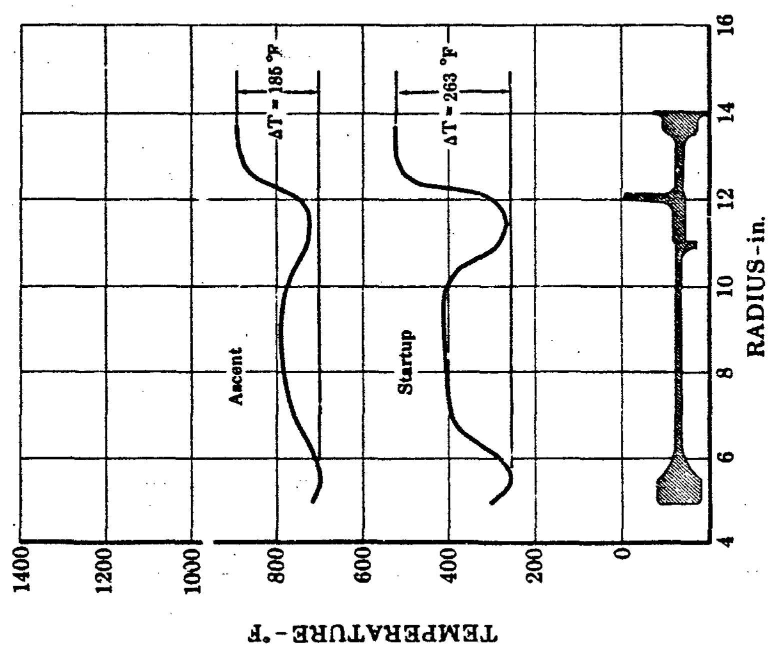

Figure 72. Compressor Disk Temperature - FD 16295 IIA
5th-Stage

CONFIDENTIAL

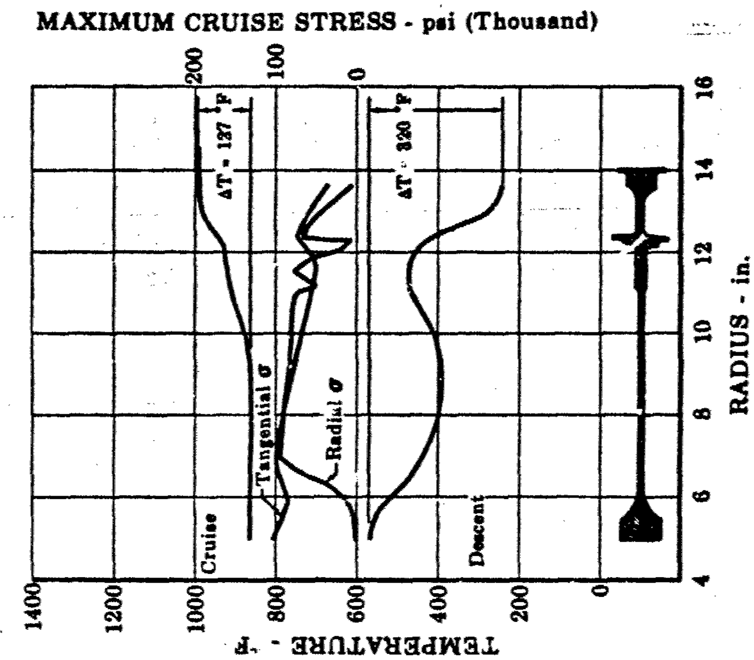

Figure 75. Compressor Disk Temperature - FD 16298
6th-Stage IIA

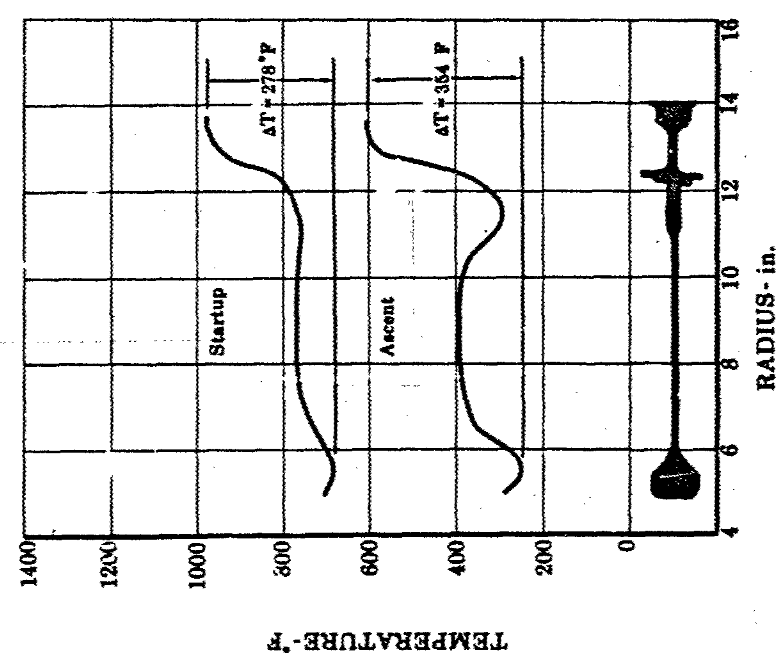

Figure 74. Compressor Disk Temperature - FD 16297
6th-Stage IIA

CONFIDENTIAL

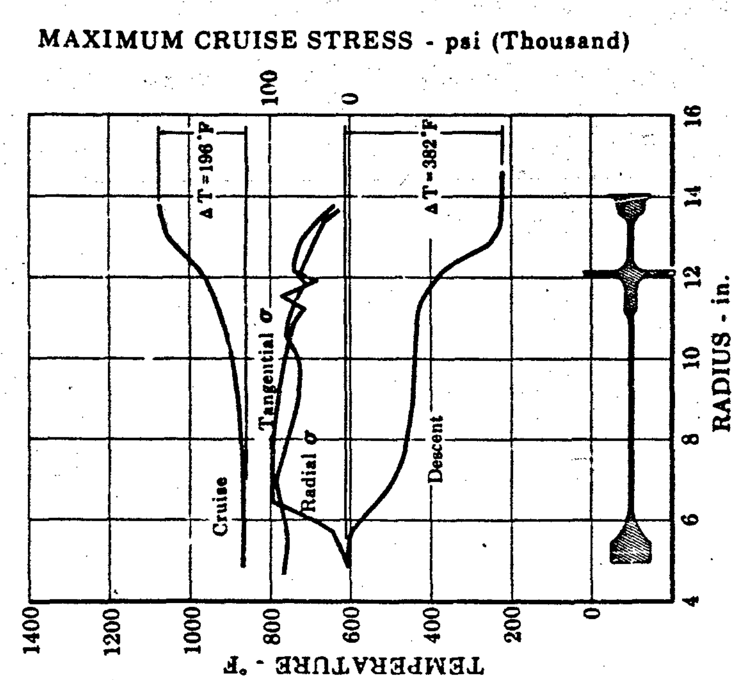

Figure 77. Compressor Disk Temperature - FD 16300 IIA
7th-Stage

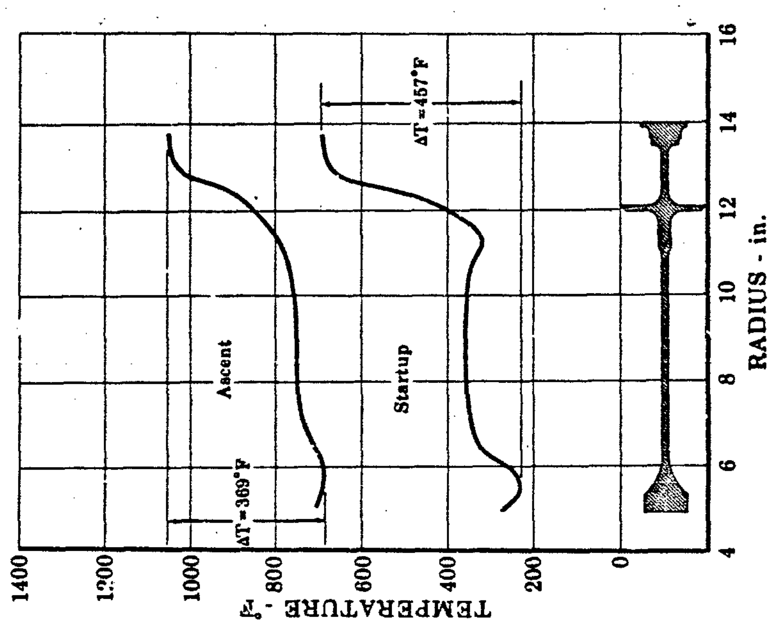

Figure 76. Compressor Disk Temperature - FD 16299 IIA
7th-Stage

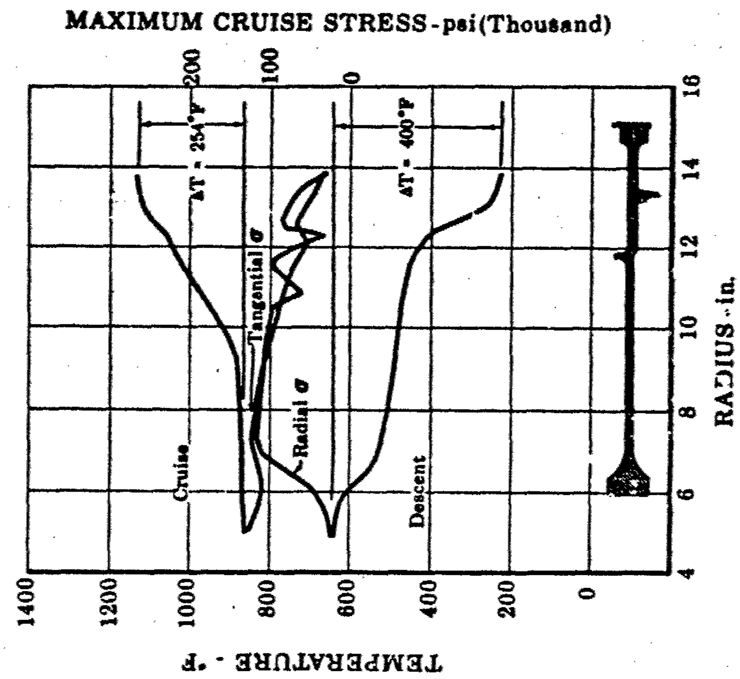

Figure 79. Compressor Disk Temperature - FD 16302
8th-Stage IIA

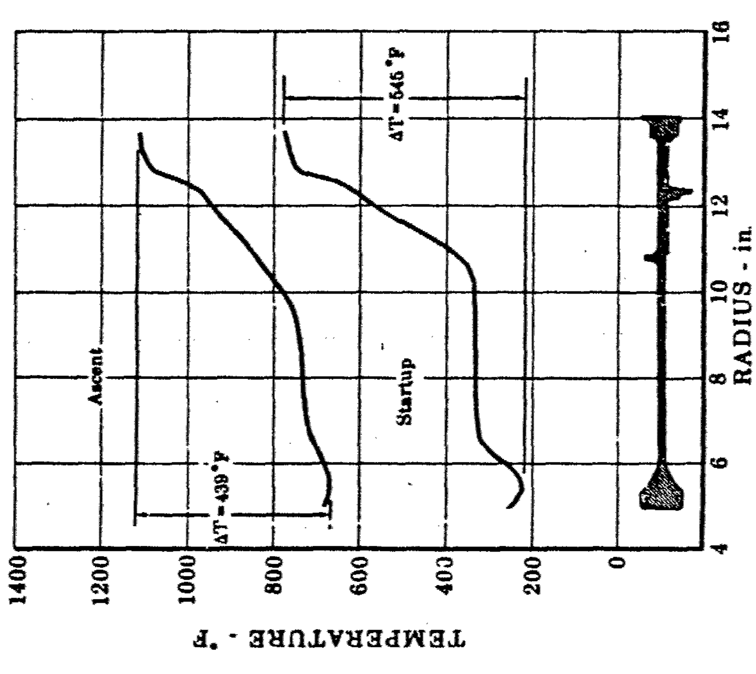

Figure 78. Compressor Disk Temperature - FD 16301
8th-Stage IIA

From a stress distribution of the recycled disk, the cyclic and steady-state stresses are determined. With these stresses the low cycle fatigue life of the disk is determined from a Goodman Diagram. A typical Goodman Diagram is shown in figures 80 through 82.

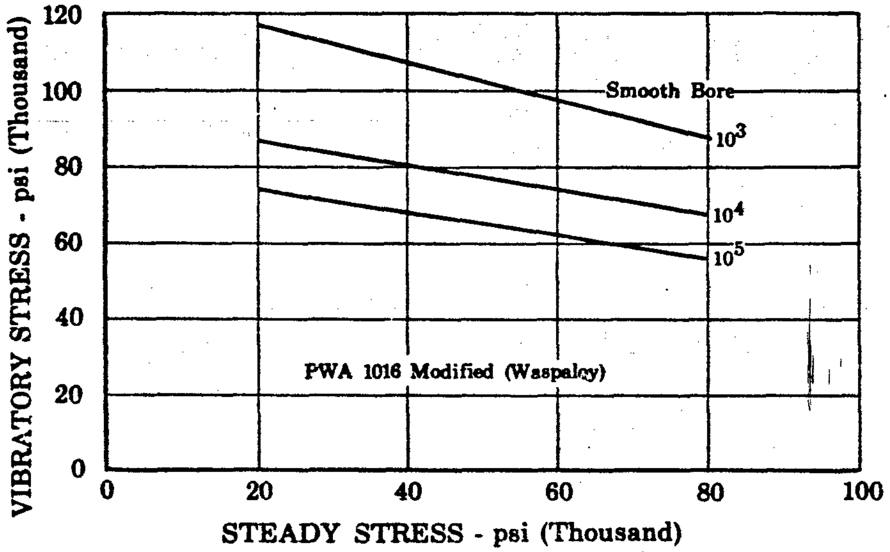

Figure 80. Low Cycle Fatigue Goodman Diagram FD 16362 IIA

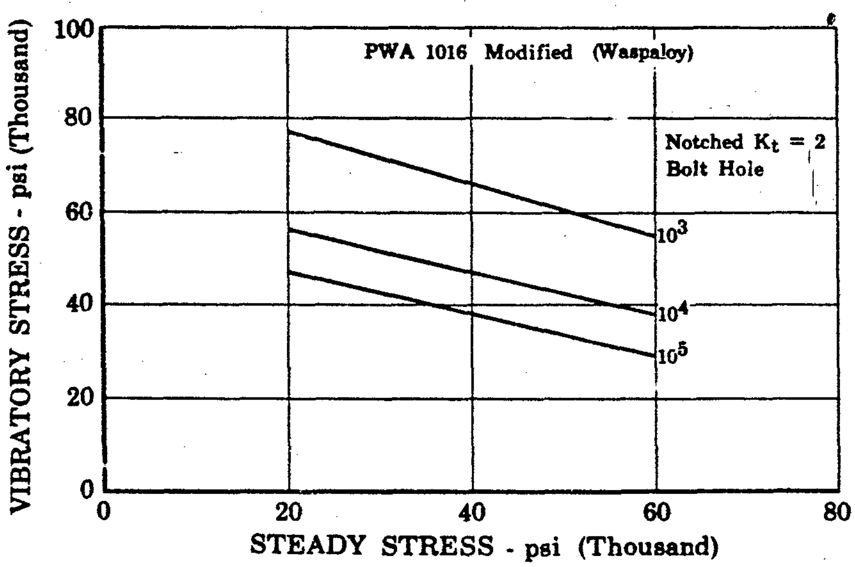

Figure 81. Low Cycle Fatigue Goodman Diagram FD 16363 IIA

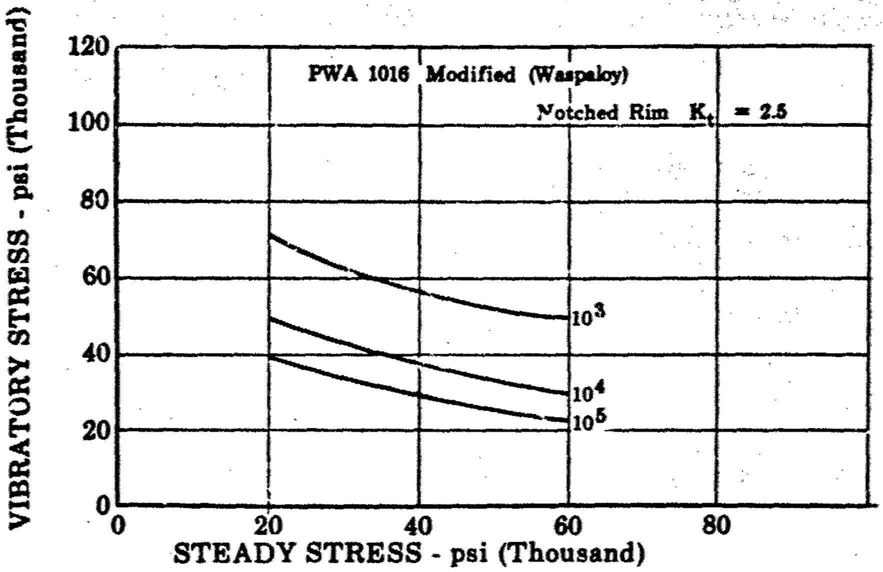

Figure 82. Low Cycle Fatigue Goodman Diagram FD 16364 IIA

Considerable testing encompassing both actual disk testing, and smooth and notched specimen testing have been made to enable an accurate prediction of low cycle fatigue life. Figure 83 shows some of the types of specimens tested, which represent configurations of disk bores, bolt holes, and rims. Figures 84 and 85 illustrate a nonrotating rig (ferris wheel) and a rotating rig (whirl pit) used for disk low cycle fatigue testing. Tables 20 and 21 represent a few of the disks that were low cycle fatigue tested and show the predicted life of the disk and the actual life obtained.

More than 300 disks and 250 specimens have been tested to establish low cycle fatigue capabilities.

In addition to in-house testing, an inspection program of commercial service used disks has been in progress. Table 22 contains results of inspecting 241 disks and gives a comparison between predicted and service life. The indications found at the life inspection cycles listed in table 22 were all less than 1/32 inch long. The results indicate that the present system is conservative on Predicted Life but that Predicted Indications need more statistical analysis.

Investigations are underway to determine the crack propagation rates by cycling disks of in-house (P&WA) engines for different disk materials.

To date some 15 disks have been cycled beyond certified service life and after initial cracking. The results of this investigation will provide information necessary to extend service life consistent with safety.

Disk life expectancies are predicted by factors having a range of values, rather than a finite value. The conservative values are always used in the prediction of disk life; as an example, the minimum material properties, minimum disk dimensions, and maximum thermal gradients are used in prediction.

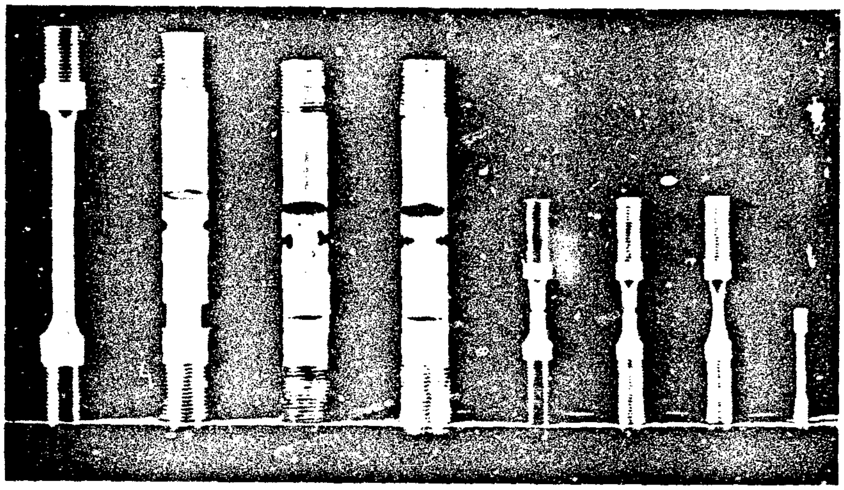

Figure 83. Special Specimen Forms Used in Laboratory Evaluation of Material Properties — H-16221-A IIA

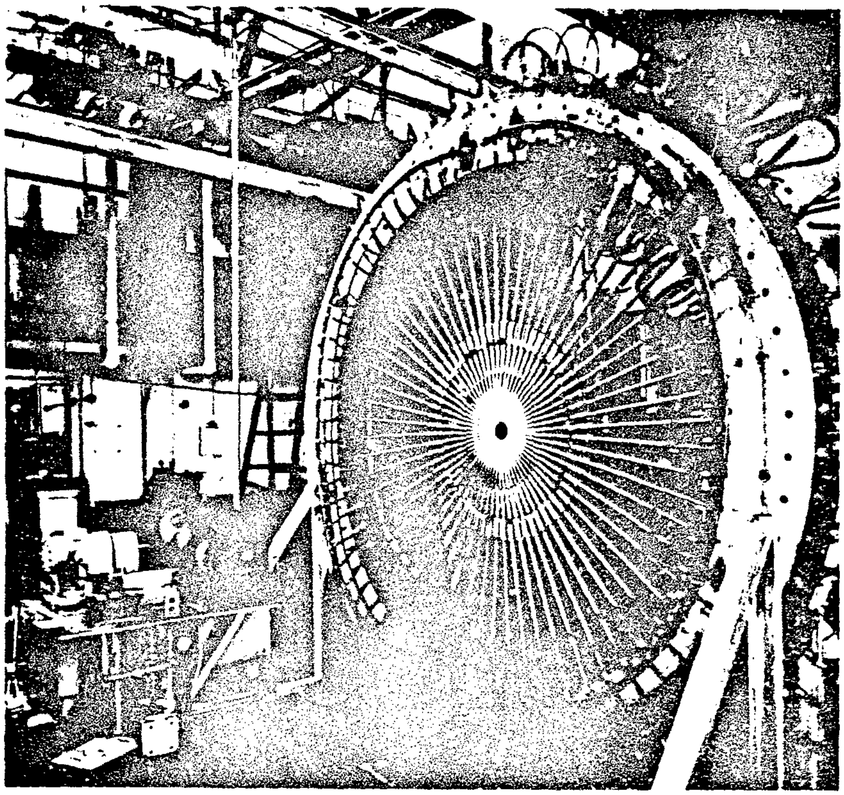

Figure 84. Non-Rotating Laboratory Test Rig for Hydraulic Loading of Rotor Disk — H-15495-A IIA

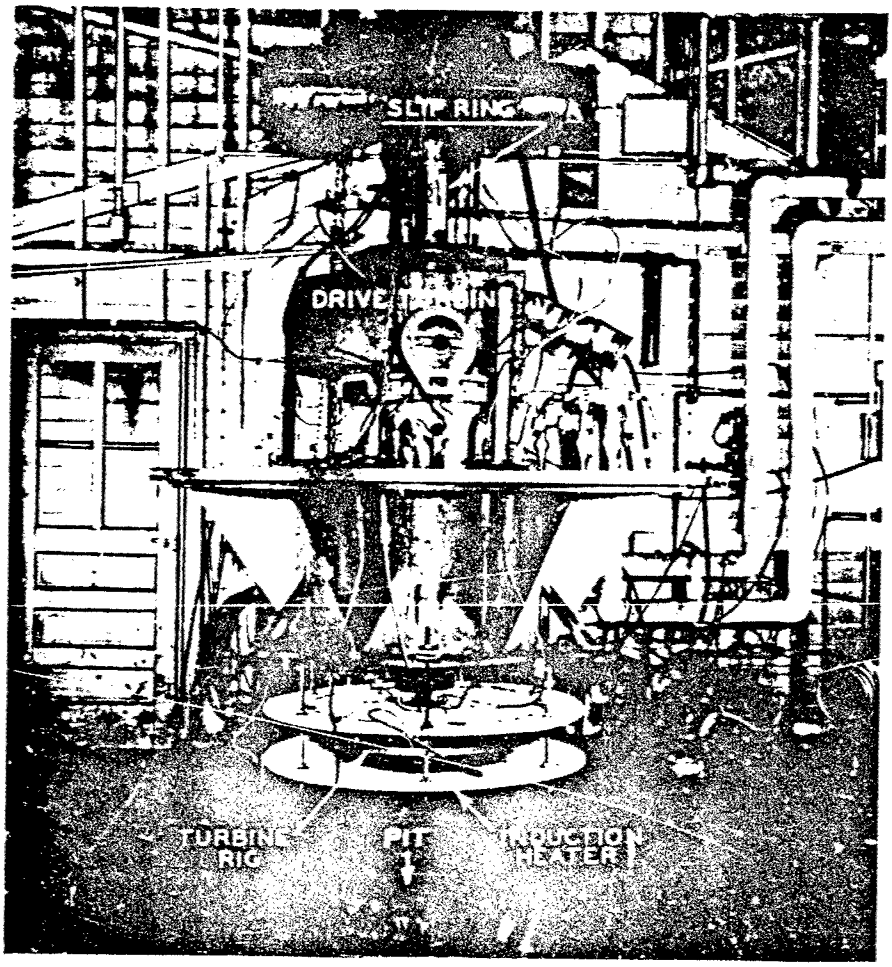

Figure 85. Typical Spin Pit Rig Used in Testing Rotor Disks   Exp-97972-A

Table 20. Low Cycle Fatigue Whirl Pit Tests of Recperated JT3D-1 1st-Stage Turbine

Test No.	Minimum Predicted LCF (Cycles	Cracks First Observed (Cycles)
1	1000	4000
2	1000	4878
3	1000	2585
4	1000	1945

Table 21. Low Cycle Fatigue Ferris Wheel Tests of Reoperated JT8D1 Compressor Fifth Stage

Test No.	Minimum Predicted LCF (Cycles)	Cycles when 1/32-inch Crack was Observed
1	4800	6800
2	4800	5200
3	4800	4300
4	4800	20,000
5	4800	31,000
6	4800	41,000
7	4800	17,000
8	4800	18,000
9	4800	28,000

Table 22. Summary of Service Disks Inspected

Engine	Stage	Part No.	Material	Disks Inspected*	Indications Found	Predicted	PWA "Blue Book" Predicted Life (Cycles)	Typical Life at Inspection (Cycles)
JT3C6	5C	199005	AMS 6415	6	0	--	14,000	4800
JT3C6	11C	310211	AMS 6304	56	0	3	9400	4900
JT3C6	16C	342216	AMS 6304	10	0	--	5000	4900
JT3C7	3C	359703	AMS 6415	25	3/1	1/2**	4000	3356
JT3C7/12	5C	374705	AMS 6415	13	2	4	7000	4250
JT3C7/12	6C	374706	AMS 6415	56	0	28	7000	4450
JT3C7	11C	358211	AMS 6304	10	0	3	10,000	5050
JT3C7	16C	358216	AMS 6304	3	0	--	3000	3200
JT3D1	14C	414114	PWA 733***	7	0	0	10,000	4650
JT3D1	15C	425615	PWA 1003	15	3	--	2000	1950
JT3D1	16C	425616	PWA 1003	8	0	--	3000	2200
JT3D1	2T	419102	AMS 5660	6	0	--	1500	1850
JT3D1	3T	418903	AMS 5660	13	13	0	5000	3700
JT4A10/12	2C	310302	AMS 4928	13	0	0	10,000	3550
Total				241				

* Includes disks that were static Zyglo inspected
** Bolt hole/rim
*** High strength AMS 6304

(i) Disk Burst

The disk burst margin is described analytically by the following equation:

$$\frac{N_b}{N_d} = \frac{BF \times Ult}{St_{ave}}$$

where:

N_b = speed at which disk should burst
N_d = design speed
BF = burst factor
Ult = ultimate strength at average temperature of disk
St_{ave} = average tangential stress of disk

The burst margin selected for the JTF17 compressor allows 1.20 overspeed relative to maximum rotor speed within the control envelope, including maximum transient conditions. This overspeed allowance is consistent with commercial engine practice.

Figure 55 shows the burst margin applied over the various engine inlet temperatures and rotor speeds.

The burst factor (BF) is the ratio of the average tangential stress of a disk at burst to the minimum ultimate strength of the disk material. An accurate means of predicting the burst speed of compressor disks was developed through a continuous program that includes evaluation of basic tensile properties (yield and ultimate strengths) of disk materials, the calculation of disk stresses, and the correlation of these stresses to the strength of the material through actual burst tests.

The evaluation of material properties is basic to the prediction of burst. Laboratory tests were conducted to determine the tensile properties of all compressor disk materials. The results of these tests, conducted for the full range of disk operating temperatures, were plotted on strength-temperature diagrams so as to provide minimum tensile strength as a function of temperature. A plot of tensile properties of the selected compressor material is shown in figure 86.

In designing all engine disks, P&WA uses high-speed digital computers, which have a stress calculation program capable of recognizing the variation of disk shape, temperature, and material properties including the actual stress-strain relationship in the plastic range. Not only does the program include thermal effects and speed effects associated with rim and body loads, but it also has provision for introducing the added loading due to bolts and spacers that the disk must support at the bolt circle region. From plastic stress analysis made at progressively higher speeds, it is possible through actual spin pit tests to determine the correlation between the calculated average stress at burst and the ultimate strength of the material. Such a correlation provided a consistent ability to predict the minimum burst speed of compressor disks within 5%.

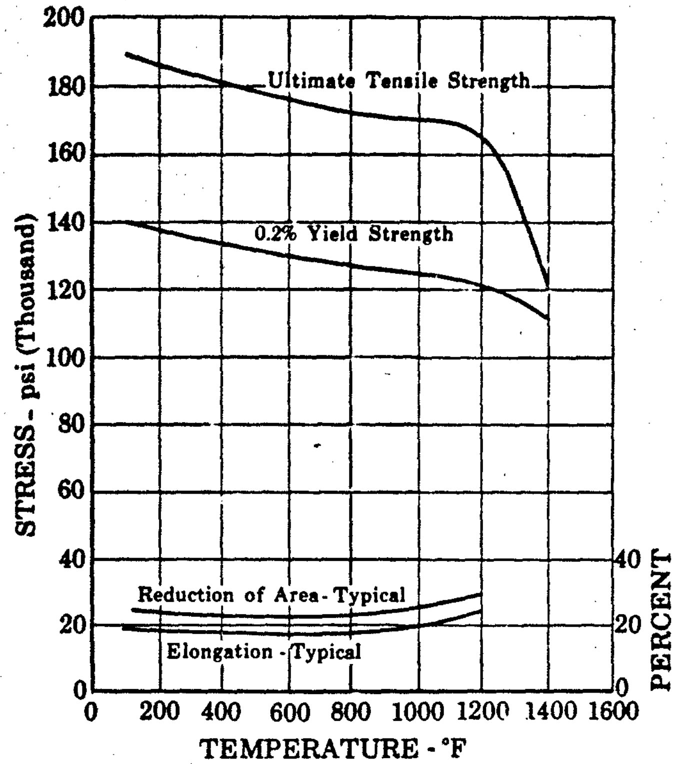

Figure 86. PWA 1016 Modified (Waspaloy) FD 16365 IIA

The burst factor of a particular disk is primarily dependent upon stress variance or on shape and material properties. The burst factor of 0.90 for compressor disks was established from spin testing. The spinning is done in spin pits that are capable of various speed and temperature conditions, including radial disk temperatures; this permits duplication of operating conditions experienced by disks during actual engine running.

Spin test results showing actual burst speed compared to predicted are shown in table 23.

(4) Stators

The compressor stator system consists of variable inlet guide vanes and six stages of fixed stator vanes to match the six rotor stages. The last or 8th-stage stator function is combined with exit stator or air straightening function so that only one slotted airfoil is used. The inlet guide vane assembly was previously described in paragraph (2) Windmill Brake - Variable Inlet Guide Vane. This paragraph will cover the fixed stators. (See figure 52.)

The compressor cases and stators are a guided cantilever type in continuous ring (nonsplit) cases. This provides a rigid, wear-free

construction. Airline service experience has shown that split cases distort and are difficult to reassemble properly after overhaul, which results in a larger blade tip clearance and a performance loss. Guided cantilever stators provide positive retention of each half of a cracked airfoil.

Table 23. Disk Burst Tests Compressor Disks

	Identification	Comments	Actual Burst Speed / Predicted Burst Speed
1.	JT4 2nd SKE-485 PWA 679	Room Temp, Burst	1.12
2.	JT3 14th 196914 PWA 722	Room Temp, Burst After Creep	.98
3.	JT3 15th SKE-24644 AMS 6302	Room Temp, Burst After Creep	1.05
4.	JT3 16th 211516 PWA 722	Room Temp, Burst After Creep	1.10
5.	JT3 15th 272915 Inco 901	Room Temp, Burst	.99
6.	JT3 14th 196914 PWA 722	Room Temp, Burst	1.08
7.	JT3 14th 196914 PWA 722	Room Temp, Burst After Creep	1.05
8.	JT3 14th 196914 (Splined Bore) PWA 722	Room Temp, Burst	1.05
9.	JT8 2nd 271622 (Split Disk) PWA 682	Room Temp, Burst	1.14
10.	JT4 2nd 310302 PWA 682	Room Temp, Burst	1.09
11.	JT3 4th 243204 (Low Hardness Disk) AMS 6415	Room Temp, Burst	1.01
12.	JT8 12th 367912 PWA 1002	690°F, Burst	.95
13.	JT3 13th 254613 AMS 4928	610°F, Burst After Eng. Run	1.14
14.	JT3 16th 241616 AMS 6304	720°F, Burst After Eng. Run	1.11
15.	J58 6th 2040906 PWA 1007	Room Temp, Burst	1.00
16.	J58 6th 2038206 PWA 1007	Room Temp, Did Not Burst	1.00

(a) Interstage Air Seals

The interstage ID knife edge air seal lands are the machined circumferential serration type. (See figure 87.) They are riveted to the vane feet stiffening rings and are replaceable.

As an alternative or backup design with substantiation (J58) honeycomb can be brazed to the seal lands in place of the machined serrations. (See figure 87.) However, this would involve an increase in cost. The seal land construction permits air loads to help stabilize the seals in the high frequency vibration modes. All stages of compressor stators and the exit guide vanes are attached to stiffening rings on the ID. The 3rd- and 4th-stage titanium stators are mechanically attached to the outer cases. The 5th-, 6th-, and 7th-stage stators are brazed into box-type cases on the OD. This provides a rigid lightweight construction. On the exit guide vane, the ID is riveted to the diffuser case flange and brazed in a floating shroud on the OD.

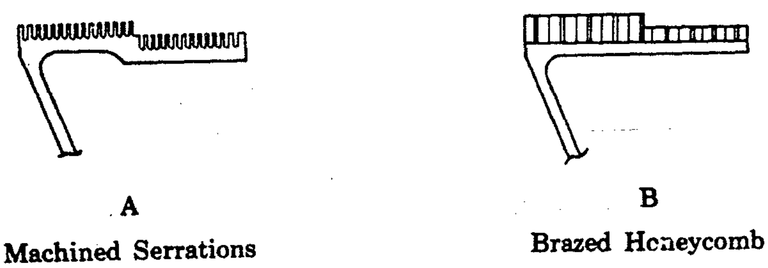

Figure 87. Compressor ID Air Seals

FD 16390
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(b) Blade Containment

The blade containment design philosophy for the compressor is similar to that of the fan. The difference is mainly in having more walls, the compressor case walls plus two duct walls, to prevent penetration of the nacelle area. One concern in any basic containment philosophy is whether or not contained blades will move downstream setting off a chain reaction that culminates in such extensive blade loss unbalance that the rotor integrity tiebolt criteria would be exceeded. No practical flightweight design would contain such a failure.

A ducted fan, therefore, has an advantage over engines having a single outer case wall in this respect; the compressor case may be permitted to have blade penetration, removing or reducing downstream blade loss, yet the blade will still be contained within the engine, thereby avoiding the above catastrophic failure.

(c) Borescope Provisions

Inspection of compressor blades is made through borescope bosses between stators on each stage. (See figure 88.) This inspection can be made from outside an installed engine by removing small caps on the

outer wall of the duct diffuser. Appropriate tools remove an air sealing plug on the compressor cases.

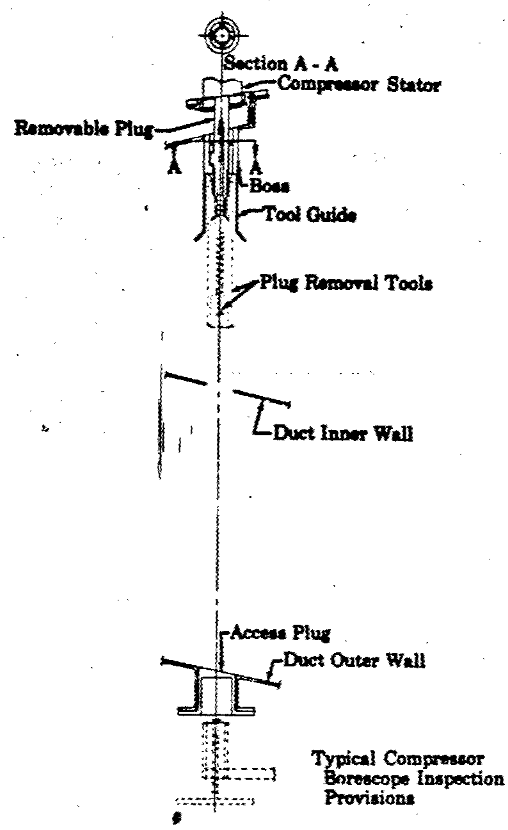


Figure 88. Compressor Borescope Inspection Provisions FD 16420 IIA

The plug and tools are made so that the wrench cannot engage the locknut until the attaching rod is sufficiently engaged. This will ensure that the plug will not become unattached and lost in the engine. During assembly, the nut will not engage plug threads until the lugs on the plug are in the boss slots. This prevents the possibility of the plug protruding into the air stream. Also, the size and angular position of the lugs and slots prevent the plugs from being assembled in the wrong boss, thus assuring a flush alignment in the airstream.

(d) Blade Tip Seal Shrouds

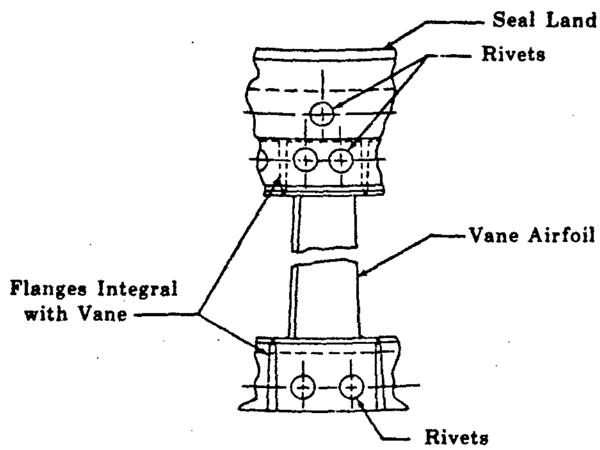
Semiabradable, honeycomb, blade-tip air seals are used in all six stages of the compressor. These seals permit relatively close clearances during normal operation because rubbing associated with abnormal transient conditions generates only slight heat. Seals are relatively inexpensive compared to blades and are completely replaceable.

The J58 engine, running with this type of honeycomb seal, has demonstrated a definite improvement in surge capability over a solid or smooth seal when installed in high pressure stages. Honeycomb durability is substantiated by extensive J58 engine operation.

In the interest of reliability and long time durability, the approach taken initially has been to make the shrouds in one piece similar to those used successfully in stages 3 and 4 of the J58 compressor. In view of the general tendency of routine wear and foreign object damage to be localized, it may prove to be expedient economically and logistically to make the shrouds segmented similar to those in stages 1 and 2 of the J58 compressor. There is sufficient versatility of the cases and stators to accept such a segmented design with minimum change.

(e) Adequacy of Structural Design

Because the front case is buckling-limited, the density-modulus ratio gave titanium the weight advantage. The 3rd- and 4th-stage stators are aerodynamically limited; therefore, the low density of titanium was an advantage. The selected temperature limits for titanium of 900°F, for continuous operation, and 950°F, for short time operation, prevent the use of titanium beyond these stages. The 3rd- and 4th-stage titanium stators are mechanically attached on both ends. (See figure 89.)



Mechanically Attached Titanium Stator

Figure 89. Mechanically Attached Titanium Stator FD 16371 IIA

(f) Design Criteria

The case and stators of the six-stage compressor are structurally similar to those used on the JT3D and JT8D engines. The design criteria concerning allowable stresses, blade-to-vane spacing, blade containment, and operating clearances are obtained from experience with these and other P&WA commercial and military engines.

Stress Criteria for Vane Airfoils

1. 85% yield for brazed stators at maximum pressure or Mach number points of the operating envelope
2. 90% yield for mechanically attached stators at the above points
3. Allowable fatigue of 2 times fatigue strength at the above points. (See figure 90.)
4. 4250 hours stress rupture at standard day cruise.

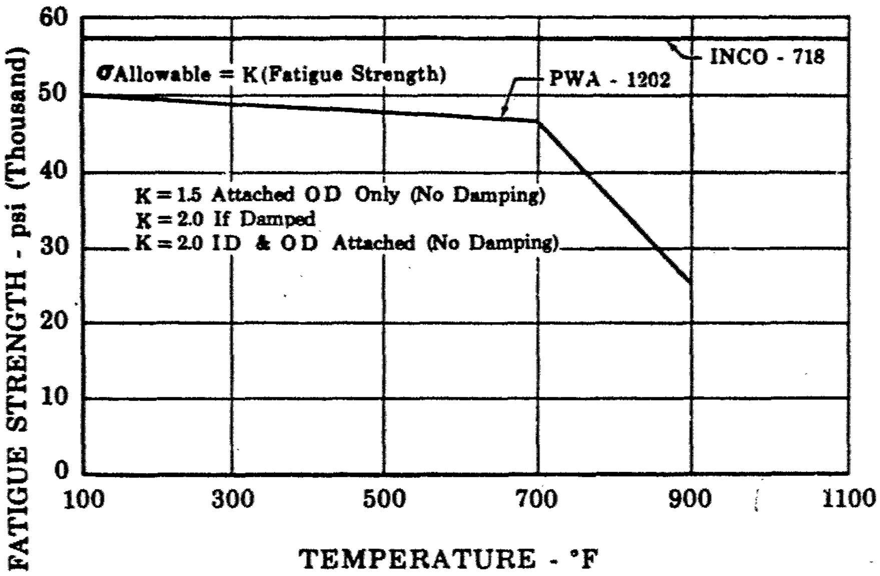


Figure 90. Allowable Bending Stress for Vanes FD 16351 IIA

Stress Criteria for Structural Cases

Creep limits: combined thermal and operating loads are as follows.

1. Total permanent growth limits of 0.5% for Ni base alloy cases and 0.1% for titanium.
2. 1.3 creep buckling stress margin.

Yield limits: combined thermal, operating, and maneuver loads at maximum pressure or Mach number points of the operating envelope are as follows.

1. Hoop limited: 90% of 0.2% yield strength or 67% ultimate strength for turned cases and 85% of 0.2% yield strength or 60% of ultimate strength for axially fusion or flash butt welded cases.
2. Bending limited: 100% of 0.2% yield strength or 75% ultimate strength.
3. Buckling limited: 1.3 stress margin for pressure and moment buckling.

(5) Interstage Compressor Bleed

A bleed is required in the compressor, in addition to the cabin air bleed, to unload the engine for starting and to match the compressor at some idle conditions. The bleed was used in preference to variable stator geometry. This eliminates the inefficient wall conditions imposed by variable stator end gaps. The most effective type bleed is an interstage bleed so that the blade gas loading on the last stages will be relieved. It is sized to prevent stalling the front stages at idle, therefore precluding vibrational problems and still maintaining a wide range of efficient operation.

A bleed area of 50 in.^{2} has been provided by using eight poppet-type valves with an area of 6.25 in.^{2} each. These valves are operated automatically through signals provided by the engine controls. The valves operate by mechanical spring and differential air pressures acting on an internal piston and the sealing poppet.

To obtain a uniform bleed completely around the high pressure compressor, an open annulus was included in the design. Gas generator structural loads through this area are carried in the manifold case. (See figure 91.)

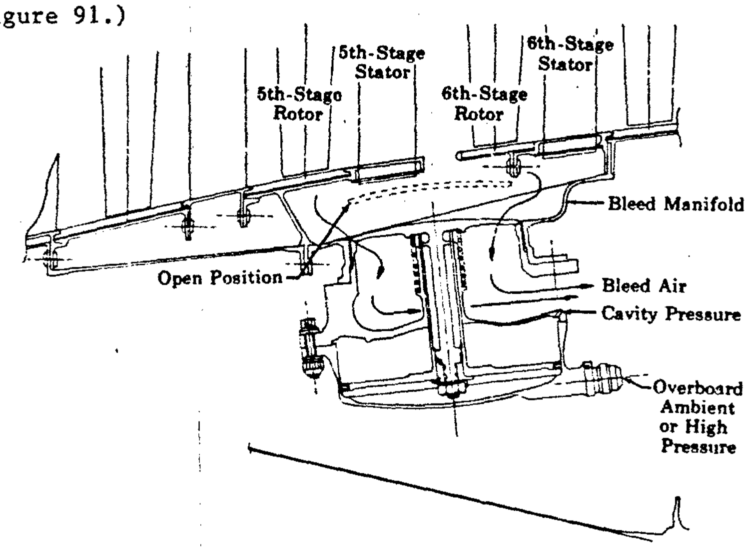


Figure 91. Compressor Interstage Bleed Valve FD 16419 IIA

The annular bleed was chosen for several reasons, primarily:

1. For the required area, the annulus results in the shortest axial length.
2. An annulus uniformly around the compressor results in a uniform bleed and does not induce unsymmetrical surges, blade excitation, or burner hot spots.

As shown in figure 91, the bleed air is discharged into the cavity between the compressor and the fan duct. The required bleed areas for starting are shown in figure 48 as a function of inlet guide vane setting and location of discharge. Paragraph (2), Windmill Brake - Variable Inlet Guide Vane, explains the selection of bleed area based on starting the engine with the inlet guide vane in the cruise position. The bleed area required could be reduced to 40 in.^{2} if it were ducted overboard to ambient pressure. The additional weight (14 pounds) and complexity in ducting the bleed air overboard was not warranted; therefore the larger bleed, which discharges into the fan cavity, was selected.

The poppet-type bleed valves used are similar to valves used on P&WA commercial JT8D and Military JT8 engines.

The valve is actuated by air pressure supplied by the engine. The valve is spring-loaded open for starting, with the dome connected to compressor discharge air. At a preset speed, the dome will be vented to ambient and the pressure differential across the piston will overcome the spring and the valve will snap closed. The pressure differential across the poppet and the piston will then keep the valve closed. The valve can be open, if desired, by supplying high pressure air to the dome for cruise or idle descent conditions.

The bleed valves are oriented between the fan duct struts and installed with three bolts each. The tubing connection and mounting bolts are accessible through the access ports in the duct diffuser to facilitate maintenance.

The sealing disk of the bleed valve will be exposed to 5th-stage air throughout the flight envelope. The valve, therefore, will have a maximum of 1000°F on the seal and fan discharge air, approximately 700°F maximum on the main body. Materials for the valve parts will be the same as for the valves used in the initial experimental engine. The major items are valve body, AMS 5382; piston, PWA 1010; and valve disk, PWA 1010. Rubbing surfaces on the piston are hardcoated with tungsten carbide and both surfaces in contact are lubricated with antigalling compound, PWA 586.

Valves made for the initial experimental engine were satisfactorily tested through 5000 cycles at 1000°F.

(6) Materials Summary

Tables 24 and 25 list material choices for the major components of the compressor, a representative design metal temperature, and the primary design criteria.

Table 24. Compressor Stator Material

Part Name	Material	Typical Standard Day Cruise Temp (°F)		Limiting Mode
3rd-stage vane	PWA 1202 (8Al-1Mo-1V)	695	776*	Aerodynamics
4th-stage vane	PWA 1202	776	863*	Aerodynamics
5th-stage vane	PWA 1009 (Inco 718)	863	948*	Aerodynamics
6th-stage vane	PWA 1009	948	1030*	Aerodynamics
7th-stage vane	PWA 1009	1030	1094*	Aerodynamics
Exit guide vane	PWA 1009	1094	115[illegible]*	Aerodynamics
Case front	AMS 4966 AMS 4910 (A110)	695		Buckling and blade contamination
Case, bleed manifold	PWA 1033 and PWA 1009 (Inco 718)	750		Yield
4th-stage case	AMS 4966	775		Manufacturing
5th-stage case	PWA 1033 and PWA 1009	860		Manufacturing
6th-stage case	PWA 1033 and PWA 1009	940		Manufacturing
7th-stage case	PWA 1033 and PWA 1009	1020		Yield
Shroud, exit guide vane	PWA 1009	1085		Manufacturing
3rd-stage blade tip seal	AMS 5540 H.C.(Inconel) AMS 5508 Ring(Greek Ascoloy)	695		
4th-stage blade tip seal	AMS 5508 Ring AMS 5540 H.C.	780		
5th-stage blade tip seal	AMS 5540	865		
6th-stage blade tip seal	AMS 5540 H.C. AMS 5665 Ring	950		
7th-stage blade tip seal	AMS 5540	1030		
8th-stage blade tip seal	AMS 5540	1095		
3rd-stage KE air seal land	AMS 5616 (Greek Ascoloy)	775		Wear depth on land Yield on support
4th-stage KE air seal land	AMS 5616	863		Wear depth on land Yield on support
5th-stage KE air seal land	AMS 5754 (Hastelloy X)	948		Wear depth on land Yield on support
6th-stage KE air seal land	AMS 5754	1030		Wear depth on land Yield on support
7th-stage KE air seal land	AMS 5754	1094		Wear depth on land Yield on support
8th-stage KE air seal land	AMS 5754	1154		Wear depth on land Yield on support

*Root temperature

Table 25. Compressor Rotor Material

Part Name	Material	Typical Standard Day Cruise Temp (°F)		Limiting Mode
Front hub, high rotor	PWA 1003 (Inco 901)	865**		Critical speed
3rd-stage disk	PWA 1016 Modified (Waspaloy)	750**		LCF
4th-stage disk	PWA 1016 Modified	825**		LCF
5th-stage disk	PWA 1016 Modified	875**		LCF
6th-stage disk	PWA 1016 Modified	925**		LCF
7th-stage disk	PWA 1016 Modified	975**		LCF
8th-stage disk	PWA 1016 Modified	1000**		LCF
Z spacer	PWA 1016 Modified	850**		Critical speed
Rear hub	PWA 1016 Modified	1060**		Critical speed
3rd-stage blade	PWA 1202	695	715*	Vibration
4th-stage blade	PWA 1202	776	820*	Z Ratio
5th-stage blade	PWA 1202	863	900*	Z Ratio
6th-stage blade	PWA 1016 Modified	948	990*	Z Ratio
7th-stage blade	PWA 1016 Modified	1030	1060*	B/2
8th-stage blade	PWA 1016 Modified	1094	1125*	R ratio
Rotating seal land,	PWA 1016 Modified	870		Bending
front hub	PWA 1016 Modified	870		Bending
KE seal, front hub	PWA 1016 Modified	870		Bending
4th-stage front KE seal	PWA 1016 Modified	775		Bending
4th-stage rear KE seal	PWA 1016 Modified	865		Bending
6th-stage front KE seal	PWA 1016 Modified	950		Bending
6th-stage rear KE seal	PWA 1016 Modified	1030		Bending
8th-stage front KE seal	PWA 1016 Modified	1095		Bending
8th-stage rear KE seal	PWA 1016 Modified	1155		Bending
Long tie rods	PWA 1016 Modified	925**		Bending and tension
4th-stage tie rods (short)	PWA 1016 Modified	810		Tension
Anti-vortex tubes	PWA 1010 (Inco 718)	860		Buckling
Blade locks, wire	AMS 5687 (Inconel)	1125 max		Shear
Outer bore tube	PWA 1010	870		Bending
Inner bore tube	PWA 1010	870		Bending
Front bore tube support	PWA 1016	870		Bending

* Root temperature
** Average temperature

Titanium is used to the highest temperature that can be substantiated by past experience on the J58, TF30, JT4, JT8D, J52 and TF33 engines. One criterion for comparing the JTF17 with this experience is the "combined stress" of the blade attachment dovetail. Figure 92 shows this stress plotted with metal temperature together with the material yield strength and stress rupture. Table 26 shows the length of experience with engines using the titanium alloys selected for JTF17. A material development program now in progress will be vigorously pursued to utilize titanium to its fullest advantage. Its principal advantage next to strength density comes as a result of characteristics of low modulus of elasticity and low coefficient of expansion. These are properties that minimize thermal gradient stresses and improve the LCF potential of a part. New alloys, such as PWA 1205 (IMI 679) and PWA 1209 (6Al-2Sn-4Zr-2Mo), which appear to have higher temperature capabilities than currently engine-qualified PWA 1202 and AMS 4923, are being evaluated.

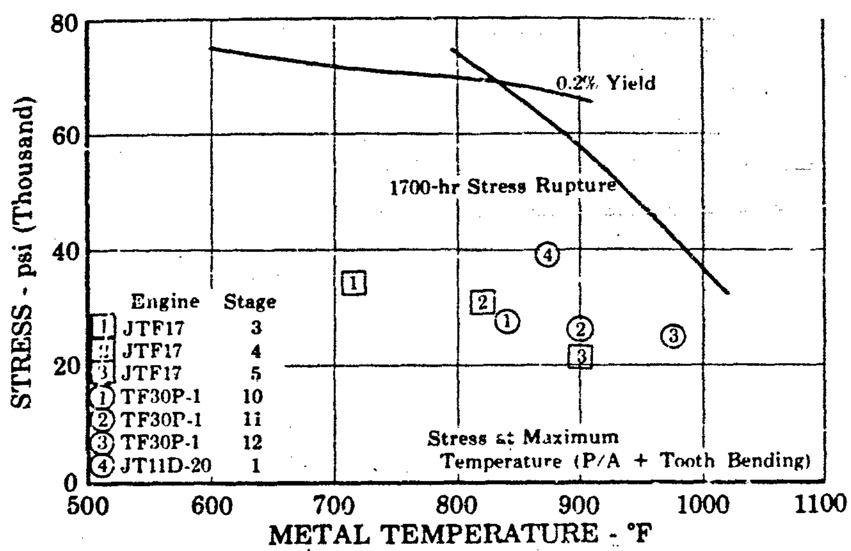


Figure 92. Dovetail Combined Stress for PWA 1202 Blades

FD 16349
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Table 26. P&WA High Temperature Experience With Titanium

Material	Engine	Operating Time
PWA 1202	J58	1st-stage compressor blade
	TF30	Compressor blades and disks
AMS 4910, 4926	J57	Front compressor cases and intermediate cases - 37,093,238 hours
	JT3D	Front compressor cases and intermediate cases - 14,985,758 hours
	TF30	Fan, compressor, and intermediate cases

To provide a durable wear surface, particularly for the titanium blade damper shrouds, a material development program to improve hard-facing is in progress. The design is also being evaluated for the purpose of reducing the area loading on these blade shroud bearing surfaces for improved wear.

Where temperatures become too high for titanium compressor blades, PWA 1016 (Waspaloy), modified by heat treatment to obtain properties shown in figure 86 is used. A cost comparison with PWA 1003 (Inco 901) blades resulted in such little difference that the PWA 1016 (Waspaloy modified) was selected on its merit of high toughness and high LCF properties. The J58 disks, both flat and integral spacer types, with forging shapes more difficult than JTF17 shapes, were made from PWA 1016 and exhibited very satisfactory engine operating experience. This was selected over AMS 5735 (Tinidur) for its superior LCF properties and toughness.

PWA 1010 (Inconel 718) is used for the high temperature vanes and cases, chosen for its high strength, low cost and ease of welding and repair welding (relative to Waspaloy). This selection is substantiated based on the J58 engine diffuser ease of this material.

(7) Summary of Product Assurance Considerations

(a) Maintainability

1. The compressor variable inlet guide vane (IGV) assembly can be assembled to the intermediate case as a subassembly module.
2. The IGV torque tube is assembled quickly by a simple spline engagement.
3. The IGV and lever are preassembled as a subassembly, reducing final build time.
4. Number of parts and snap diameters of the rotor are minimized to facilitate rotor balance.
5. Z-spacer and anti-vortex tubes are preassembled and balanced, reducing final build time.
6. Knife edge seals are separate from rotor spacer arms for economical replacement.
7. The rotor bore cooling tube is bolted to the rear hub to prevent wear. The front is supported by a wear bushing in the front hub.
8. Long pilot engagements are incorporated in all rotating parts to ensure positive seating during assembly and reliable balanced rotor.
9. Borescope ports are provided for each stage for inspection without excessive disassembly.
10. Rear compressor cases are made of PWA 1009 material which is easily repair welded.
11. Mechanical joints of titanium stators permit easy replacement.
12. The total number of final assembly bolted flanges is minimized.

b) Reliability

1 Wire blade locks that require shearing four wire cross sections before a blade can become loose. Conventional sheet metal tablocks have unsatisfactory service experience. Interstage bleed concept reduces number of moving parts by eliminating need for additional variable stator stages. Rotor unbalance problems are minimized by integral disk spacers, long snap engagement, etc.
2. Guided cantilever stator and IGV have redundancy of an additional support in event of a cracked vane.

(c) Safety

1. Adequate overspeed disk burst margin
2. Containment of blade failures
3. Rotor tie bolts capable of withstanding bending moments created by the loss of 10% of the airfoils of any one stage while continuing torque transmission to keep the turbine loaded, preventing overspeed
4. Windmill brake for reducing rotor speed in emergency conditions.

(d) Value Engineering

1. Honeycomb replaced with machine serrated seal lands on stages three through eight
2. KE seal in front of third disk removed
3. Redesigned cooling air boretube from two machined forgings to a weldment
4. Front hub constructed of PWA 1003 (Inco 901) instead of PWA 1016 (Waspaloy)
5. Simplified torque train of the IGV brake.

(e) Human Engineering

1. Means are provided on disks and blades to ensure correct assembly
2. Rotor balance weights are interchangeable to eliminate misassembly
3. Rotor tie bolts capable of being stretched hydraulically
4. Stator ID seal lands have offset rivet and hole combinations to ensure correct assembly.

B. PRIMARY COMBUSTOR

1. General Description

The primary combustor designed and developed to meet the objectives and requirements of the JTF17 engine is the annular ram-induction burner. The ram-induction burner concept is based on efficient use of the velocity head of the air to create a turbulent combustion zone. Use of this burner concept results in low total pressure loss and a short diffuser.

2. Objectives and Requirements

The design objectives for the primary combustor are consistent with those for the engine, as stated in Section I. Specific objectives for the design of the primary combustor are:

1. To furnish gases to the turbine at the desired temperature and temperature profile
2. To provide the required heat release of the fuel in a minimum combustor volume
3. To obtain a maximum turbine inlet pressure by minimizing combustor pressure losses
4. To provide a low cost, lightweight combustor
5. To achieve high reliability and maximum maintainability
6. To generate little or no smoke and carbon for extended turbine life and to minimize community air pollution.
7. To provide for extended service life.

The mechanical design requirements for the primary combustor are to attain a usable life consistent with 50,000 hours of airframe life and with 10,000 hours of airplane use without major repair. Additional mechanical requirements which provide safety features and ease of maintainability are:

1. Provision for automatically clearing the combustion areas of combustible fluids after a false start or engine shutdown.
2. Removal of igniters and fuel nozzles without a major disassembly of the duct heater or primary combustor
3. Removal of the annular combustor without removal of the turbine with the engine supported only by the integral mount system.

Design requirements relating to the performance of the primary combustor are shown in table 1.

Table 1. Primary Combustor Design Requirements

	SLTO	Cruise
Combustor Average Exit Temperature, °F	2300	2200
Combustor Maximum Exit Temperature, °F	2530	2375
Burner Temperature Rise (ΔT)	1598	1149
Combustion Efficiency, %	99.0	99.0
Total Combustor Pressure Loss, $\Delta P/P_t$, %	5.7	7.0

3. Design Approach

The primary combustion section of the JTF17 engine includes that part of the gas generator between the compressor exit and turbine inlet. Figure 1 illustrates the combustor configuration and the following component parts:

1. Diffuser
2. Dome and swirlers
3. Igniters
4. Fuel nozzle and support housings
5. Annular combustion chamber
6. Transition duct
7. Inner and outer rear cases.

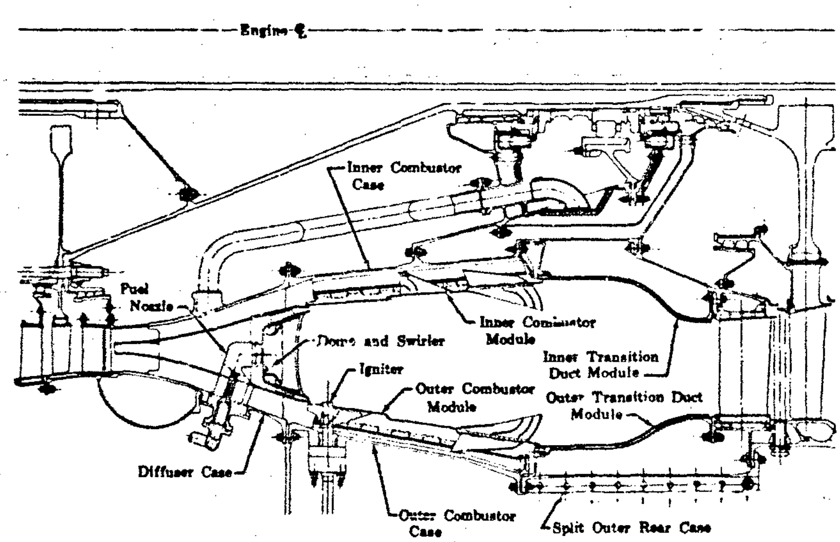

Figure 1. Primary Combustor Cross Section

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The airflow path between the compressor and the annular combustion chamber is formed by the inner and outer walls of the diffuser. Air enters the combustion chamber through dome-mounted swirler assemblies and ram air scoops in the combustion walls. The swirler assemblies provide a flow stabilization region for easy ignition. Fuel is uniformly supplied to this region through a series of nozzles located in the forward section of the combustion chamber. Ignition of the fuel-air mixture is achieved by electric spark igniters. The hot gases resulting from the combustion process are directed to the turbine inlet by an annular transition duct contained within the outer and inner walls of the rear case assembly. The rear cases form a flow path for turbine cooling air that has bypassed the combustion chamber.

Introduction of air into the burner is accomplished by a ram-induction process instead of the more conventional full diffusion process, which depends mostly upon a static pressure drop. Combustion testing has demonstrated that approximately 95% of the velocity head in a ram-induction combustor produces turbulence as contrasted with only 60% in the more conventional static pressure drop burners.

As requested in the RFP, combustor cooling airflows and pressures as well as metal temperatures are shown in figure 2. These parameters are described for engine operation at sea level takeoff, acceleration ($M = 1.5$ at 45,000 feet) and supersonic cruise ($M = 2.7$ at 65,000 feet). Predicted combustor exit temperature radial profiles for these operating modes are shown in figure 3.

In addition to desirable combustion performance features, several mechanical advantages are obtained through the use of the ram-induction burner concept. These advantages, broadly stated, are:

1. Short combustor length
2. Low combustor weight
3. Efficient combustor component cooling methods
4. Low combustor component static pressure loads
5. Ease of removal and replacement of major components.

4. Detailed Description

A detailed description of the major components and the considerations that dictated the design approach in response to the objectives and requirements of the primary combustor follows.

a. Diffuser

The diffuser consists of an inner and outer wall and a splitter. The walls form the airflow path from the exit guide vane of the compressor to the inlet of the combustion chamber. The splitter extends the full length of the diffuser and provides two independent annular passages for control of flow within the diffuser.

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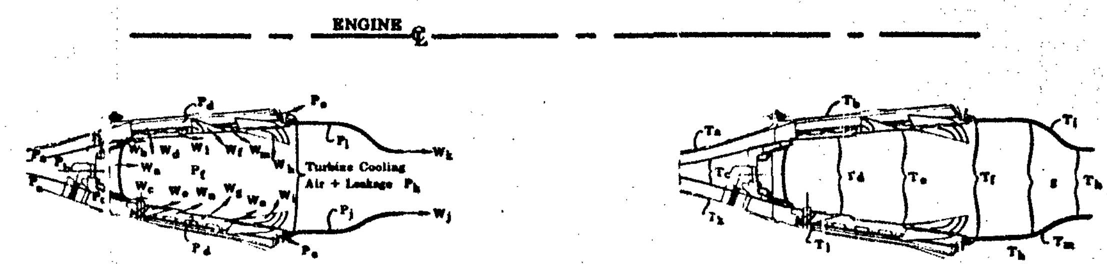

	SLTO	M 1.5, Altitude 45,000 Ft	M 2.7, Altitude 65,000 Ft
W_a	10% Gas Generator	10.47%	10.55%
W_b	2.5 Flow	2.62	2.64
W_c	3.5	3.66	3.7
W_d	5.8	6.07	6.1
W_e	8.5	8.9	9.0
W_f	5.8	6.07	6.1
W_g	8.5	8.9	9.0
W_h	11.6	12.14	12.2
W_i	17	17.8	18.0
W_j	5.26	3.64	3.33
W_k	4.2	2.4	2.2
W_l	2.08	2.18	2.23
W_m	2.08	2.18	2.23
W_n	2.71	2.86	2.91
W_o	2.71	2.86	2.91
P_a	174.5 psia	79.8	67.1
P_b	190.5	86.7	74.9
P_c	182.5	83.0	70.8
P_d	185.5	84.3	72.3
P_e	186	84.5	72.6
P_f	180	81.7	69.4
P_g	178	81.0	68.8
P_h	174	79.0	67.2
P_i	181.2	82.3	71.0
P_j	181.2	82.3	71.0
Turbine Cooling Air + Leakage	7.76% Gas Generator Flow	7.21	6.96

	SLTO	M 1.5, Altitude 45,000 Ft	M 2.7, Altitude 65,000 Ft
T_a	701 °F	741°F	1050°F
T_b	730	770	1080
T_c	745	785	1095
T_d	1695	1680	1640
T_e	1605	1585	1590
T_f	1510	1530	1550
T_g	1315	1360	1515
T_h	1410	1450	1590
T_i	840	880	1190
T_j	840	620	1190
T_k	655	595	1040
T_l	865	910	1110
T_m	810	855	1065

Figure 2. Calculated Metal Temperature, Pressure, and Combustor Airflow Distribution for Takeoff, Acceleration, and Cruise

FD 16991
BIIB

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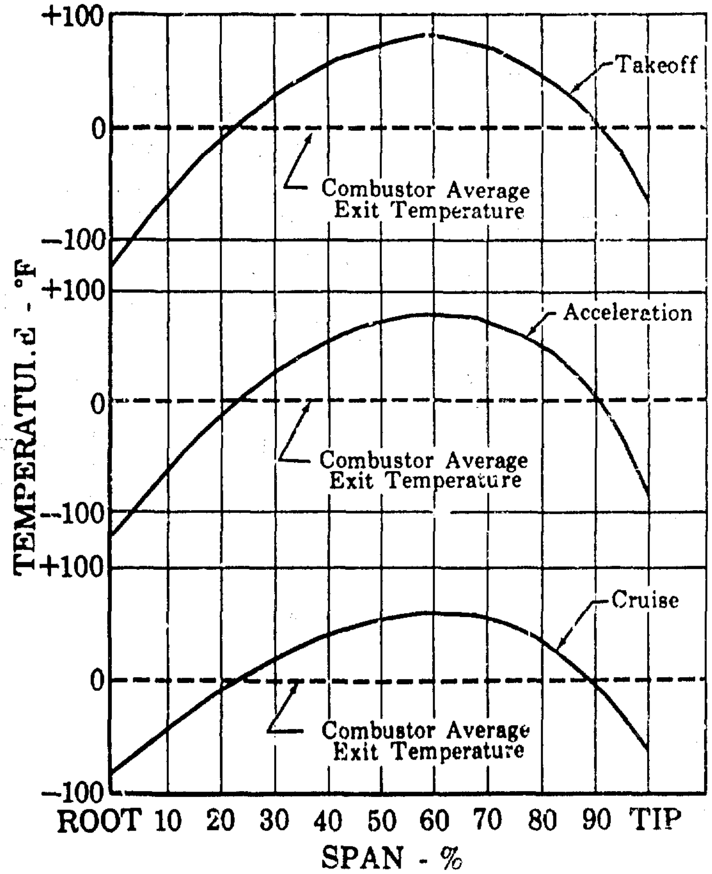

Figure 3. Predicted Primary Combustor Exit Temperature Radial Profiles

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The diffuser case is a weldment consisting of the inner and outer machined forged walls and 12 cast struts, which are butt-welded to airfoil shaped standups integrally machined into the case walls. The struts furnish support for the externally pressure-loaded inner wall. A pair of rings on the ID of the inner wall and the OD of the outer wall, located over the leading and trailing edges of the struts, distribute the loads into the cases.

Turning vanes are cast into the leading edge of the diffuser struts to divert air for the aircraft cabin air-bleed system, aircraft anti-icing system, and the duct heater fuel turbopump. The air is fed from the leading edge of the strut into a manifold welded to the outside of the diffuser case wall. From the manifold, the air is ducted to the airframe.

PWA 1010 (Inconel 718) was selected as the diffuser case and strut material on the basis of cost and strength-to-weight ratio comparisons with other candidate materials. PWA 1010 (Inconel 718) diffuser case weldments have been successful in the initial experimental engine and the J58 production engine. Experience with the J52 and JT80 engines have shown that a cast diffuser strut provides a durable low-cost unit.

Strength-to-weight ratio and thermal compatibility between the inner and outer diffuser walls governed the choice of AMS 5536 (Hastelloy X) as the material for the splitter.

The function of the diffuser is to convert a portion of the compressor discharge air velocity head into static pressure. High velocity (M = 0.35) discharge air is diffused in the two independent, parallel annular passages leading to the combustor inlet. The air at reduced velocity (M = 0.20) is then introduced into the combustion chamber through a series of ram airscoops. These airscoops provide the turbulence and mixing required to obtain the desired combustion efficiency and turbine inlet temperature profile.

Use of the ram-induction combustor reduces the amount of diffusion required, as compared to a static fed burner, and permits a reduction in diffuser length and in total pressure loss usually inherent in the diffusion process.

Combustor development experience has shown that insufficient airflow to the front end of the combustor results in carbon deposits on the combustor forward face. Hydraulic analogy of the initial experimental engine diffuser splitter assembly indicated that the assembly did not provide sufficient airflow to the combustor dome. Verification of this can be seen by the circulation areas upstream of the dome shown in figure 4, and by the low static pressure differentials (less than 1%) recorded on the 120-degree combustion chamber rig. Design studies indicated that eliminating the small central diffuser in the nose of the initial design, and extending the splitter forward to the exit guide vane of the compressor would improve dome airflow characteristics for the production engine. Operating experience with this design in the initial experimental engine duct heater has been excellent with little carbon formation and subsequently no overheating.

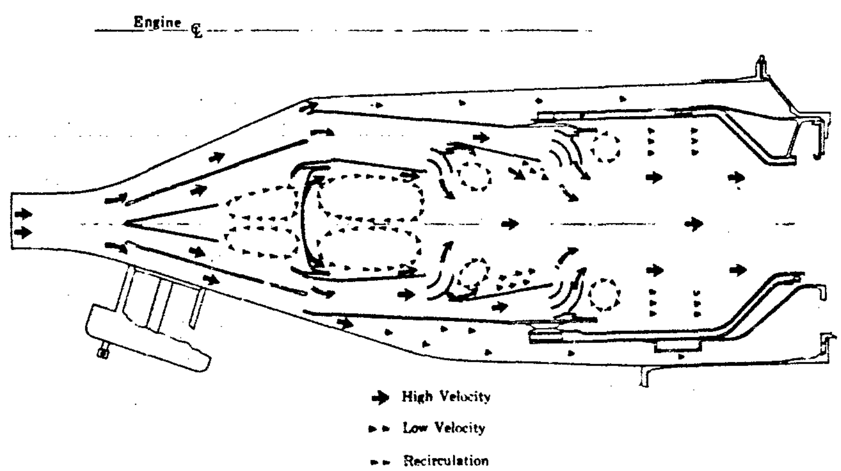

Figure 4. Water Table Flow Pattern

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In the design of the diffuser, emphasis has been placed on attaining a satisfactory turbine inlet temperature profile by minimizing local separation and associated wakes due to flow obstructions such as struts and fuel nozzle stems. Background information provided by P&WA experimental rig cascade testing has shown that to eliminate wakes it is necessary to maintain diffuser strut thickness ratio at a minimum. In addition, providing generous fillets at the intersection of the strut and case reduce interaction of wall boundary layer and the tendency of flow separation. Figure 5 shows an exit temperature profile as obtained from the initial experimental engine. No hot spot problems are apparent. Comparison of the hydraulic analogy of the initial experimental engine, figure 6, and that of the production engine, figure 7, shows that the production engine diffuser permits better flow adhesion to the diffuser walls as a result of the more gradual turning angle allowed with the extended splitter. The modest diffusion rates governing the diffuser flow paths have been based on P&WA commercial engine diffuser designs. (See figure 8.) This comparison indicates that the proposed diffuser is well within the limits required to ensure stable, unseparated flow with good recovery efficiency and low flow losses. Table 2 indicates the design performance for the diffuser.

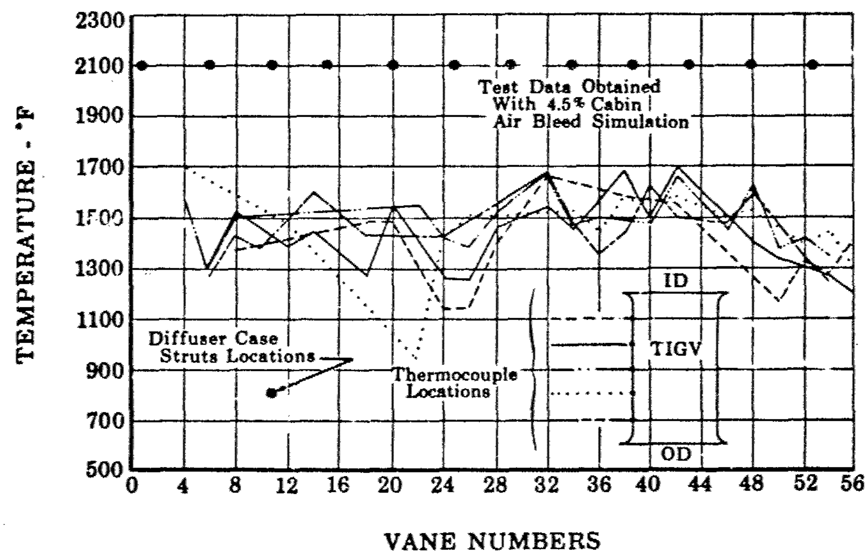

Figure 5. Diffuser Case Strut Locations vs Temperature

FD 16289
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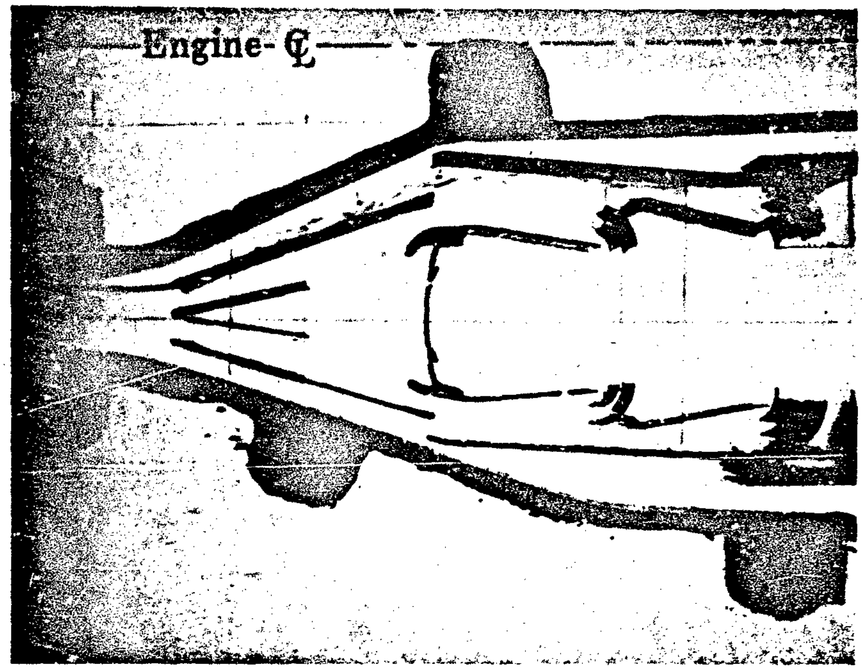

Figure 6. Initial Experimental Engine Primary Combustor on Water Table Showing Streamlines in Inner Diffuser Annulus — FD 16986 BIIB

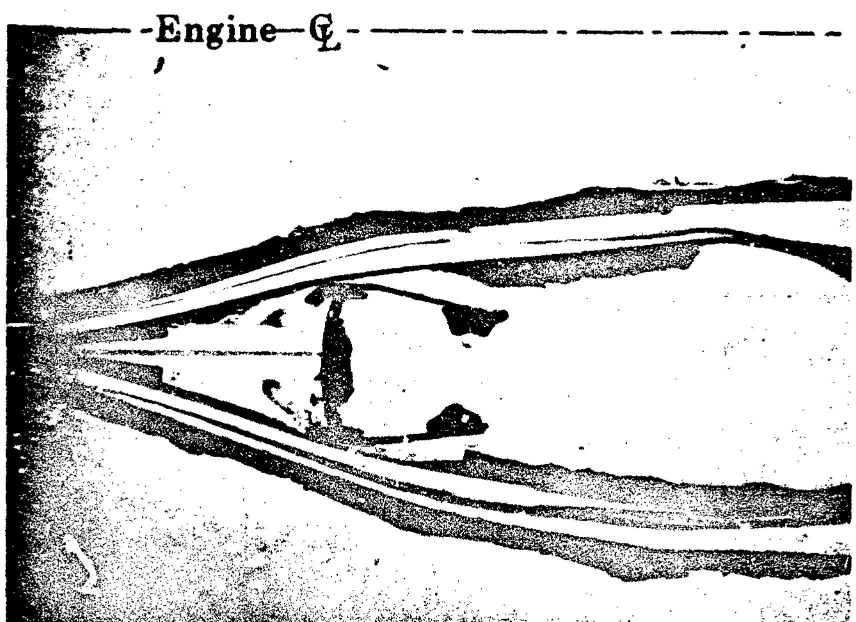

Figure 7. JTF17 Primary Combustor Water Table Model — FC 12241 BIIB

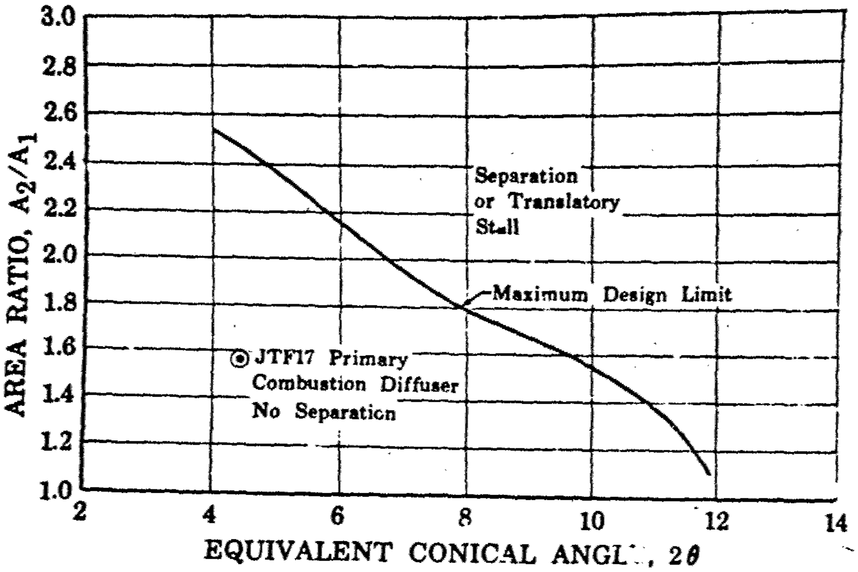

Figure 8. P&WA Diffuser Experience FD 16988 BIIB

Table 2. Primary Combustor Diffuser Design Parameters

	SLTO	Cruise
Mach No. (inlet)	0.347	0.409
V (inlet), ft/sec	515	687
Mach No. (exit)	0.25	0.293
$\Delta P/P_t$, %	1.31	1.84

b. Dome and Swirler

The forward face or dome of the primary combustor is an integral part of the diffuser case splitter weldment. There are 24 swirl cups welded to the dome. Tange ial flow swirlers are then butt-welded into the forward face of the swirl cups.

AMS 5536 (Hastelloy X) was selected as the dome and swirl cup material on the basis of high temperature oxidation resistance. The swirler material, AMS 5382 (Stellite 31) has demonstrated excellent high temperature and anti-erosion characteristics in the J58 combustor.

The purpose of the dome and swirlers is to produce a stabilization region in the primary combustor by the action and interaction of swirler flow and airflow through the first set of scoops. The tangential velocity component off the swirler blading promotes circulation. Static air from inside the splitter is supplied to the louvers formed by the dome, swirl cup, and swirler. The design utilizes the film of air formed on the dome and swirl cup walls to cool the parts. The air film also provides a wash to prevent carbon accumulation, thereby reducing turbine airfoil erosion.

The swirler geometry is governed by the amount of turning required to provide a stabilizing region for combustion. The pitch-to-chord ratio for the production engine swirlers is 0.58. This pitch-to-chord ratio is used on the J58 with excellent swirler characteristics.

c. Ignition System

The ignition system consists of dual 4-joule electric spark igniters. These igniters are designed to extend from the primary combustor case through the duct heater combustor support case struts. This extension allows the igniters to be removed without removing the duct. (See figure 9.) The two igniter bosses on the outer duct may also be used as borescope bosses for inspection of other primary combustor components.

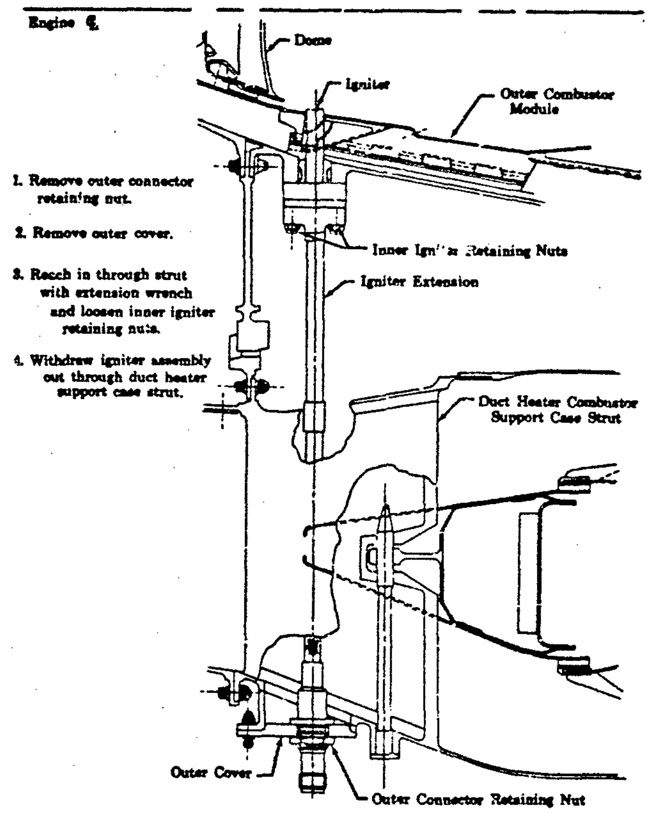

Figure 9. Primary Combustor Igniter Removal FD 16879 BIIB

Ignition tests on the 120-degree segment rig, figure 10, show that the fuel/air ratios are in a region where ignition is rapid. Both the initial experimental engine and full-scale combustor rig tests at sea level start conditions confirm excellent ignition characteristics.

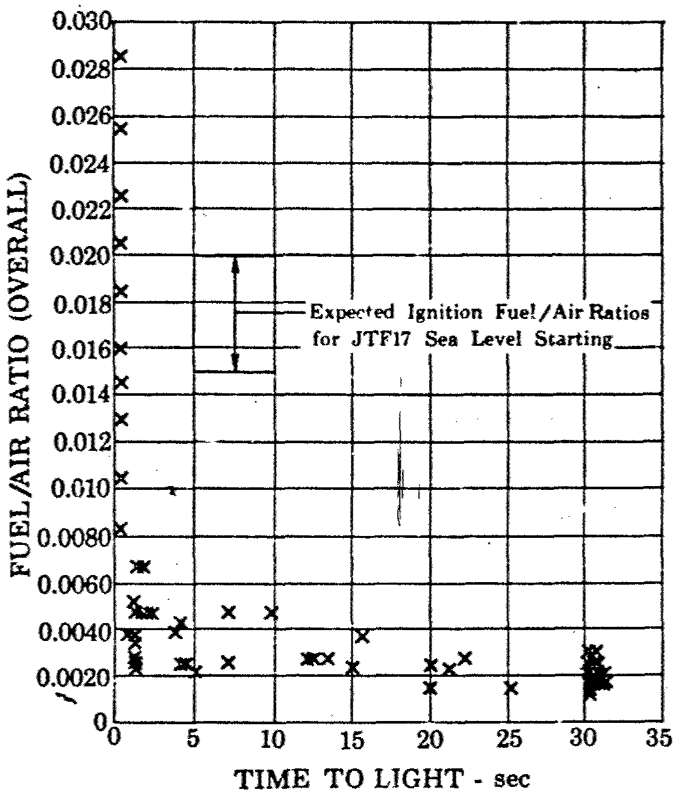

Figure 10. 120-degree Segment Rig Ignition Tests FD 16933 BIIB

d. Fuel Nozzles and Support Housings

The fuel system for the annular, ram-induction combustion chamber consists of 24 individually removable, dual-orifice, variable-area secondary nozzles. The system has a maximum-to-minimum flow ratio capability of 60:1.

The fuel nozzles are housed in individual supports mounted on the diffuser case. Each assembly slides radially inward through the diffuser case and is retained and positioned by a flange. This construction permits ease of inspection or replacement of the nozzles. Removal of a nozzle is illustrated in figure 11. Disassembly of the access port covers and the inner panels on the duct heater exposes the fuel nozzle housing assembly of the primary combustor. The fuel nozzle housing can be removed after the fuel line to the nozzle assembly is disconnected.

The outer end of the nozzle support houses a replaceable fuel strainer. An airfoil-shaped, sheet metal heatshield is attached to the support where it passes through the diffuser flowpath.

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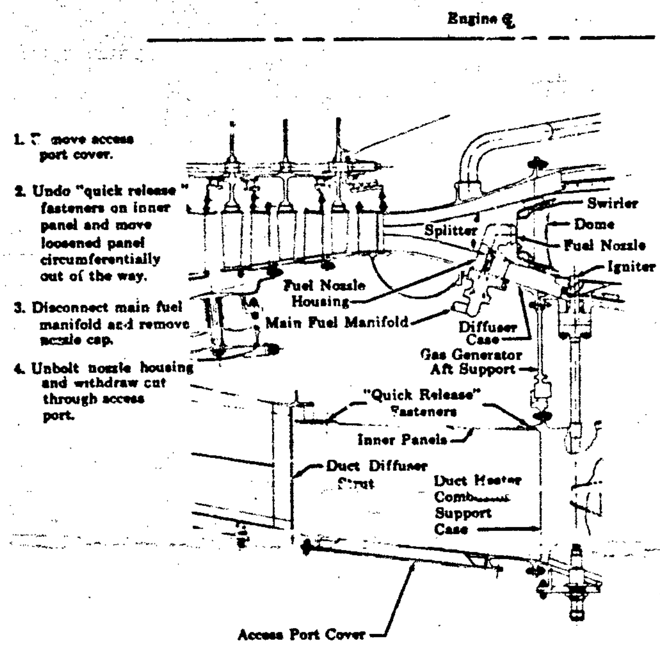

Figure 11. Primary Combustor Fuel Nozzle Removal FD 16880 BIIB

Pressurizing valves for the dual-orifice system are incorporated into the nozzle housings as close as possible to the orifices to avoid fuel boiling and coking effects that might occur in the interconnecting passages if the valves were located farther from the nozzles. An air-sweep guide has been incorporated as part of the nozzle to reduce turbine erosion by preventing carbon accumulation on the nozzle tip, and to augment the fuel swirl provided by the fuel nozzle.

The materials and material selection for the many detailed parts of the fuel nozzle are based on extensive J58 combustor experience at environmental temperatures similar to that in the production engine.

The nozzle support is a one-piece cast AMS 5382 (Stellite 31) structure. The material choice was made on the basis of weight and high temperature capability.

Fuel passage size is limited to those passage sizes which commercial engine experience has shown to be acceptable to ensure minimum susceptibility to contamination and sticking of the variable-area nozzle valve.

The adequacy of nozzle spacing to provide adequate fuel coverage has been proved by the excellent temperature profile results reported on the Performance Report, Volume III, Report A. The fuel system flow characteristics are shown in figure 12. The spray characteristics of the nozzles are illustrated in figures 13 and 14.

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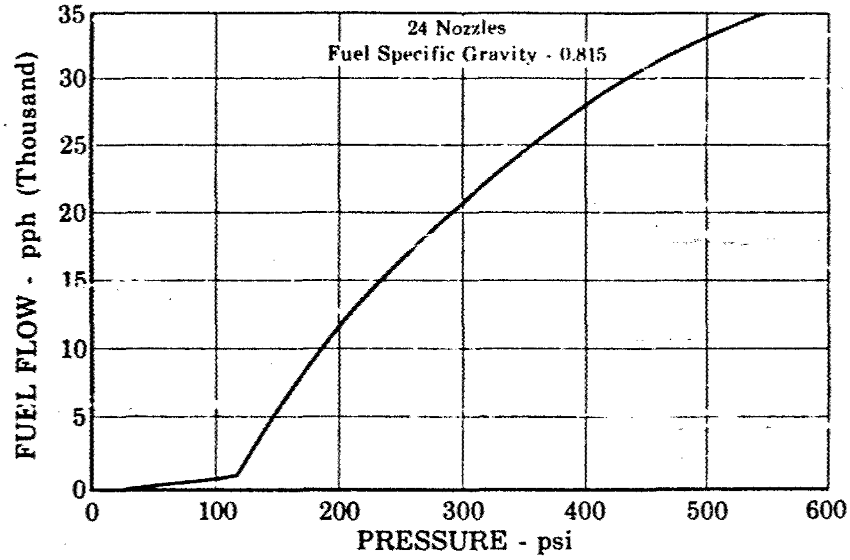

Figure 12. JTF17 Primary Combustor Nozzle Total Flow vs Pressure

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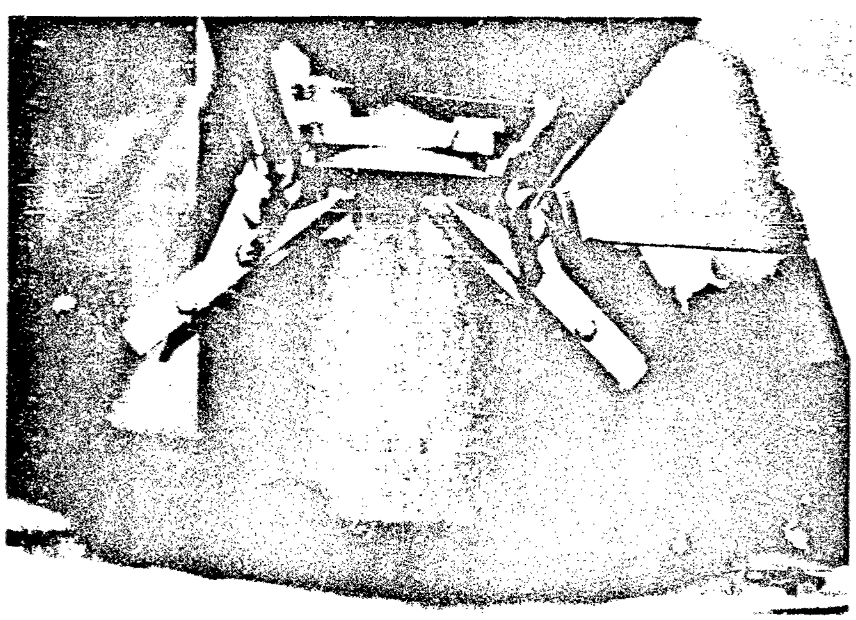

Figure 13. Fuel Nozzle Spray Characteristic (30 psi)

FE 58658A
BIIB

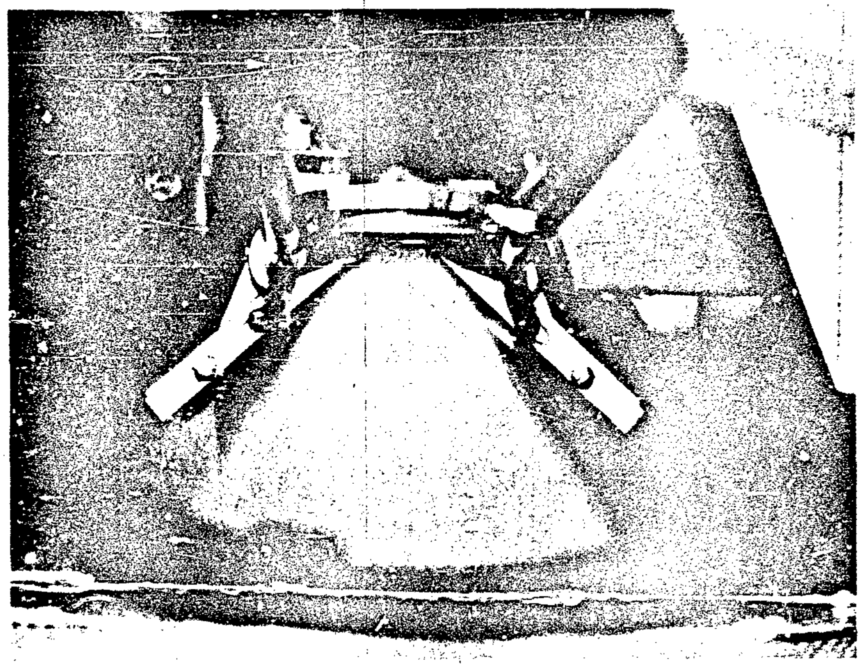

Figure 14. Fuel Nozzle Spray Characteristic (300 psi) FE 58659A BIIB

e. Annular Combustion Chamber

The annular combustion chamber of the initial experimental engine was formed by welding the ram airscoops into a one-piece annular shell that was supported by a cooler secondary wall. This weldment was then retained in the combustion chamber cases by a series of radial pins in the diffuser housing struts. The combustion chamber (figure 1) has been greatly improved by suspending modules containing the ram airscoops directly from the combustion chamber cases. This modular concept minimizes the structural and thermal problems associated with the previous design and improves maintainability. The outer combustor shell is composed of 24 modules with two primary ram airscoops and two secondary ram airscoops per module. (See figure 15.) The inner combustor shell is composed of 24 scoops per module as shown in figure 16. The modules are weldments made from sheet metal stampings. Turning vanes are nicrobrazed into the scoop.

The scoop pattern used in the design is divided axially into three sections; the forward two sections direct the primary air, and the rear section directs the secondary air. Air is introduced into the forward sections through axially staggered ram airscoops mounted on the outside of the shells that compose the combustion chamber. This air provides a stoichiometric fuel/air mixture within the chamber. The single row of secondary scoops is mounted on the shells and is located within the chamber. Air introduced through the secondary scoops ensures uniform, mixing and dilution of the burning primary gases.

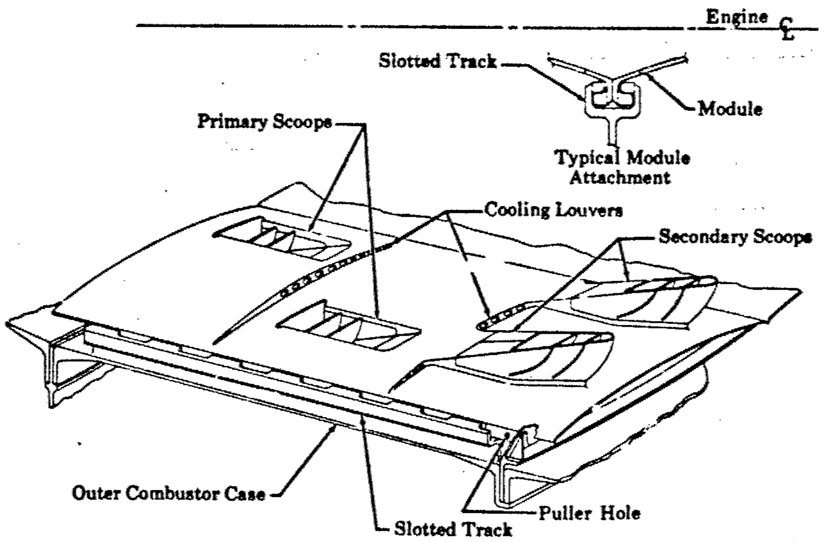

Figure 15. Outer Combustor Module FD 16857 BIIB

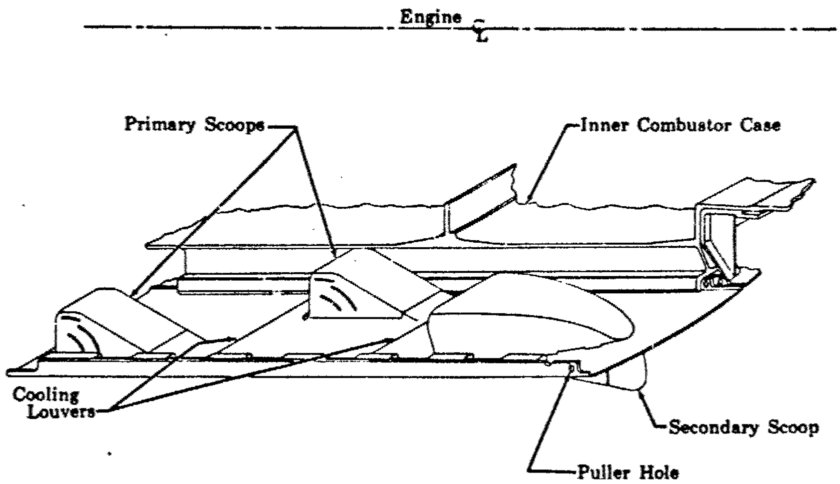

Figure 16. Inner Combustor Module FD 16856 BIIB

Radial support for the scoop modules is provided by slotted tracks butt-welded into the combustor cases as illustrated for the outer case in figure 15. Lugs on the inner and outer scoop modules are positioned between axially spaced lugs on the tracks. The scoop modules are slid axially 0.7 inch to engage the track lugs. Retaining segments prevent further axial movement.

There are eight bosses on the outer case, two of which are used for primary combustor igniters. The remaining bosses can be used for borescope inspection. These borescope ports allow inspection of the primary combustor, fuel nozzles, swirlers, combustor scoops, transition duct, and the 1st-stage turbine vanes.

PWA 1010 (Inconel 718) was selected as the material for the combustor cases on the basis of comparative cost and strength-to-weight ratio with other materials. Module material is AMS 5536 (Hastelloy X) chosen on the basis of high oxidation resistance characteristics and creep properties at elevated temperatures.

The structural design approach was to utilize the ram-induction concept to maximum advantage. Because this type burner does not rely on a static pressure differential across the combustion chamber liner (as does the conventional combustion chamber), the pressure loading of the scoop modules is much lower. The high velocity flow over the modules and through the ram airscoops provide basic cooling by convection with supplemental cooling supplied to those surfaces directly exposed to the hot gases. These inherent features of the ram-induction burner provided for an extremely reliable and lightweight design.

The modular concept used in the annular combustion chamber minimizes the buckling load on the outer combustion chamber wall. Use of the modules effectively breaks the external pressure load on the wall into a number of small inward loads that are sustained by the hoop-loaded outer case. The buckling load imposed on the inner burner case is partially counteracted by the hoop-loading of the inner combustion chamber wall.

The concept of supporting the combustor and transition duct from the inner and outer burner case walls ensures structural efficiency for the design, and provides the following other advantages:

1. Allows removal of the entire annular combustor or replacement of individual modules without removing the turbine.

2. Avoids thermal growth differentials resulting in supporting and sealing problems by allowing the modules to flex and move with the case.

Full-scale rig testing of the initial experimental engine combustion chamber demonstrated excellent combustion performance. Further development of the combustion chamber in a 120-degree segmented rig resulted in modifications that simplified the construction and increased the durability.

The rig tests indicated that the production engine combustor, which uses internal secondary scoops and external primary scoops as compared to the initial experimental engine combustor that has internal scoops for both secondary and primary, equals the performance of the initial experimental engine. A reduction in the number of required primary scoops and the use of external scoops in the primary, simplifies construction and eliminates primary scoop cooling air requirements.

The number and arrangement of the ram airscoops required in the production engine to furnish the desired temperature and temperature profile to the turbine are shown in figure 17. The ram scoop configuration is contoured to (1) minimize drag, (2) minimize the associated wakes that result in a reduction of pressure potential for mixing, and (3) to improve the temperature profile. The importance of available pressure loss for mixing is evident from the segmental rig test data.

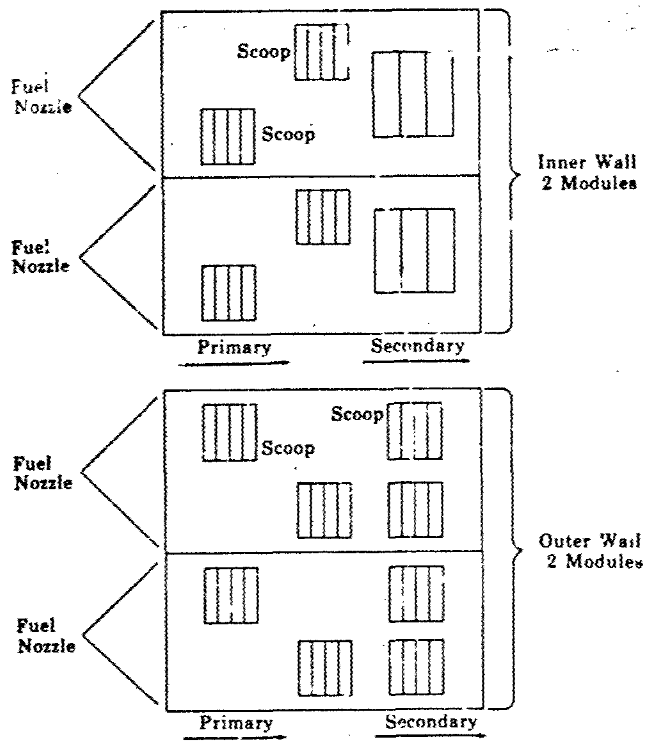

Figure 17. Schematic of Scoop Arrangement FD 16264 BIIB

The scoop aerodynamics for the production engine design were established by full-scale rig testing. The tests indicated that scoop discharge shape affects both penetration characteristics and pressure losses. Penetration correlations developed for different discharge hole shapes show that external scoops with square holes provide adequate penetration for the primary combustor zone. Figure 18 shows the similarity of penetration for the initial experimental engine and the production engine.

The secondary scoops require better penetration characteristics than the primary scoops because of the higher combustion gas momentum at the secondary scoop injection point. The rig test data show that scoops with high length-width ratios provide better penetration than scoops with square shaped discharge. Figure 19 shows a comparison at similar flow conditions for a square scoop discharge configuration and the 1.55 length-to-width ratio configuration that was selected for the final design.

Combustor primary airflow in the production engine design has been increased from that used in the initial experimental engine combustor. Phase II-B statistical combustor air distribution data and Phase II-C full combustor rig testing have shown that a lean primary mixture reduces carbon and smoke formation and produces a more desirable temperature

profile. The variation of percentage of open air distribution (which is directly proportional to the airflow) with burning length for the initial experimental engine combustor and the production engine design is shown in figure 20. This figure also illustrates the shift of airflow to the combustor primary scoops, which results in a leaner primary mixture.

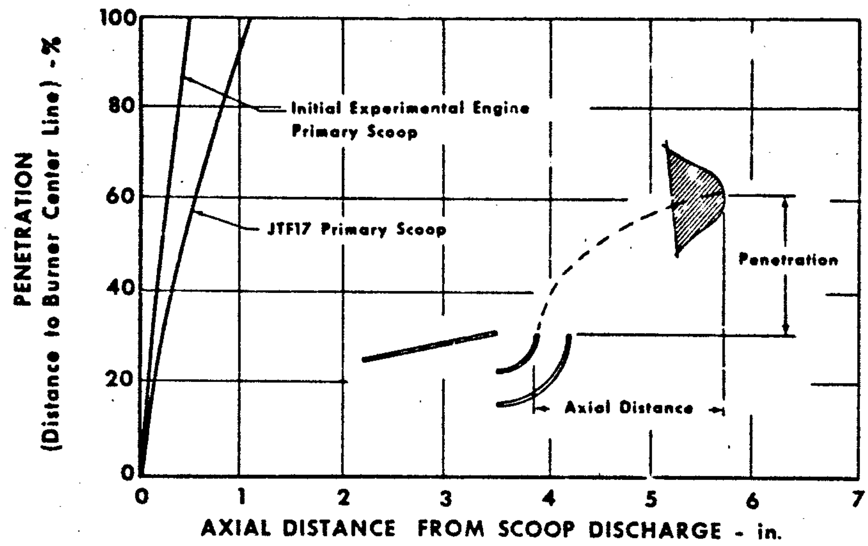

Figure 18. Primary Scoop Penetration FD 16260
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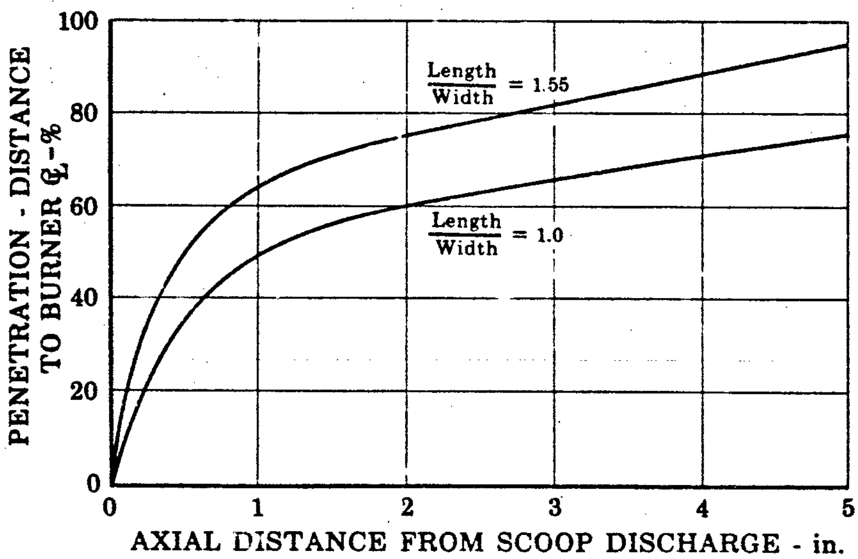

Figure 19. Secondary Scoop Comparison FD 16272
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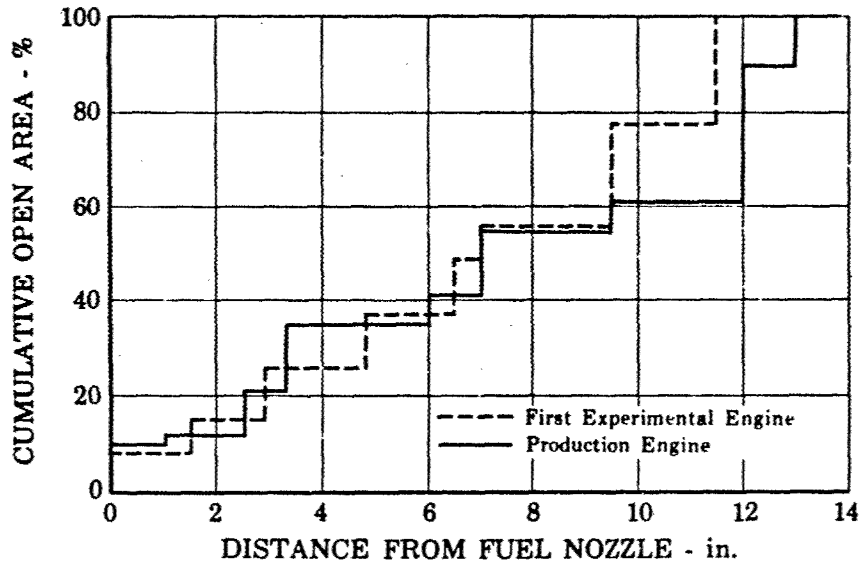

Figure 20. Primary Combustor Open Area Distribution vs Burning Length FD 16268 BIIB

Development of the initial experimental engine combustor has shown that staggering scoops between the ID and OD walls also helps to reduce carbon formation and increase combustor durability. This feature is incorporated into the production engine scoop pattern. The single inner and outer rows of secondary scoops for the production engine design are located in the same axial plane, but the inner and outer scoops are alternated around the circumference. Segmental rig test data have shown the capability of a single row of secondary scoops to produce a flat exit temperature profile. (See figure 21.) The penetration characteristics of the production engine and the initial experimental engine secondary scoops are shown in figure 22.

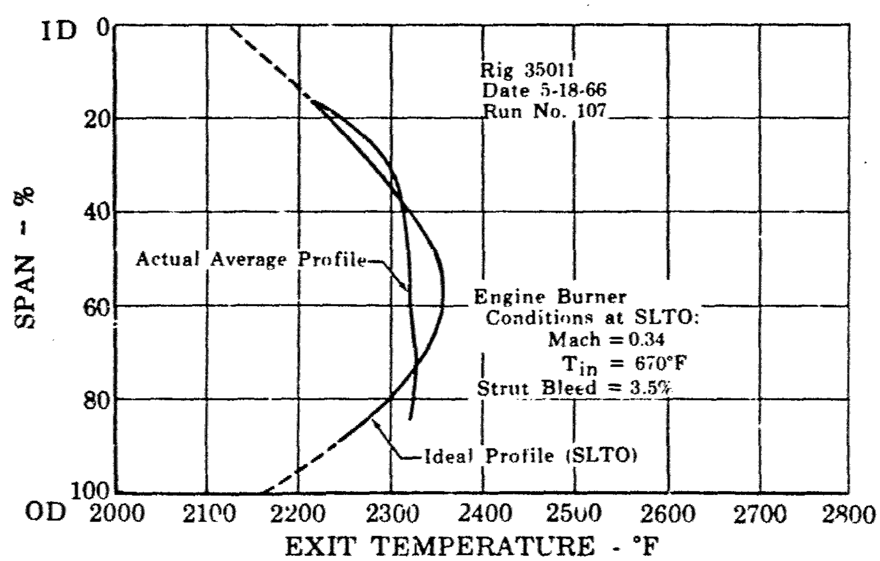

Figure 21. Radial Temperature Profile FD 16269 BIIB

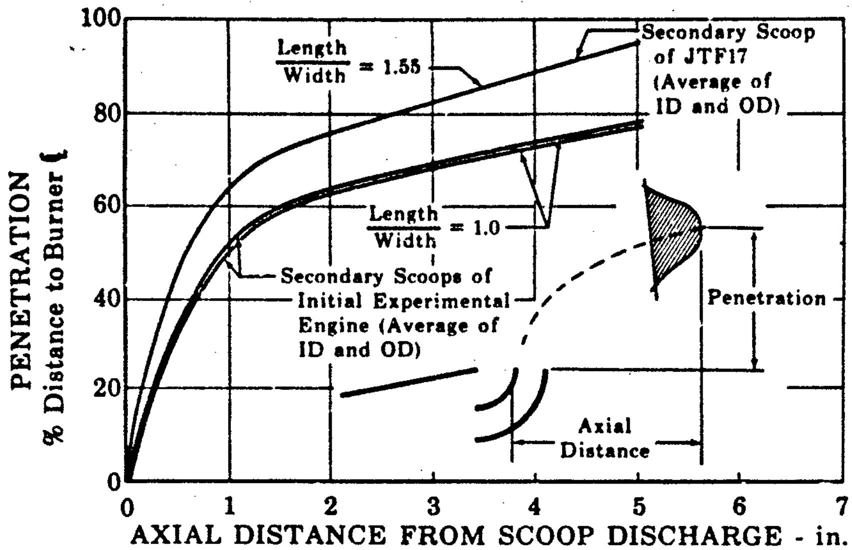

Figure 22. Secondary Scoop Penetration FD 16261 BIIB

Early tests on the full-scale JT4 combustor rig showed evidence of scoop burning on the discharge edges, and overheating of the internal scoop ramps. (See figure 23.) Improved cooling techniques have since been developed and successfully demonstrated. (See figure 24.) These techniques have resulted in increased reliability. Furthermore, additional air is allowed to spill over the scoop edges to provide an insulation blanket between the hot combustion gases and the scoop walls in the production engine. Penetration air is kept in a separate partition in the scoop and guided to the scoop exit. The desired scoop mass flow, velocity, and area characteristics are illustrated in figure 25.

Substantiating performance data, in addition to that already discussed, may be obtained in Volume III, Report A. The following pertinent data are reported:

1. Outlet temperature profile map and radial profile
2. Efficiency
3. Pressure loss
4. Fuel/air ratios
5. Combustor metal temperature profiles
6. Combustor stability during operation and inflight lightoff
7. Heat release rates
8. Cooling air effects

Figure 23. Mod 5-1D Primary Combustor Burning and Carbon Accumulation FE 55027 BIIB

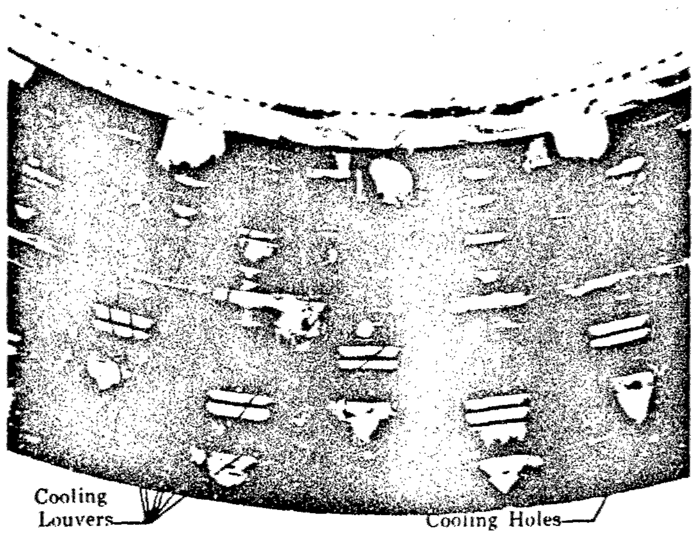

Figure 24. Mod 5-1P Primary Combustor After Testing FD 16707 BIIB

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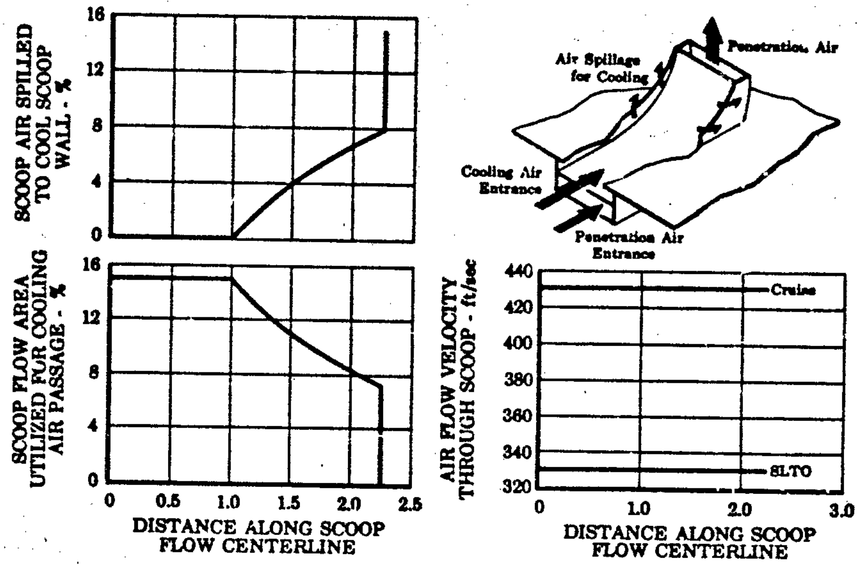

Figure 25. Internal Secondary Scoop Design Characteristics

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f. Transition Duct

The transition duct section of the primary combustor is an annular passage directing combustor gases from the combustor to the turbine.

The convectively cooled duct is formed by an inner and outer shell section consisting of 15 circumferential segments in the inner shell, and 24 in the outer shell. The higher temperature gas path wall of these modules is attached to the cooler structural wall by a bushing-and-fastener arrangement. (See figure 26.) This type construction permits the hotter wall to float in both the axial and tangential directions to accommodate the thermal growth differences between the two elements.

AMS 5536 (Hastelloy X) was chosen as the transition duct material on the basis of excellent high temperature oxidation resistance properties and thermal compatibility with other primary combustor components. The capability of AMS 5536 (Hastelloy X) as a burner material has been demonstrated on numerous P&WA commercial engines and the J58.

The unique modular construction of the transition duct and the annular combustion chamber evolved from design studies directed toward:

1. Low replacement and repair costs
2. Minimum spare part storage requirements
3. Maximum maintainability.

The modular construction allows for replacement or repair of individual modules rather than on entire duct or chamber. Spare elements are small and can be easily stored.

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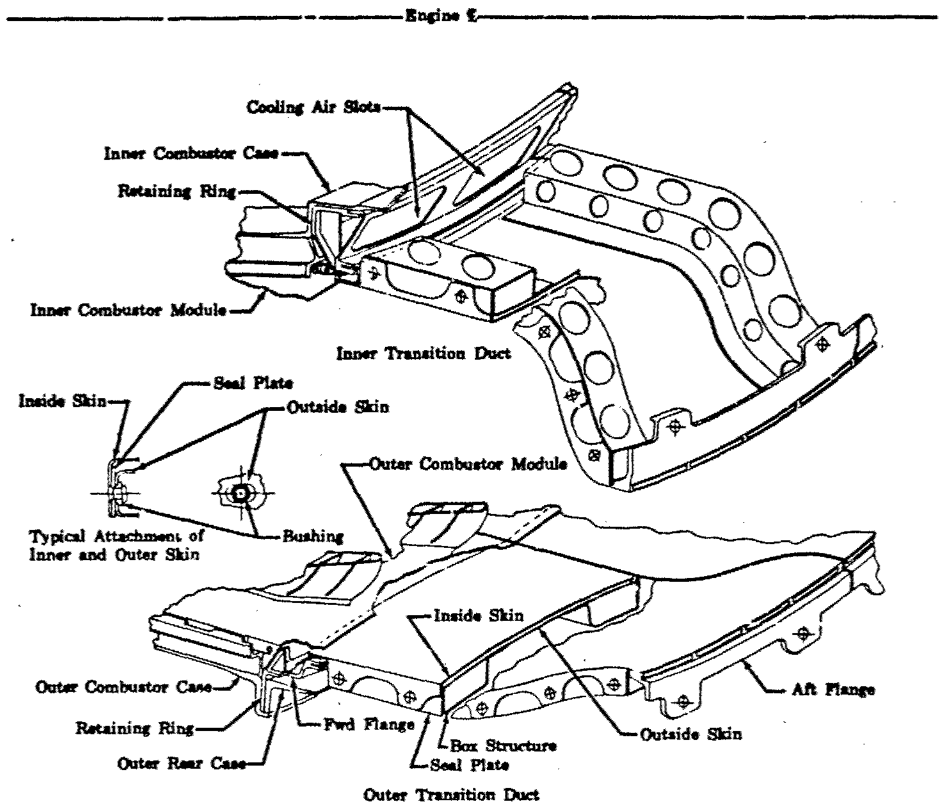

Figure 26. Inner and Outer Transition Duct Module

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Figure 27 illustrates the maintainability features of the design. With the reverser-suppressor removed from the engine, and the engine supported on its mount system, the various components of the duct heater, shown in figure 27, can be removed to expose the split outer case of the primary combustor. With the removal of this case, the OD transition duct modules are exposed. Removal of these modules provides access to the OD combustor scoop modules, and the 1st-stage turbine vanes. Removal of the 1st-stage turbine stator vane support segments, the paired 1st-stage turbine vanes, and the ID transition duct modules allows the ID combustor scoop modules to be removed.

Various methods of cooling the transition duct were studied to establish the most efficient design considering heat transfer, structure, manufacturing, weight, and cost requirements. Figure 28 illustrates transition duct metal temperature versus percent cooling air for the convection cooling of an annulus, a tube wall, and a louver-cooled structure. The modular convective annulus design with a cooling airflow path height of 0.1 inch for the inner and outer section of the transition duct proved to be the best design choice for the production engine.

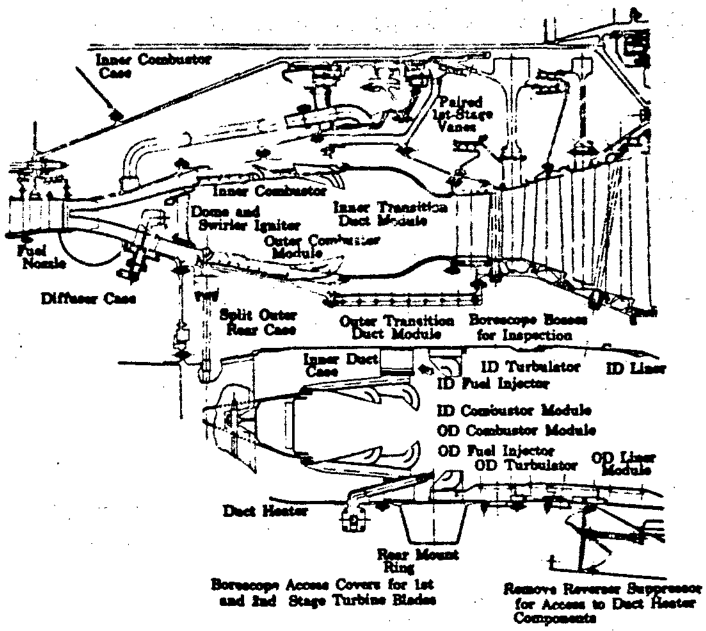


Figure 27. JTF17 Primary Combustor Maintainability FD 16989 BIIB

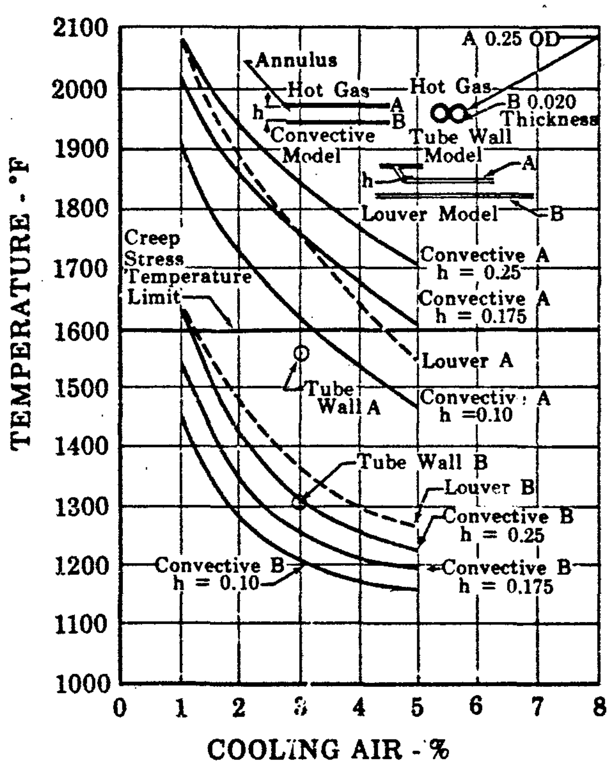


Figure 28. Transition Duct Trade Study FD 16271 BIIB

The reliability of the convective cooling concept used for the transition duct has been successfully demonstrated on the initial experimental engine and on the full-scale combustion rig tests utilizing the JT4 engine.

g. Inner and Outer Rear Cases

The inner and outer rear cases of the primary combustor contain the transition ducts, and provide boundaries for the turbine cooling airflow. The outer case is split into two 180-degree segments to permit access to the modular combustor, transition duct, and 1st-stage turbine vanes. (See figure 29.) A 10-inch, axial access is also provided for inspection or minor repairs in the primary combustor.

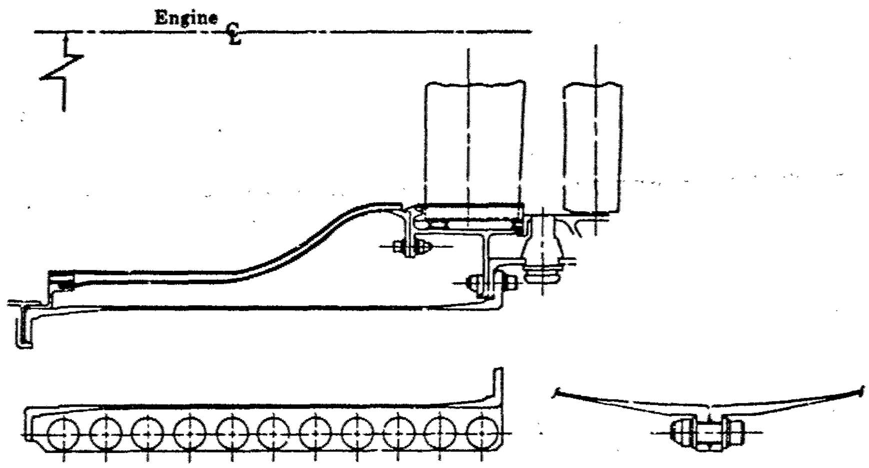


Figure 29. Primary Combustor Outer Rear Case FD 16273 BIIB

The outer rear cases are designed with a simple valve to automatically open after a false start or engine shutdown. Opening of the valve permits drainage of combustion fluids from the combustion areas.

Case material is PWA 1033 (Inconel 718) selected on the basis of cost and strength-to-weight ratio and weld repairability comparisons with other materials.

h. Primary Combustor Temperature Analysis

Primary combustor wall cooling requirements are determined through the use of a digital computer which performs an analysis of a thermal nodal network of one-dimensional heat flow. Equations describing the heat flux into a combustor wall element are based on both radiation from the luminous combustion products and combustion gas forced convection. Heat is rejected from the element by (1) forced convection to the high velocity shroud air, (2) radiation to the cases, and (3) film cooling techniques of conventional multinozzle combustors. Combustion gas temperatures vary with burning length and are determined by the local overall fuel/air ratio.

4. Materials Summary

Materials selected for the primary combustor are as follows:

Diffuser Case	PWA 1010 (Inconel 718)
Struts	PWA 649 (Inconel 718) (Cast)
Splitter	AMS 5536 (Hastelloy X)
Dome	AMS 5536 (Hastelloy X)
Swirl Cups	AMS 5536 (Hastelloy X)
Swirlers	AMS 5382 (Stellite 31)
Nozzle Supports	AMS 5382 (Stellite 31)
Scoop Modules	AMS 5536 (Hastelloy X)
Combustor Cases	PWA 1010 (Inconel 718)
Transition Duct	AMS 5536 (Hastelloy X)
Inner and Outer Rear Cases	PWA 1033 (Inconel 718)

5. Product Assurance Considerations

Specific maintainability, reliability, safety, and value engineering features are listed below. These features were discussed in detail within the design approach section.

a. Maintainability

1. Igniters are designed to extend from the primary combustor through the duct heater case, allowing removal of the igniters without removing the duct.

2. Individual support of the fuel nozzles allows inspection or replacement of nozzles without removal of the primary combustor.

3. The combustion chamber and transition duct are of modular construction which permits replacement or repair of individual elements of the combustor.

4. Support of the combustor and transition duct from the inner and outer burner case walls permits removal of the entire annular combustor or replacement of individual elements of the combustor without removing the turbine.

5. Complete removal of the annular combustor may be accomplished with the engine supported on its integral mount system.

6. The outer rear case is split into two 180-degree segments to permit access to the modular combustor, transition duct, and 1st-stage turbine vanes.

7. Weld repair capability is characteristic for all case material selections.

8. A 10-inch, axial access is provided for inspection or minor repairs in the primary combustor.

9. A total of eight borescope ports, including igniter bosses, allow inspection of the fuel nozzles, swirlers, combustor scoops, transition duct, and the 1st-stage turbine vanes.

b. Reliability

1. Diffusion rates governing the diffuser flow paths are well within the limits required to ensure stable, unseparated flow with good recovery efficiency and low flow losses.

2. Adequate cooling is used in high temperature regions to ensure long life for component parts.

3. Pressurizing valves are incorporated in the nozzle housing as close as possible to the discharge orifices to avoid fuel boiling and consequent fuel coking.

4. Thermal problems are minimized in the design of the annular combustion chamber by suspending individual modules from the case walls.

5. Buckling loads on the outer combustion chamber wall and the inner burner case are counteracted by hoop loadings on the outer burner case and the inner combustion chamber wall, respectively.

6. Component wear surfaces are hardcoated and easily replaced.

7. Erosion of the turbine as a result of carbon formation in the combustor is minimized by providing an air wash to prevent carbon accumulation.

8. Extensive use on other commercial engines has proven the reliability of the ignition system.

9. Fuel passage size is limited to those sizes that commercial engine experience has shown to be acceptable to ensure minimum susceptibility to contamination and sticking of the variable area nozzle valve.

10. Fuel nozzle strainers are retained in the nozzle housings during disassembly of the fuel manifold, thereby protecting the fuel nozzles from contamination during handling.

c. Safety

The design provides for automatically clearing the combustion areas of combustible fluids after a false start or engine shutdown.

d. Value Engineering

1. Cost and strength-to-weight ratio comparisons were made for all materials chosen for the primary combustor.

2. Experience with the J58 and the initial experimental engine has shown that large case weldments from PWA 1010 (Inconel 718) have excellent repairability.

3. Sheet Metal construction has been used wherever practical to avoid costly forgings.

4. Cast diffuser struts provide a low cost unit as shown by experience on the J52 and JT8D engines.

5. Modular construction of the combustion chamber and transition duct allows for replacement or repair of individual low cost, easily stored elements rather than an entire chamber or duct.

6. Engine ground handling costs have been minimized by allowing the removal of major combustor components with the engine supported on its integral mount system.

C. TURBINE DESIGN

The JTF17 turbine is a three-stage axial flow reaction type that employs a controlled vortex flow pattern to attain high efficiency at low velocity ratios. A single high pressure stage drives the high pressure compressor and two low pressure stages power the fan. The objective was to design a low weight turbine with sufficient efficiency and power margin to achieve an engine with a high thrust-to-weight ratio that delivers high performance. A three-stage turbine was selected as a result of optimization studies evaluating the weight, diameter, efficiency, cost, and life trade factors necessary to set the flow path for the optimum engine. In sharp contrast to current commercial transport engines, the SST engine will spend approximately 50% of its operating life at or near its maximum turbine temperature. This fact is brought out in figure 1, which shows the amount of operating time over the life of the engine that will be spent at various turbine temperatures. Note that current engines spend very little time at maximum turbine temperature. Because of the long time at severe operating conditions (turbine inlet temperature of 2200°F) proper turbine cooling and careful selection of turbine materials are of primary importance. Much of the discussion which follows treats our approach to these important areas of turbine design.

The aerodynamic design of the turbine, aimed at achieving light weight and adequate efficiency, is also discussed in this section of the proposal as are mechanical design features of the machine. Our method of integrating the independent turbine test programs to arrive at the final airfoil configurations is illustrated in figure 26 section 1, report 4 of Volume 5.

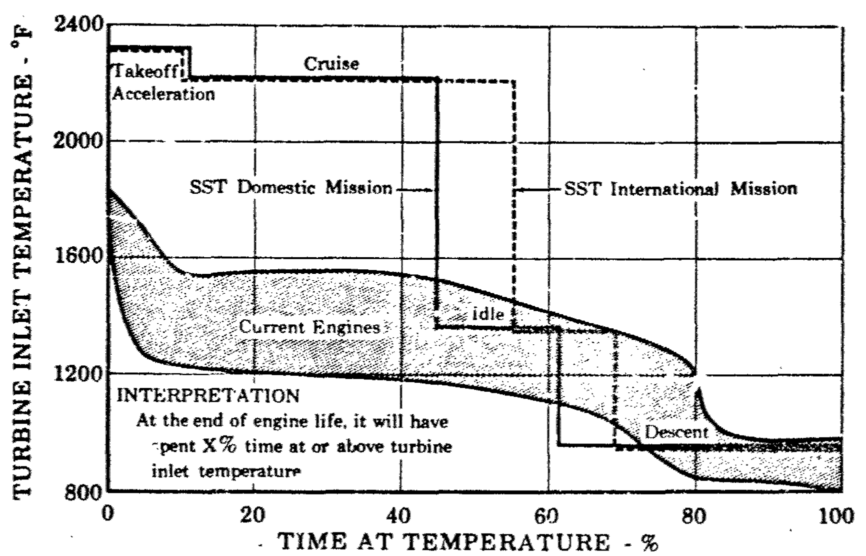


Figure 1. Turbine Inlet Temperature Schedule FD 16218 BIII

1. Turbine Cooling

Recognizing that the supersonic transport imposes the requirement on commercial engines of cruising at high Mach number for prolonged periods near maximum turbine inlet temperature, we undertook a comprehensive program directed toward the development of turbine cooling methods that would

make such operation practical. This program covered not only the heat transfer and cooling aspects of the problem, but the equally important materials and fabrication aspects as well.

The goal of this work was an objective evaluation of all of the important factors to establish a solid technical foundation for our SST engine design. Convective, film, and transpiration cooling methods were analyzed, built and tested both in experimental test rigs and in full-scale engines. It is frequently assumed that at the heat fluxes associated with high turbine inlet temperatures, film or transpiration cooling must be used because convection cooling cannot adequately protect the metal. However, in rocket engines we have applied convective cooling successfully at heat fluxes ten or twenty times those encountered in the turbines of supersonic air-breathing engines. Therefore we have investigated the application of these advanced convection cooling techniques, as well as film and transpiration cooling methods, to turbine cooling.

The fundamental criterion used for the heat transfer evaluations is the cooling effectiveness (ϕ) defined as follows:

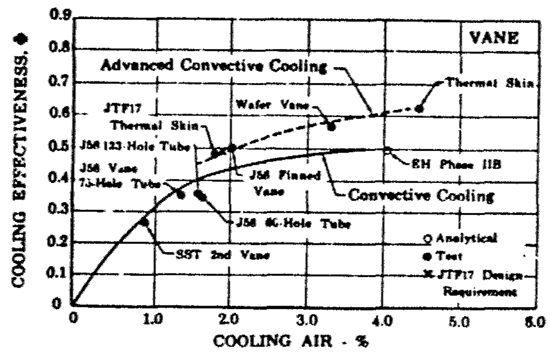
$$\phi \equiv \frac{T_{gas} - T_{metal}}{T_{gas} - T_{coolant}}$$

As can be seen from its definition, the cooling effectiveness is a measure of how well the coolant is used in controlling the metal temperature. Since the use of cooling air lowers the turbine efficiency (discussed in detail in Volume III, Report A) it is necessary to keep the amount used as small as possible. This means that particularly for the hottest, most severe environment, the cooling effectiveness must be as high as possible. Figure 2 summarizes the results of some of the tests performed on various cooled vane and blade designs. The results are presented as a function of cooling air flow expressed as a percentage of the engine gas generator flow. The test data points shown are representative of some of the 63 configurations tested at FRDC. These data were further substantiated by testing of 75 more configurations at our East Hartford facility. The test results can be grouped into three general categories: current operational convective, advanced convective, and film cooled. The difference between the advanced convective and the current operational convective method is the cooling heat transfer coefficient augmentation provided by impingement cooling of the leading edge and high coolant velocities on the inner wall.

The fundamental reason for cooling the turbine airfoils is to lower the metal temperature enough to obtain long life from the available materials at the stress level applied in the engine. Calculations were made using the experimental data presented above to determine the cooling method to be used in the JTF17 engine and the percent cooling air required. Our previous engine experience, rig test data, and material properties research indicated that the average metal temperature should be kept below about 1700°F, to provide reasonable turbine life. The results of such an analysis are summarized in table 1.

Table 1. Results of Airfoil Cooling Analysis

Airfoil	1st-Stage Vane	1st-Stage Blade	2nd-Stage Vane	2nd-Stage Blade	3rd-Stage Vane	3rd-Stage Blade
Average Mid-Span Gas Temperature, °F	2256	2011	1830	1721	1600	1509
Average Metal Temperature, °F	1700	1644	1617	1616	1543	1509
Cooling Flow, %	1.9	2.0	1.0	0.5	0.3	0
Cooling Effectiveness	0.48	0.421	0.31	0.18	0.12	0



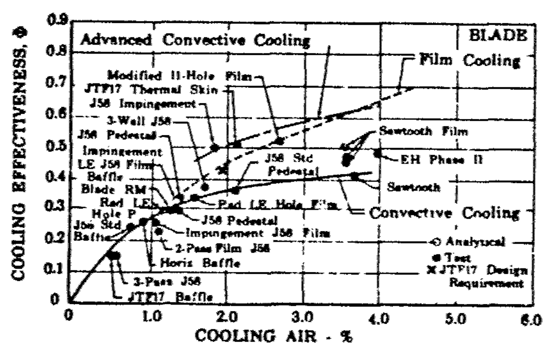


Figure 2. Blade and Vane Cooling Effectiveness FD 17093 BIII

It can be seen from the table that the desired metal temperatures can be obtained with relatively low cooling air flows in all turbine stages and with cooling effectiveness values that have been demonstrated. For the first-stage vanes and blades, which run in the hottest environment, a cooling flow of about 2% is needed at an effectiveness in the 0.4 to 0.5 range. Figure 2 shows that either advanced convective cooling or film cooling can meet the requirement. We have selected the advanced convective cooling rather than film cooling for the following reasons.

1. Demonstrated Producibility and Reliability - The JTF17 airfoil cooling system proposed is substantially that used in current production J58 engines, with improved heat transfer performance. In the course of developing and producing the J58, the turbine airfoil manufacturing and quality assurance processes have been thoroughly established. In thousands of hours of operation above 2000°F turbine inlet temperature, the dependability of this cooling configuration has been demonstrated.

2. Higher Aerodynamic Efficiency - The aerodynamic losses of the convective-cooled airfoils are lower than losses with film-cooled airfoils. All of the cooling air is discharged parallel to the main gas stream at the trailing edge where it serves to energize the wake and minimize losses. No cooling air is discharged into the flow passage between blades, thus there is no disturbance to the main gas stream flow and boundary layer. Test date shown in figure 3 illustrates the change in turbine stage efficiency between the trailing edge discharge convective method and a film-cooled airfoil. An efficiency advantage of 1.2 percentage points is shown for the trailing edge discharge method.

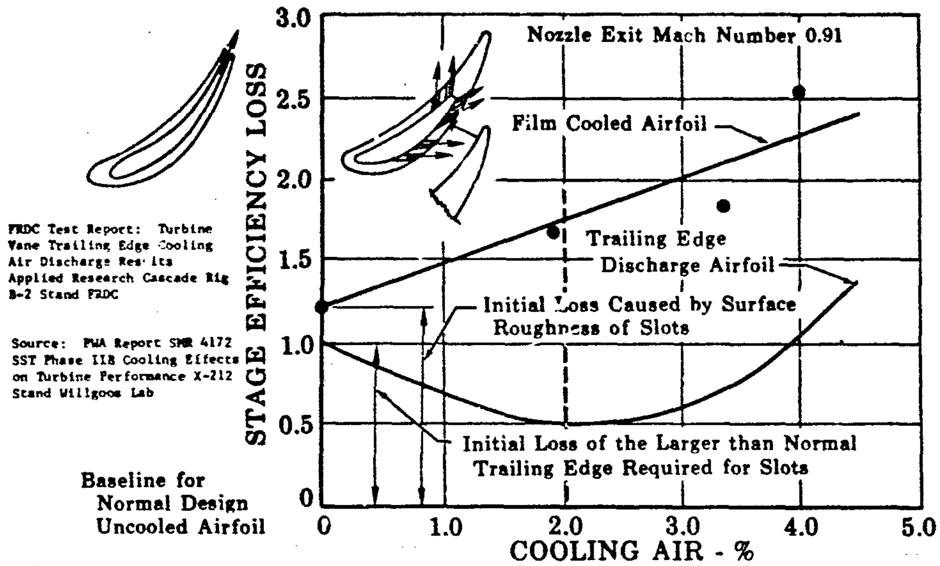


Figure 3. Turbine Stage Efficiency Loss vs Percent Cooling Air — FD 16369 BIII

3. Greater Foreign Object Damage Resistance - Because of the superior cooling afforded by impingement, the leading edge can be made solid, thereby eliminating film cooling slots that could clog or be closed by foreign object damage. Figure 4 shows representative foreign object damage that has occurred in a commercial engine, namely the JT3C.

4. Greater Low Cycle Fatigue Life - The exterior wall of the airfoil has no holes or slots having severe temperature gradients, thereby eliminating the stress risers that are the major factor in low cycle fatigue.

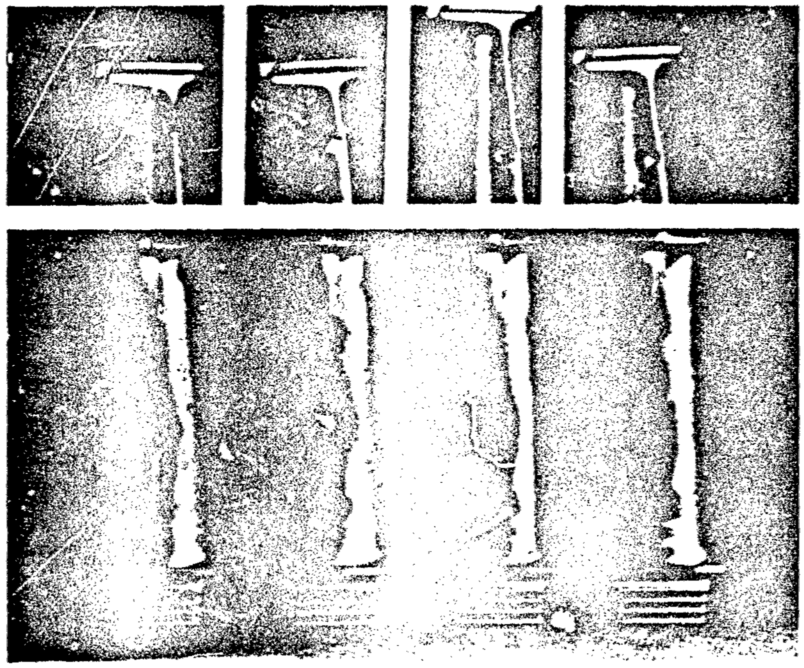


Figure 4. JT3C 1st-Stage Blades Showing Impact Damage (Commercial)　　XP 33603 BIII

5. Easier Protective Coating Application - Without the multitude of tiny slots that characterize film-cooled airfoils, protecting coatings are easier to apply to the convective-cooled airfoil. Also, stripping and recoating is likewise simplified.

Referring again to table 1, it can be seen that for the second and third stages, where the environment is less severe than in the first stage, relatively little cooling air is needed even with low cooling effectiveness to achieve the desired metal temperatures. Therefore, from the experimental data of figure 2, it is clear that convective cooling configurations used in current engines are adequate without refinement. Here again, as with the 1st-stage airfoils, the selection of convective cooling for the second and third stages allows us to take advantage of a well-developed technology to effect lower costs and proved reliability. The operating environment of the 3rd-stage blade is such that a solid, uncooled blade is adequate.

Figures 5 and 6 show the designs selected for the 1st stage vanes and blades. The cooling passages, attachments, and general construction are illustrated. As can be seen in figure 7, the 1st-stage blades selected for the JTF17 are similar to those used in the J58 engine in size, shape, and manufacturing technique. The more than 20,000 hours of test stand and service time accumulated on J58 convectively cooled blades provides confidence that this type of blade will perform properly in the JTF17. The JTF17 turbine blade life will be improved over that of the J58 by cooling the JTF17 1st-stage turbine blade to an average metal temperature of 150°F lower than on the J58.

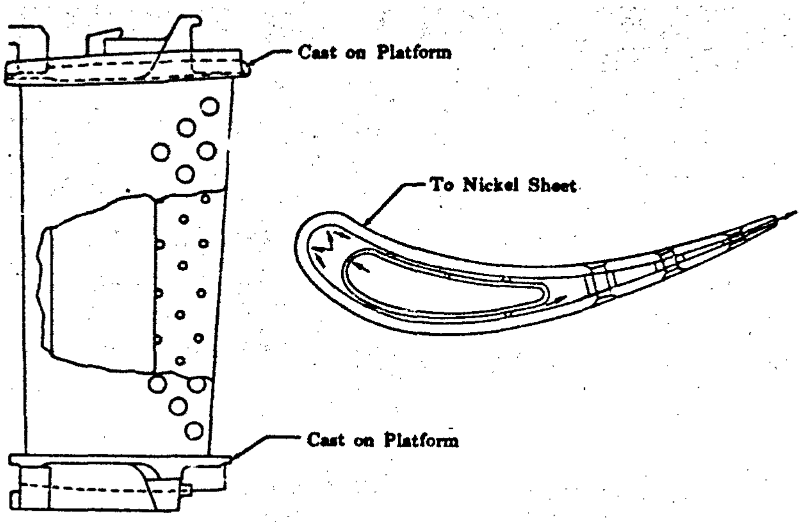


Figure 5. JTF17 1st-Stage Vane FD 16774 BIII

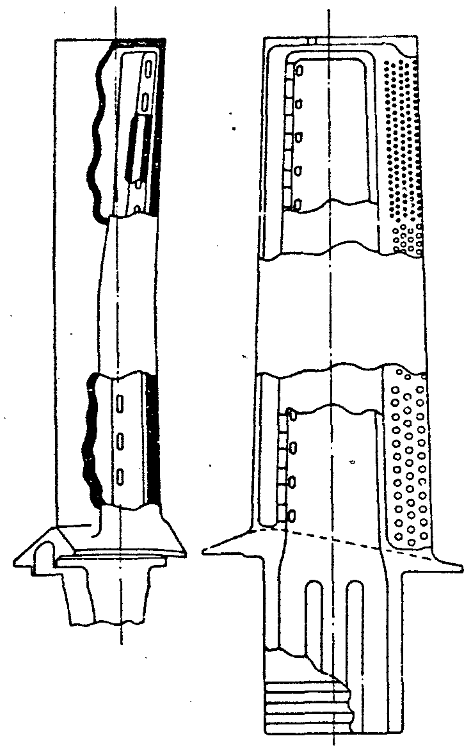
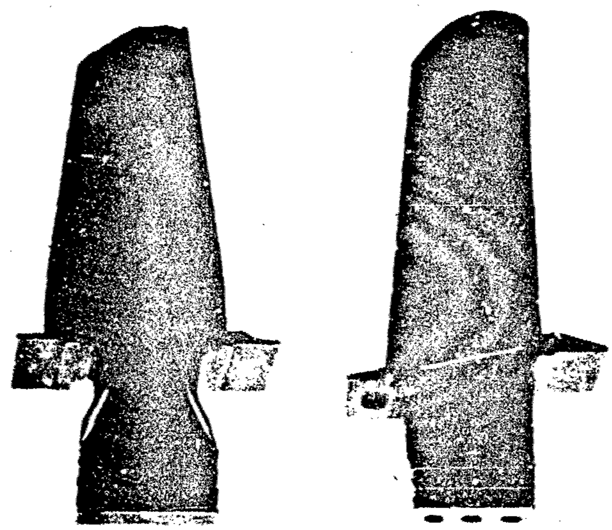


Figure 6. 1st-Stage Turbine Blade FD 14287 BIII



J58 JTF17

Figure 7. 1st-Stage Turbine Blade

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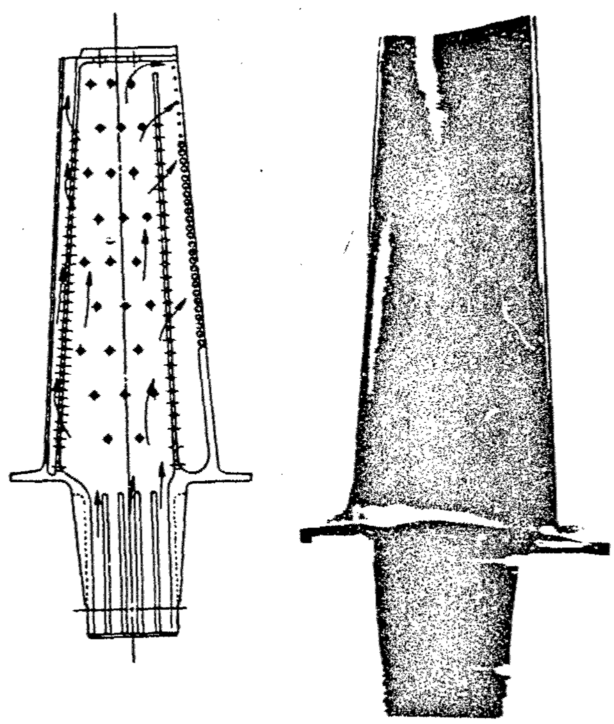

J58 Blade with
Same Cooling Scheme

Figure 8. Alternative 1st-Stage Blade
Cooling Method

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Figure 8 illustrates a slight variation of the impingement-cooled JTF17 1st-stage turbine blade design. This type of blade is currently being tested in the J58 engine test program.

Another convective-cooled blade design currently being evaluated for the J58 1st-stage turbine is shown in figure 9. This design is patterned after the 1st-stage J58 turbine vane with a separate cooling tube inserted from the blade tip, and retained by a pin. This configuration facilitates incorporation of cooling passage changes within the airfoil, and if initial J58 tests are successful, will be considered for application to the JTF17. The 2nd-stage vanes and blades for the JTF17 are shown in figures 10 and 11. These airfoils are quite similar to J58 design.

Although not illustrated, the 3rd-stage vanes are similar to the 2nd-stage vanes. The 3rd-stage blade, because it does not require cooling, is of conventional solid construction.

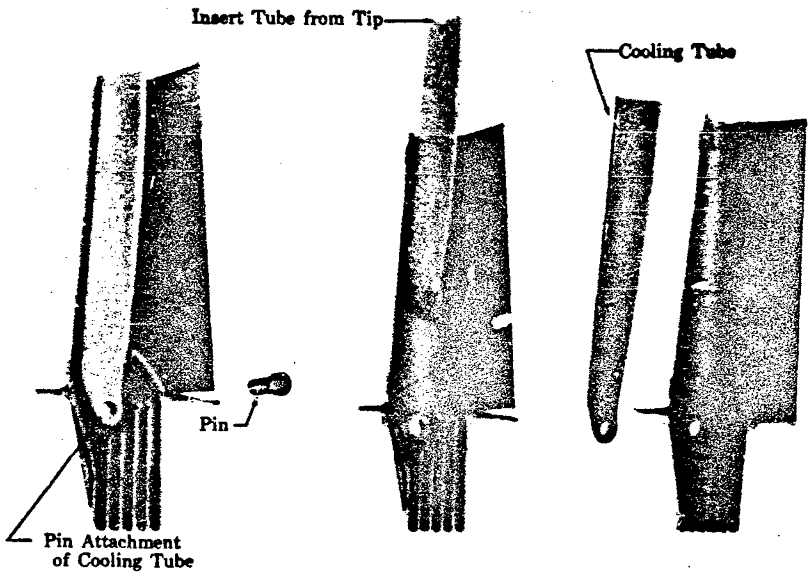

Figure 9. J58 Thermal Skin Blade Assembly FD 17095 BIII

By means of a detailed analytical model that represents the turbine including its aerodynamic performance, as well as the cooling and leakage airflows, the 1st-stage metal temperature profiles for the vanes and blades were predicted. This analysis was made for the following conditions:

1. Sea Level Takeoff, Maximum Nonaugmented
2. Mach 2.7, 65,000 ft, Maximum Nonaugmented
3. Mach 2.7, 65,000 ft, Idle
4. Mach 0.5, 5000 ft, Maximum Nonaugmented
5. Mach 0.5, 5000 ft, Idle.

The resulting temperature profiles are presented in figure 12. These profiles were confirmed in tests conducted during Phase II-C.

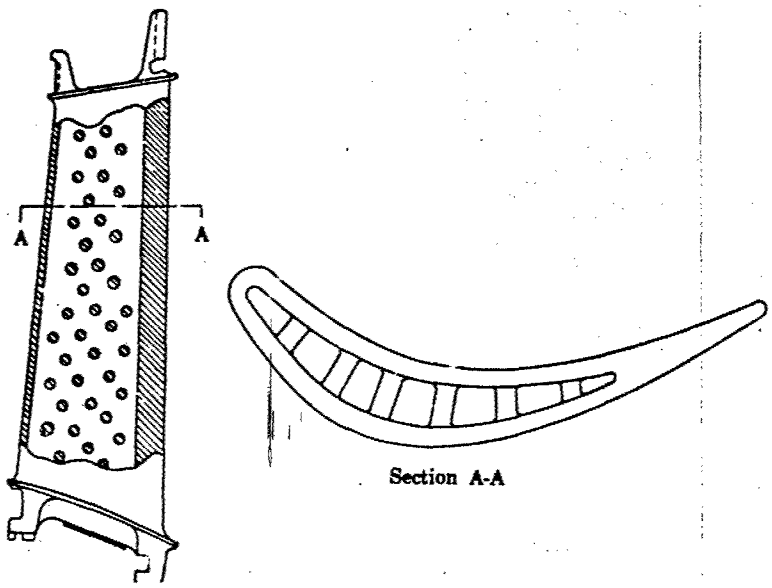

Figure 10. JTF17 2nd-Stage Vane

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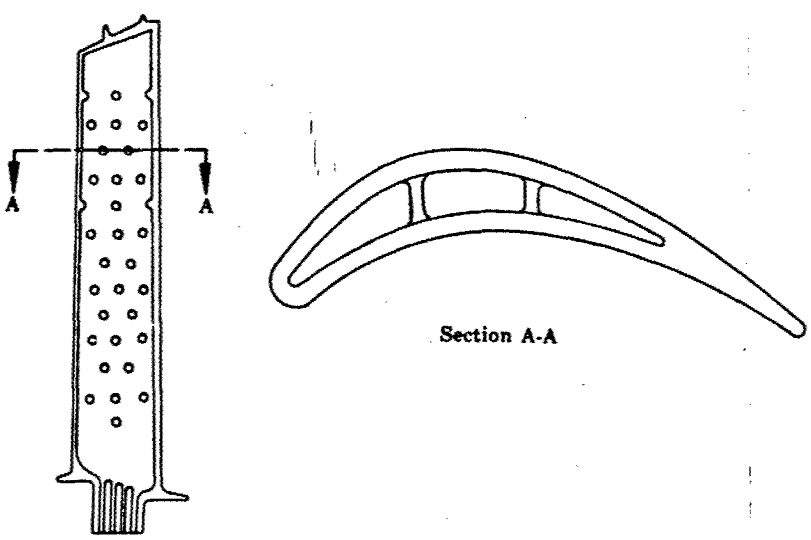

Figure 11. JTF17 2nd-Stage Blade

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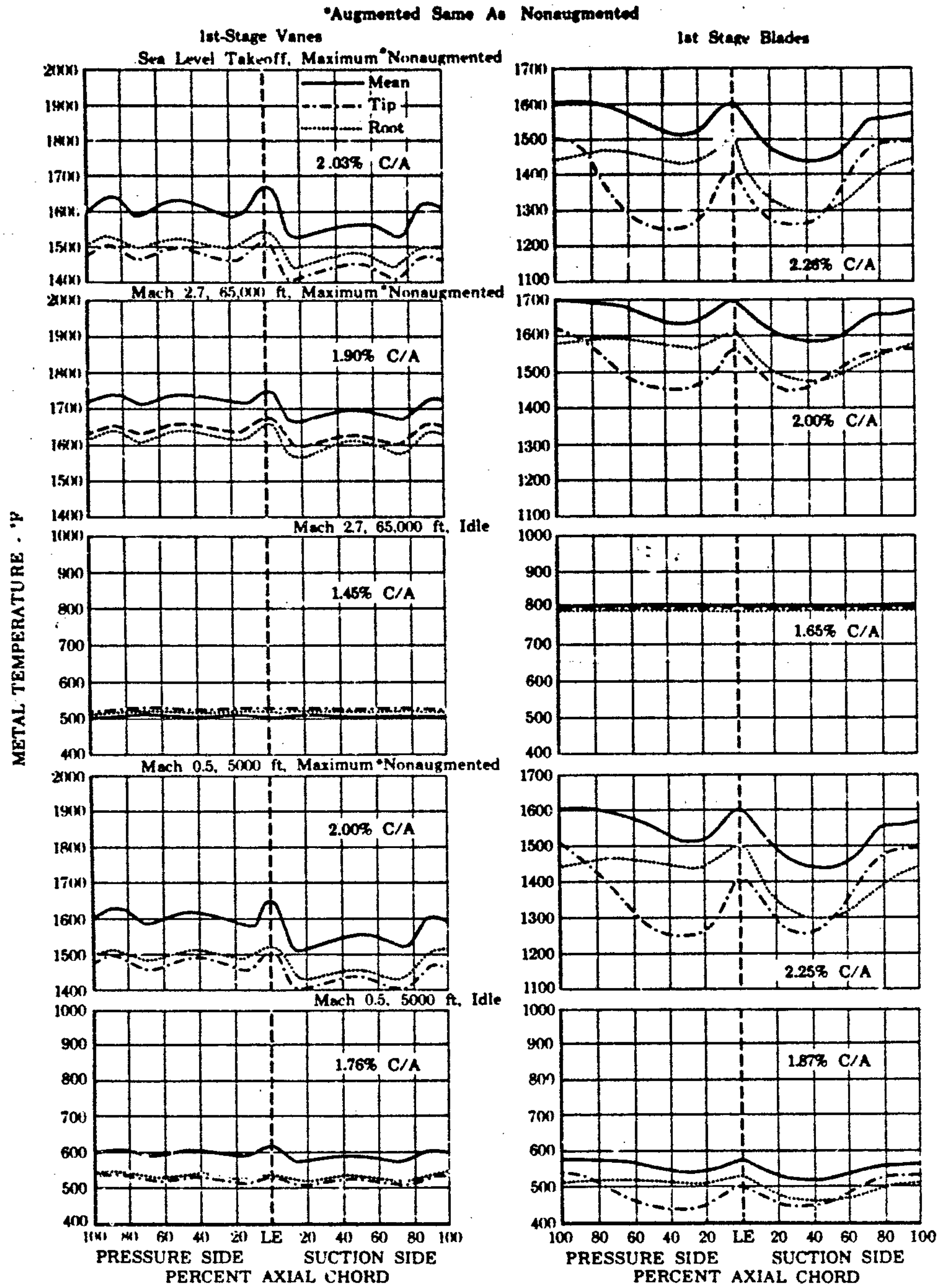

Figure 12. 1st-Stage Turbine Vane and Blade Temperatures at Various Flight Conditions

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2. Selection of Turbine Airfoil Materials

Although the selection of all of the construction materials used in the JTF17 turbine is an important part of the design, the materials and coatings used for the turbine airfoils are by far the most critical. Therefore this section deals specifically with those while other material selections, such as for turbine disks and shafts, are discussed under Turbine Mechanical Design in a following paragraph.

We design our turbine blades based on the achievement of a given life expectancy in the operating environment. Our experience indicates that the best approach to forecasting the life expectancy of a hot turbine blade is through the relationship of the 1% creep strength to the imposed centrifugal stress. For the rotating blade it has been found that gas bending, vibrating, and thermal stresses are not major factors affecting the blade life, and a prediction based on centrifugal stresses alone is adequate. Gas bending stresses can be eliminated by tilting the blade, the vibrations can be damped, and with proper cooling thermal stresses can be minimized. A life of 10,000 hours for the blades was selected as the design goal.

With the operating environment and life goal in mind, candidate materials were examined for their suitability for this application. Because the severity of the requirements make very few suitable materials available, the field was quickly narrowed to PWA 658 (IN-100), PWA 664 (Mar-M200 directionally solidified), and PWA 663 (B-1900). These three candidates were carefully considered on the basis of their mechanical properties, the amount of operating experience available, and the accumulated background on fabrication technique and process control.

On the basis of important mechanical properties such as high strength at high temperature and creep characteristics, all three are adequate to meet the life expectancy goal, with PWA 664 material showing an advantage by a wide margin. Figures 13, 14, and 15 are 1% creep property curves for PWA 658, PWA 663, and PWA 664, respectively. The curve representing the design 1st-stage blade metal temperature has been drawn on each of these curves. Also the blade stress level for blades of the same configuration for each of the three materials is shown on the respective figures. The intersection of the temperature curve with the stress level line determines the life expectancy which is seen to be 10,000 hours for PWA 658, 10,000 hours for PWA 663, and 60,000 hours for PWA 664.

While laboratory tests of mechanical properties and limited engine testing indicate an apparent superiority of PWA 664, previous experience has shown that we cannot rely on these indications alone but must confirm this choice with hundreds of hours of actual engine experience to ensure the adequacy of a critical blade material. For example, the original laboratory properties of SM 200 showed it to have superior high temperature strength compared to other turbine blade alloys, and it was the material originally used for J58 turbine airfoils. As experience accumulated, this material proved to be difficult to produce in sound castings and inconsistent in mechanical properties, and it was replaced by IN-100. We have therefore weighed heavily in our selection process

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the amount of fabrication and operating experience available on the materials. On this basis we have selected PWA 658 (IN-100) for all of the turbine blades. To our knowledge, PWA 658 is the only turbine blade alloy that has been produced in sizable quantities and has demonstrated reliability in operational experience at 2000°F turbine inlet temperature.

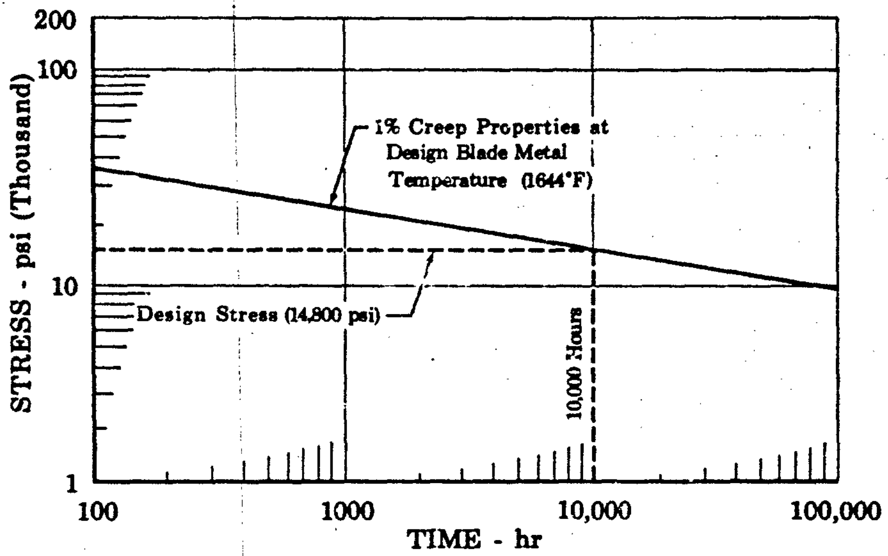

Figure 13. PWA 658 (IN-100), 1% Creep FD 17800 BIII

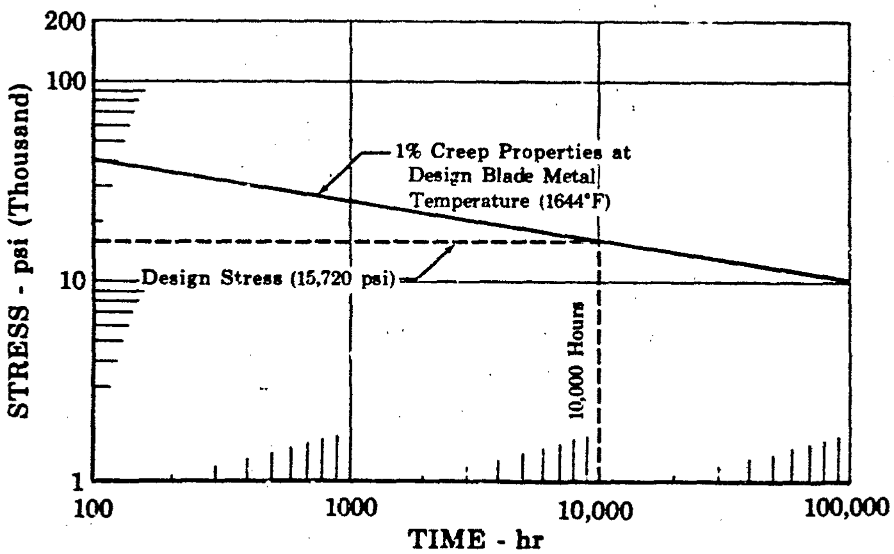

Figure 14. PWA 663 (B-1900), 1% Creep FD 17748 BIII

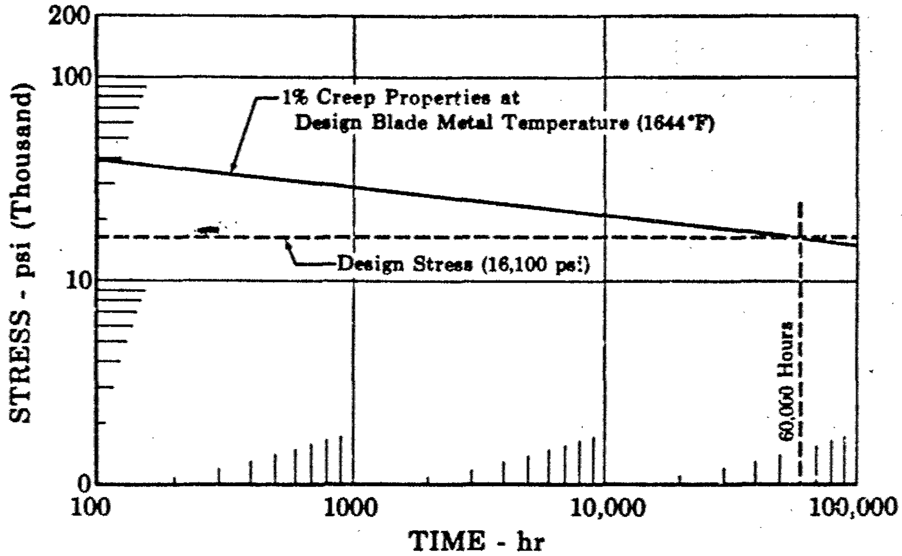

Figure 15. PWA 664 (Mar-M200), 1% Creep FD 17750 BIII

While both PWA 663 and PWA 664 have been subjected to extensive laboratory research work, they have only 500 to 600 hours of engine operating time, making them a less attractive choice than PWA 658 at this time. Of the two, PWA 664 represents the greatest future potential, since figure 15 indicates that it may have a potential blade life approximately six times that of PWA 658. It may also be possible that a higher blade metal temperature can be tolerated by this material, thereby lowering cooling air requirements and increasing turbine performance. We plan to continue testing PWA 664 because of this durability potential, and it can be incorporated into the JTF17 engine if the results of this testing indicate that this is desirable.

Considerably further in the future, but showing great potential is PWA 1409 (Mar-M200 single crystal). This MONOCRYSTALOY* has mechanical properties that exceed those of PWA 664 by a wide margin except in the direction transverse to the crystal axis where they are equal. These transverse properties are not believed to be important in turbine blades. Blades and vanes of this material have been fabricated and engine tested at East Hartford and blades have been made for the J58 engine for testing at FRDC.

The turbine vanes, although operating under different conditions of stress and temperature than the blades are no less difficult a problem. Since the vanes do not rotate they do not benefit from temperature "averaging" by moving through circumferential hot spots as the blades do. This means that the vanes may be subjected to local temperatures that exceed the average gas temperature and thus need to be able to withstand these higher temperatures. We have selected PWA 1035 (TD Nickel) for the 1st-stage vanes because of its high strength at high temperature. Unlike other high temperature materials, PWA 1035 does not depend on heat treatment to produce the dispersed phase strengthening; and the dispersed thoria particles are essentially insoluble in nickel at temperatures up to and including the melting point, 2650°F. Temperature excursion beyond

*UAC Trademark

the design temperature of 1700°F will not significantly reduce its mechanical properties. Available data indicate that exposure to a temperature of 2400°F for one hour lowers the room temperature tensile strength only slightly. At higher testing temperatures, the effect is even less significant. Other properties such as hardness, average thoria particle size, and average nickel grain size are essentially unchanged by the above thermal exposure. A stress-rupture curve for PWA 1035 is given in figure 16 and indicates the high strength at high temperature feature of this material.

The 2nd- and 3rd-stage vanes operate in a less severe environment and PWA 658 was selected for these, again on the basis of our J58 experience with this metal. It also has the lowest density of any of the readily available high temperature alloys that meet the strength requirements. The stress-rupture characteristics of PWA 658 are given in figure 17.

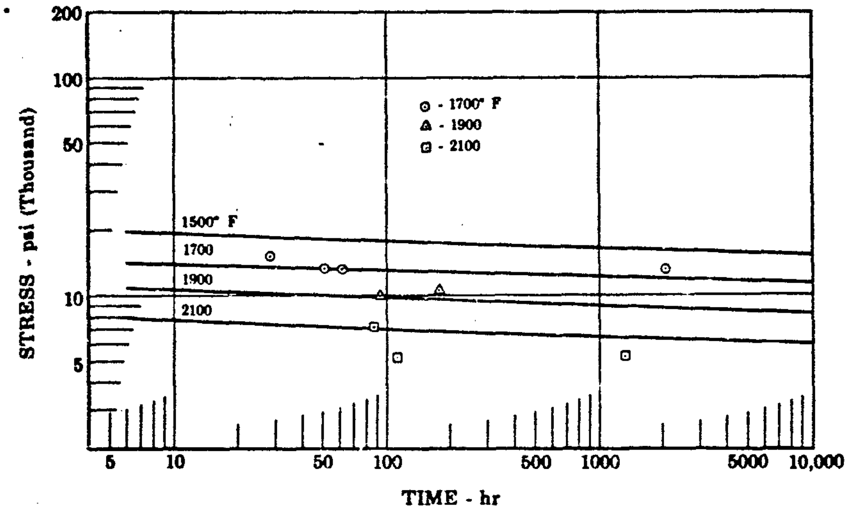

Figure 16. TD Nickel (PWA 1035) Stress Rupture Data FD 15596B BIII

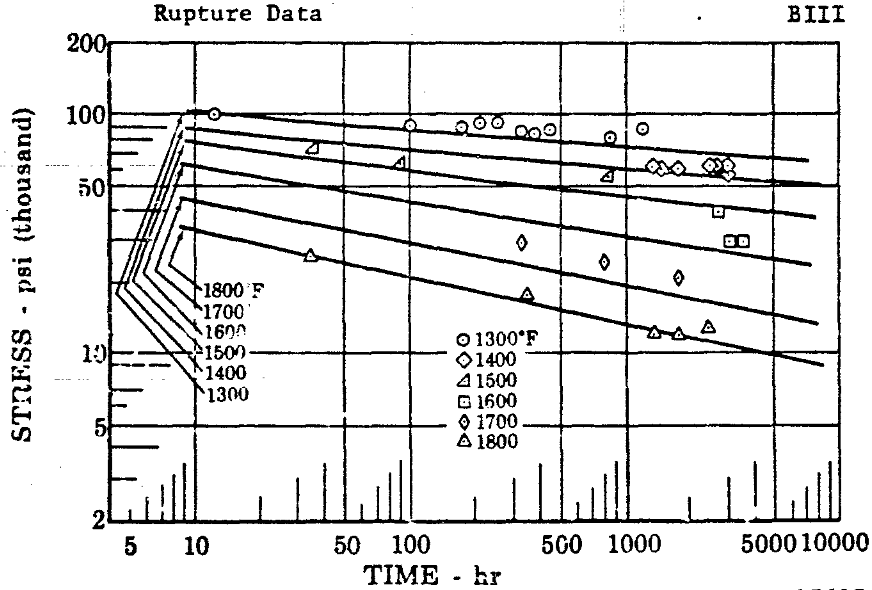

Figure 17. PWA 658 (IN-100) Stress Rupture FD 17605 BIII

The materials, PWA 658 and PWA 1035, cannot survive long in the actual engine environment unless protected from attack of oxygen and sulfur by the use of special coatings. A detailed discussion of the selected protective coatings is given in Volume III, Report F, Manufacturing Techniques and Materials. Briefly, however, we have selected PWA 62 coating for the 1st-stage vanes. This coating is a complex ternary composition consisting of nickel-aluminum-chromium. It has a melting point greater than 2400°F, and has shown no distress in accelerated sulfidation testing at 1800°F (100° higher than average vane metal temperature) for over 600 hours. This testing is still in progress and is expected to show a considerably longer life. It should be pointed out that the coating is easily stripped from the vanes and recoating can be readily accomplished.

All of the other turbine airfoils are coated with PWA 64, a chromium-aluminum modified aluminide. This coating, like PWA 62, has not been completely tested and the work is currently in progress. We already have, however, 800 hours of successful oxidation life testing, 900 hours of erosion life testing, and 1200 hours of sulfidation life testing. The hardware may also be stripped and recoated as necessary to improve serviceability.

Table 2 is provided as a convenient summary of all of the important materials used in the turbine. The discussion and justification of our selections of materials for the rotor, case and shroud seals, and the exhaust are presented later.

Table 2. Turbine Material Summary

Airfoil	Material	Coating
1st-Stage Vane		
Airfoil	PWA 1035 (TD Nickel)	PWA 62
Platform	PWA 658 (IN-100)	
2nd-Stage Vane	PWA 658 (IN-100)	PWA 64
3rd-Stage Vane	PWA 658 (IN-100)	PWA 64
1st-Stage Blade	PWA 658 (IN-100)	PWA 64
2nd-Stage Blade	PWA 658 (IN-100)	PWA 64
3rd-Stage Blade	PWA 658 (IN-100)	PWA 64
Rotor		
Disks	PWA 1013 (Astroloy)	
Spacers	PWA 1007 (Waspaloy)	
Low Turbine Shaft	PWA 1016 (Waspaloy)	
Knife Edge Seals	PWA 1007 (Waspaloy)	
Case and Shroud Seals		
Case	AMS 5707 (Waspaloy)	
Shroud Seals	AMS 5754 (Hastelloy X)	
Exhaust		
Case	AMS 5707 (Waspaloy)	
Guide Vanes	PWA 655 (Inco 713)	
Temperature Probe	AMS 5536 (Hastelloy X)	

Figure 18 presents a chart of predicted airfoil life versus metal temperature for each airfoil and illustrates which life limitation will first be encountered. The low cycle fatigue life of every airfoil exceeds the 10,000 cycles design requirement. This design requirement is equivalent to 10,000 hours of airfoil life as stated in section I of this report.

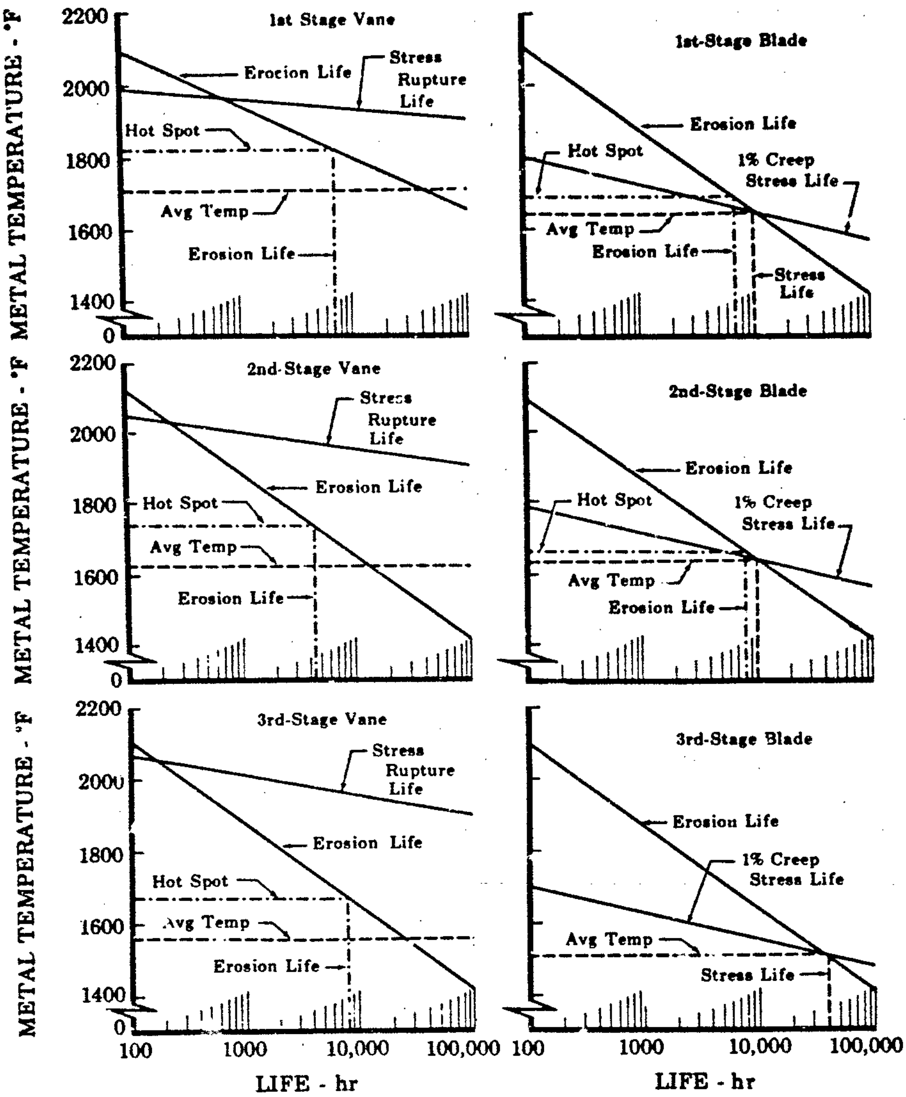

Figure 18. Turbine Airfoil Life Summary

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3. Aerodynamic Design

The overall thermodynamic performance of the turbine, or its net efficiency, is determined not only by its aerodynamic performance, but also by cooling and leakage losses. Airfoil cooling, and the associated losses have been previously described. In the following discussion the aerodynamic design of the turbine is first presented. This is followed by a description of the means provided to minimize turbine tip-leakage.

The turbine aerodynamic design requirements of inlet temperature, pressure, airflow, rpm, specific work, and efficiency were established from the engine optimization studies performed in Phase II-B and Phase II-C. During the studies, tradeoffs between turbine weight, turbine efficiency, and turbine diameter were evaluated. Influence factors for each item were obtained for the various typical flight conditions. The results showed that turbine diameter and weight more strongly affected overall engine performance than turbine efficiency. Consequently, the turbine has been designed to meet the cycle requirement for efficiency in a lightweight, compact package. Figure 19 shows the turbine elevation with number of airfoils per row, root and tip diameters, mean chord, and materials. This configuration combines advanced convective cooling, refined aerodynamic design (controlled-vortex), and minimum blade tip clearance.

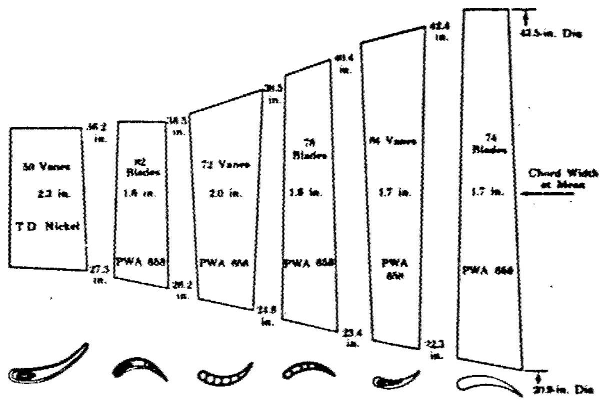

Figure 19. Turbine Configuration Schematic FD 16223 BIII

The conventional free-vortex method of turbine design has been in use for many years at Pratt & Whitney Aircraft and elsewhere. Turbines designed by this method provide high efficiency at low work coefficient ($2gJ\Delta h/U^2$) and at low ratios of axial velocity to wheel speed (C_X/u). This is illustrated by the parametric lines of constant efficiency in figure 19A. High efficiency free-vortex turbines, however, are relatively heavy, and require long highly stressed blades.

The analytical prediction of free-vortex turbine performance is well established, and accurate, having been verified by tests of a large variety of turbines over a wide range of operating parameters. This is also illustrated in figure 20A which shows analytical predictions of free-vortex turbine performance as lines of constant efficiency, and the measured

efficiencies of a large number of P&WA engine turbines. The agreement between analytical predictions and experimental results is excellent, being on the order of within 0.5%.

In the controlled vortex design, turbine efficiency is increased by modifying the airfoil spanwise aerodynamic loading distribution to optimize the performance at each radial station. This design results in a smaller diameter, lighter turbine which will meet the work and efficiency requirements of 86.9% for the high spool and 88.0% for the low spool.

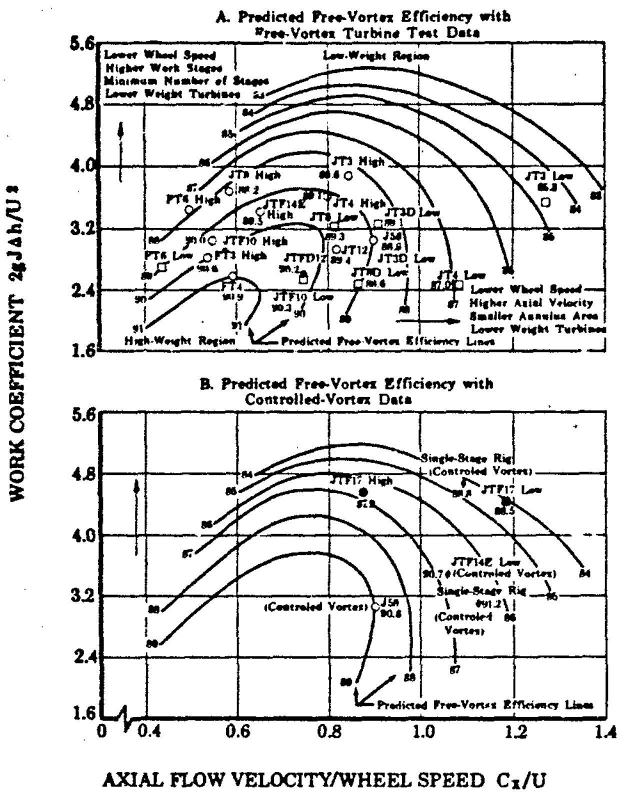

Figure 20. Comparison of Predicted Free-Vortex Efficiency With Free-Vortex and Controlled-Vortex Turbine Test Data

FD 16878
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Measured efficiencies of controlled vortex turbines are compared with the free-vortex analytical predictions in figure 20B, and are seen to be considerably higher than the free-vortex predictions. Further evidence of the potential of controlled-vortex turbines is given in figure 21, which shows data points for the JTF14 engine with free-vortex and with controlled-vortex low pressure turbines. In these tests a gain of more than 2% in low pressure turbine efficiency was measured for the controlled vortex design.

The performance of JTF17 turbines and the J58 controlled-vortex turbines are also shown in figure 20B. The efficiencies reflect the gains from controlled-vortex design, and in the case of the JTF17, reduced turbine tip clearance as well. All the efficiencies (including

the JTF17) given on these two curves are for uncooled turbines. Efficiency losses of 1% in the high pressure turbine and 0.5% in the low pressure turbine are expected because of cooling requirements. The necessary turbine efficiencies will be met even with these losses.

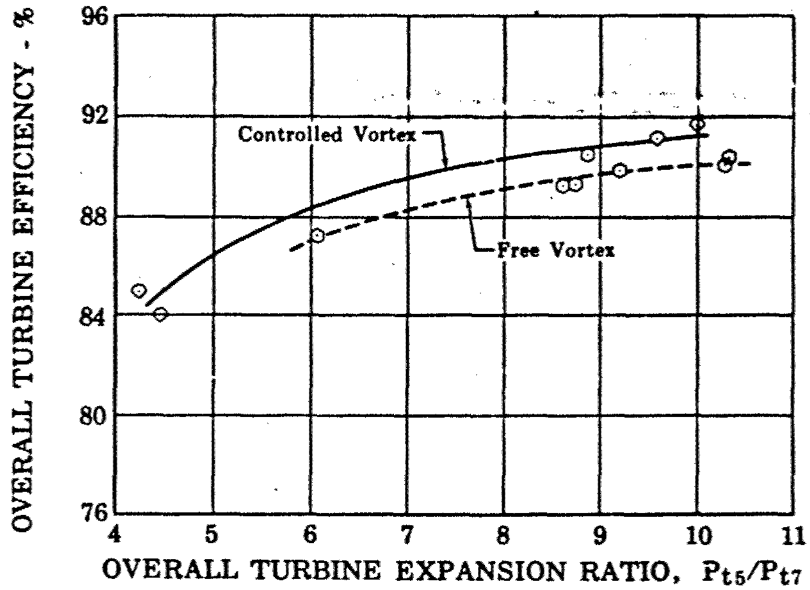

Figure 21. Controlled and Free-Vortex Turbine Test Results FD 16370 BIII

The analytical model for controlled vortex turbine design uses data from the axial flow compressor calculations used in compressor design to solve for radial continuity. This model gives the detailed gas velocity triangles for which the turbine airfoils must be designed. The midspan velocity triangles for the JTF17 turbine at cruise are shown in figure 22.

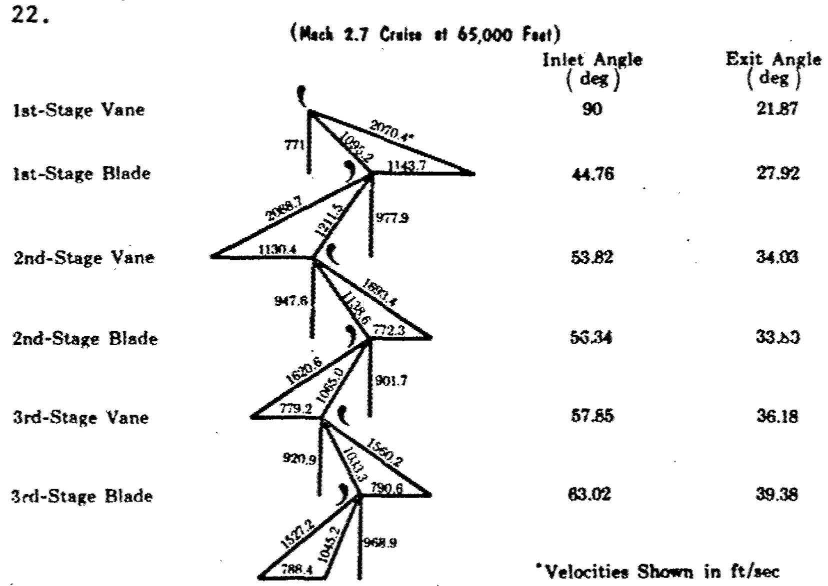

Figure 22. JTF17 Turbine Design Velocity Diagrams FD 16220 BIII

Table 3 lists detailed results of the controlled vortexing. Gas inlet and exit angles and Mach numbers for each blade or vane are listed. The work, velocity ratio, and predicted cooled efficiency of each stage are also given.

Table 3. Controlled-Vortex Results at 65,000 ft - M 2.7 Turbine Design Point

	Percent Reaction	Inlet Angle, deg	Exit Angle, deg	Inlet Mach No.	Exit Mach No.	Absolute Exit Axial Mach No.	Stage Work Btu/lb	Stage Mean Velocity Ratio	Stage Cooled Efficiency, %
1st-Stage Vane									
Root		90.0	26.12	0.24	1.07	0.47			
Mean		90.0	21.87	0.22	0.94	0.35			
Tip		90.0	17.99	0.21	0.84	0.26			
							112.38	0.479	86.9
1st-Stage Blade									
Root	27	40.54	30.58	0.72	0.95	0.48			
Mean	39	44.76	27.92	0.51	0.92	0.45			
Tip	51	47.51	27.82	0.35	0.94	0.44			
2nd-Stage Vane									
Root		50.0	35.5	0.62	0.90	0.52			
Mean		53.8	34.0	0.52	0.79	0.45			
Tip		61.0	34.2	0.48	0.72	0.38			
							61.87	0.445	86.5
2nd-Stage Blade									
Root	43	46.0	35.4	0.67	0.89	0.52			
Mean	49	56.3	33.8	0.48	0.80	0.44			
Tip	56	71.8	31.4	0.37	0.75	0.39			
3rd-Stage Vane									
Root		49.9	39.8	0.72	0.95	0.60			
Mean		57.8	36.2	0.49	0.76	0.44			
Tip		63.1	35.5	0.37	0.63	0.36			
							51.85	0.491	88.6
3rd-Stage Blade									
Root	44	50.9	41.8	0.77	0.93	0.62			
Mean	50	63.0	39.4	0.47	0.77	0.48			
Tip	57	91.8	34.9	0.33	0.72	0.41			
Exit Guide Vane									
Root		55.4	90.0	0.79	0.56	0.56			
Mean		66.7	90.0	0.50	0.45	0.45			
Tip		81.4	90.0	0.42	0.49	0.49			

Table 4 contains the results of the off-design predictions for operating gas temperatures and pressures at three maximum nonaugmented power settings.

Table 4. Turbine Stage Conditions at Three Nonaugmented Flight Conditions

Flight Condition		1st-Stage Inlet	2nd-Stage Inlet	3rd-Stage Inlet	Turbine Exit
Sea level static maximum non-augmented	Total pressure, psia	179.71	83.03	50.28	32.27
	Total temperature, °R	2760.0	2307.9	2062.6	1911.1
36,000 ft Mach 0.9 maximum nonaugmented	Total pressure, psia	74.99	35.04	21.77	14.99
	Total temperature, °R	2705.8	2312.4	2078.6	1956.6
65,000 ft Mach 2.7 maximum nonaugmented	Total pressure, psia	69.71	31.80	19.35	12.48
	Total temperature, °R	2660.0	2233.6	2006.7	1857

The controlled vortex method of turbine design maximizes the reaction at the aerodynamically critical spanwise sections, the root and the tip. The reaction along the span of each blade is compared with a standard free-vortex design in figure 23. Because reaction blading is more efficient than impulse blading, this maximizing of reaction at the blade root and vane tip reduces the tendency for flow to separate at these critical sections and increases overall blading efficiency. It also provides a lower blade tip static pressure ratio that reduces tip leakage.

Tip Leakage: Tip leakage losses in a high inlet temperature turbine operating over a wide temperature range, such as the JTF17, will be large if a conventional stationary-shroud, tip-seal design is employed. However, as shown in figure 24, a gain of over three percent in efficiency of the high pressure turbine can be achieved if tip clearance is reduced from the 0.082 inch required by a conventional stationary-shroud design to 0.020 inch. The "thermal-response" stationary-shroud design utilized in the JTF17 turbine allows operation at this low tip clearance of 0.020 inch with a corresponding 3+% improvement in high pressure stage efficiency. Also, similar, though smaller, gains are made in the low pressure turbine through incorporation of this tip-seal design.

The minimum allowable turbine tip clearance is determined by the worse transient condition of rapid engine deceleration from takeoff power to full shutdown. As shown in figure 25, during steady-state takeoff power operation, the temperature of a conventional stationary-shroud is considerably higher than the temperature of the turbine disk. However, as turbine inlet temperature drops rapidly during the deceleration, the low-mass shroud cools more rapidly than the heavier disk. The faster thermal response of the stationary-shroud results in more rapid shrinkage in its diameter than the disk causing a maximum reduction in tip clearance during the transient, as shown in figure 25. The minimum allowable clearance occurs at approximately 30 seconds from the start of the deceleration. This requires operation of clearances at cruise and sea level takeoff of 0.082 inch and 0.100 inch, respectively, with a serious efficiency penalty.

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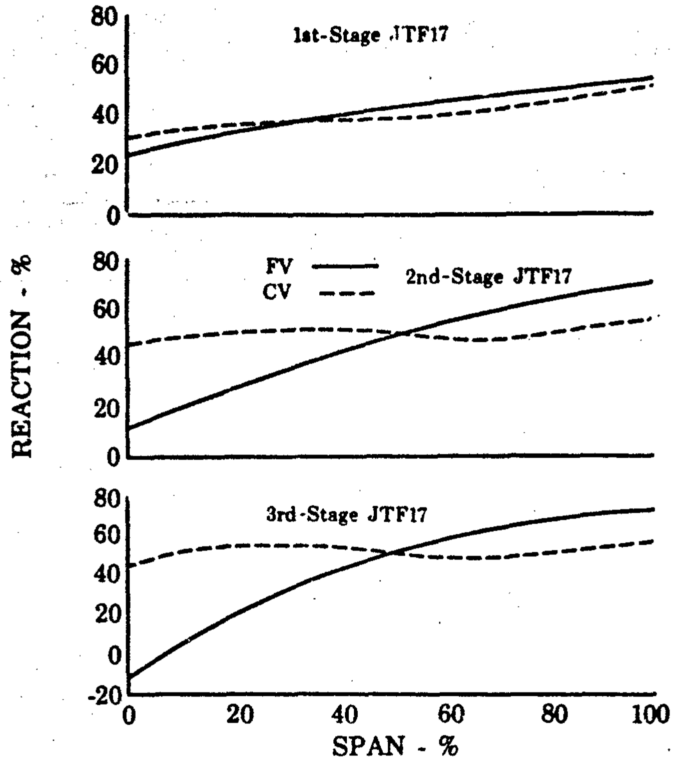

Figure 23. Comparison of Controlled Vortex and Free-Vortex Turbine Reaction FD 16885 BIII

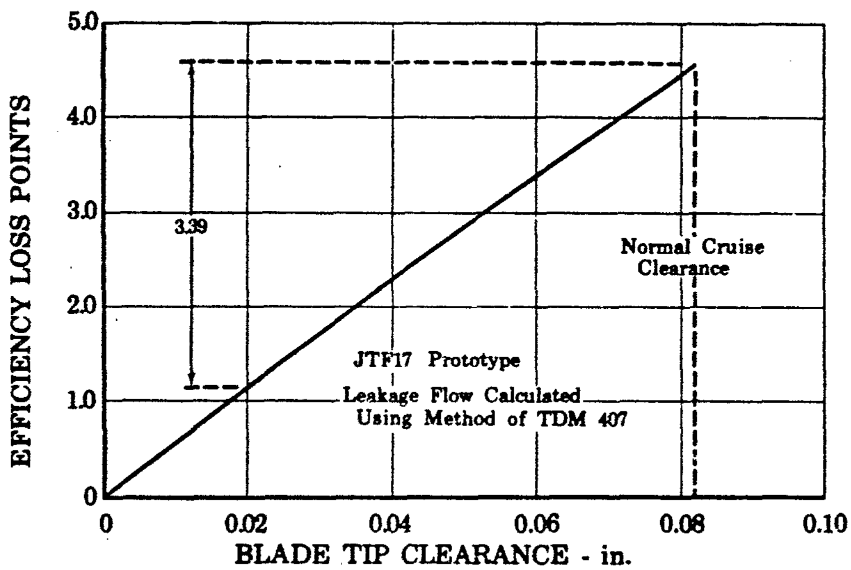

Figure 24. Effect of Blade Tip Clearance on JTF17 1st-Stage Turbine Efficiency FD 16216 BIII

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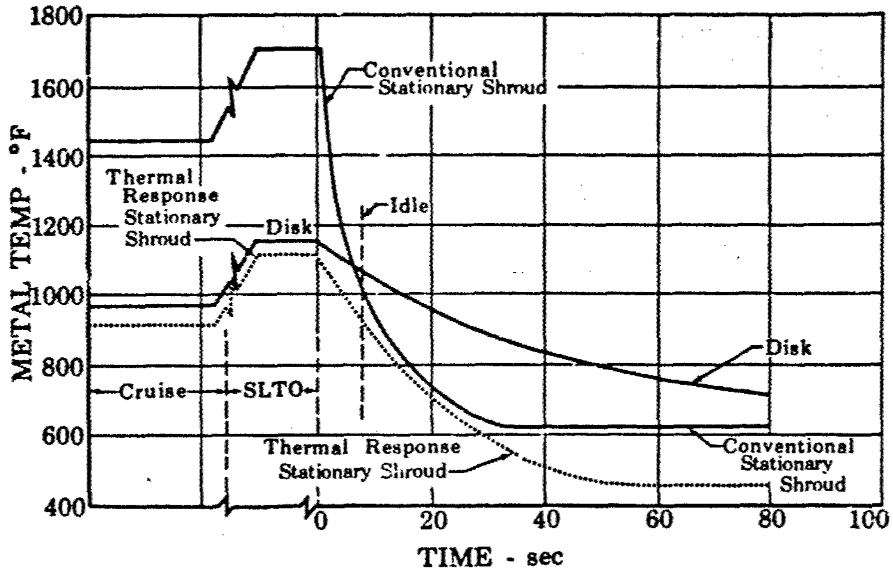

Figure 25. Tip Clearance Control for High Temperature Turbines FD 17601 BIII

If the stationary-shroud is designed to operate at lower peak temperatures, and its thermal response is matched to that of the disk, steady-state running clearance can be reduced to 0.020 inch with a significant improvement in turbine efficiency. This is illustrated by the curves for the "thermal-response shroud" design in figures 25 and 26.

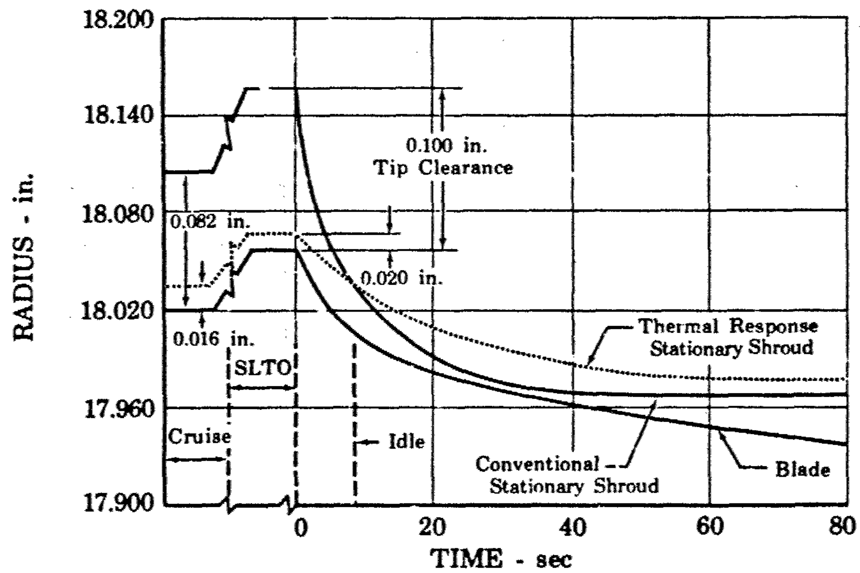

Figure 26. Tip Clearance Control for High Temperature Turbines FD 17602 BIII

The controlled thermal-response stationary-shroud design is shown in figure 27. The maximum steady-state operating temperature is held to approximately 1150°F, slightly lower than the disk temperature, by internal cooling using 0.5% of the gas generator flow. This is no greater than the flow required to "film-cool" the back face of a conventional stationary-shroud; however, using it more effectively results in a lower shroud operating temperature.

The transient thermal-response of the stationary-shroud is determined by its mass, which is substantially larger than that of a conventional stationary-shroud. The thermal-response of the "controlled" shroud design is lower than that of the disk. This provides automatic control of close clearances, and results in high efficiency and a degree of reliability unattainable with other methods of clearance control.

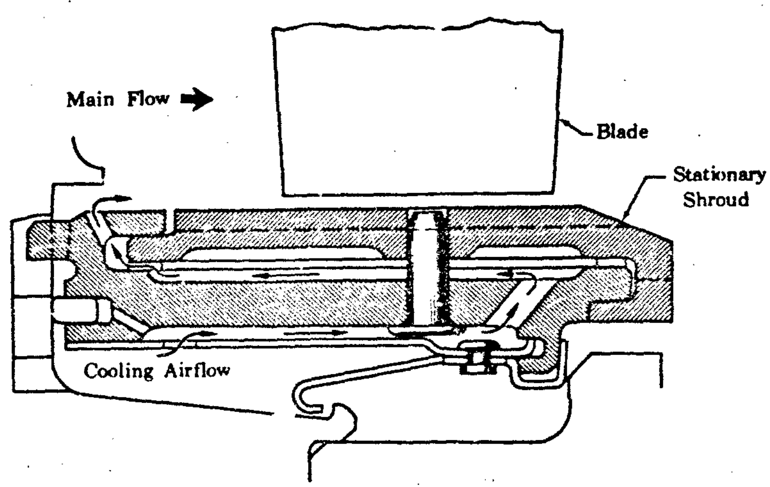

Figure 27. Controlled Thermal-Response Stationary-Shroud for Turbine Blade Tip Clearance Control

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4. Mechanical Design

The JTF17 turbine is a three-stage axial flow reaction type. The first stage drives the high compressor and the second and third stages drive the fan. The general mechanical arrangement is illustrated in figure 28. The major components of the turbine section are the rotors, which consist of blades, disks, and shafts; and the stationary parts, consisting of the turbine case, vanes, tip-seals (stationary shrouds) and turbine exhaust. Because of their major importance, materials selection, and the cooling techniques of the airfoils have been described separately, and this is not repeated in the following description of the mechanical design of the turbine. Cooling is covered (where appropriate) in the description of the mechanical design of the other turbine components.

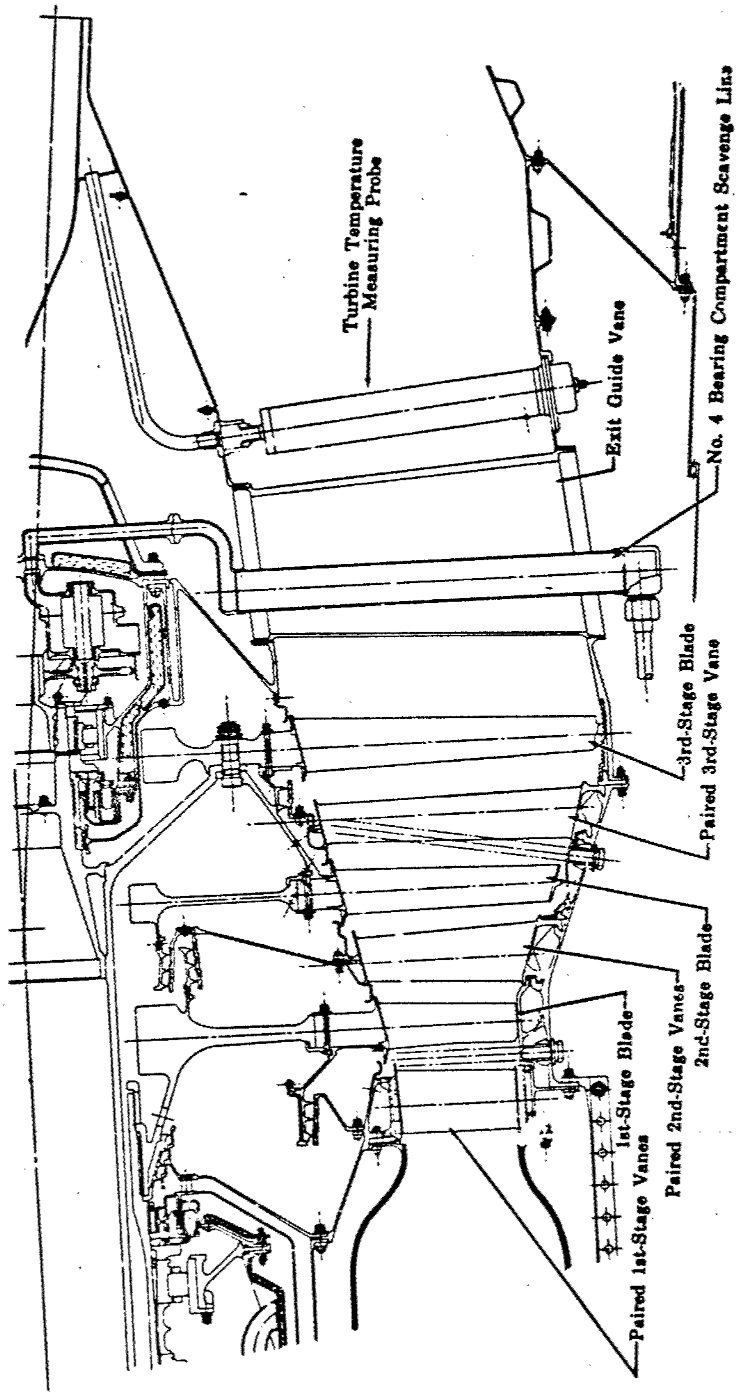

Figure 28. Turbine Mechanical Arrangement

a. Rotor

(1) Blade Attachment

The turbine blades are retained in the disks by a bolted fir tree design. Figure 28 shows the attachment method for the 1st-stage blades. The bolts that provide blade retention in the axial direction also secure the coverplate to the disk. The 1st-stage blades may be removed from the aft side of the disk (after low turbine balanced unit removal) without removal of the disk from the high compressor shaft. All three rotor stages incorporate blades that are moment-weight classified. Blades of equal moment-weight are installed 180 degrees apart to permit replacement of blades in pairs without subsequent rebalancing.

The disk and blade attachment point is cooled by the flow of blade cooling air through the manifold immediately under each blade fir tree, as shown in figure 29. The portion of the blades inside the platform is protected from excessive radiant heating by heatshields fore and aft just inside the platform, and is protected from convective heating by the coverplate on the front (first stage), which seals against hot air leakage across the disk. The coverplate on the front of the 1st-stage disk also provides a blade vibration damping system proved effective in the J58 engine.

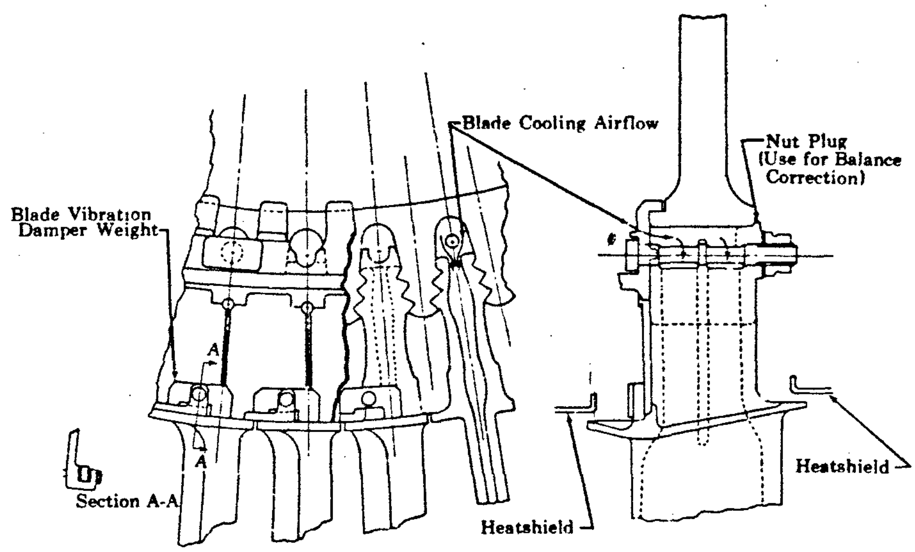

Figure 29. 1st-Stage Turbine Blade Attachment FD 16221
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Table 5 lists the stress ratios for the blade and the disk attachments. The stress ratio is the ratio of the applied stress to the allowable material strength at the operating temperature. Also listed on table 5 are the stress ratios that are considered allowable. These allowable ratios are consistent with those in use in current P&WA commercial engines.

Table 5. Turbine Blade and Disk Attachment Stress Ratios

Blade Attachments	Allowable Ratio	Stage 1	Stage 2	Stage 3
1st-neck tensile	0.50	0.50	0.50	0.30
2nd-neck tensile	0.50	0.50	0.50	0.22
3rd-neck tensile	0.50	0.50	0.50	0.13
Tooth bending	0.50	0.23	0.27	0.31
Tooth shear	0.30	0.25	0.22	0.19
Tooth bearing	0.80	0.68	0.55	0.72
Disk Attachments				
1st-neck tensile	0.50	0.50	0.42	0.46
2nd-neck tensile	0.50	0.50	0.37	0.43
3rd-neck tensile	0.50	0.49	0.29	0.33
Tooth bending	0.50	0.22	0.30	0.35
Tooth shear	0.30	0.24	0.24	0.21
Tooth bearing	0.80	0.73	0.57	0.75

(2) Vibration

Turbine blade vibration results from two principal sources of excitation: turbulent flow buffeting which produces broad-band excitation, and integral order excitation which is a function of speed.

Figures 30, 31, and 32 are resonant diagrams of the first, second, and third stages. In the diagram, the lines of constant excitation (E) show the frequency response for integral order excitation as a function of speed. The blade/disk combination has a natural frequency, at which it is easily excited, which changes with rotor speed as shown in the figures. The natural frequency of the blade/disk combinations have been designed to avoid any integral order frequencies within the normal operating range. This was accomplished by the selection of proper disk thickness and cone attachment points.

The integral orders of excitation due to inlet distortion are of low frequency, as described in Volume III, Report B, Section II. Another source of integral order excitation in turbines of previous engines was the separate multiple burner cans; however, this source is eliminated by the use of the annular ram-induction combustor in the JTF17.

Another potential source of turbine vibration excitation has been avoided by locating the turbine exit guide vanes well rearward. The 16 vanes produce a 16E excitation that is suppressed within four airfoil thicknesses when propagating upstream.

Since the 1st-stage blades are not shrouded, vibration damping is accomplished with a system of toggle weights similar to those successfully used in the J58 turbine. The toggle weight configuration is illustrated in figure 28. This toggle weight design has been found to exhibit low wear and long life. Also, the absence of tip shrouds reduces airfoil centrifugal stress and pull on the attachment.

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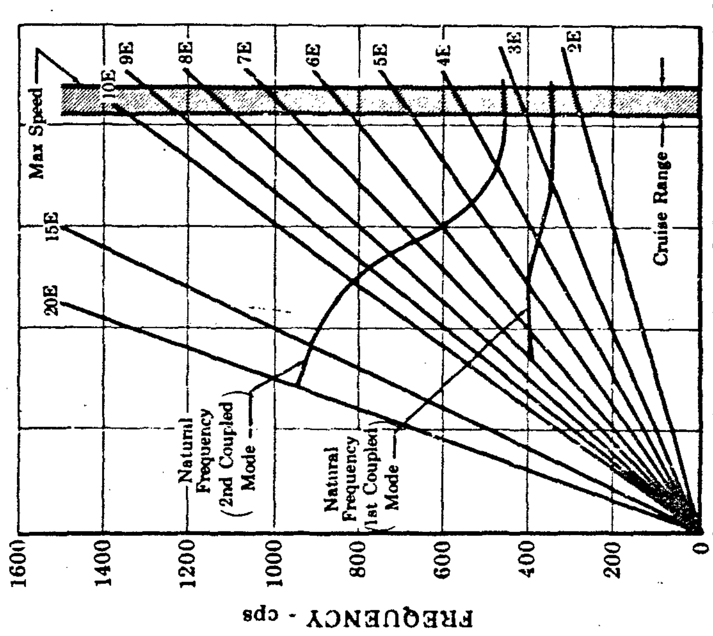

Figure 30. JTF17 1st-Stage Turbine Resonant Speeds FD 16235 BIII

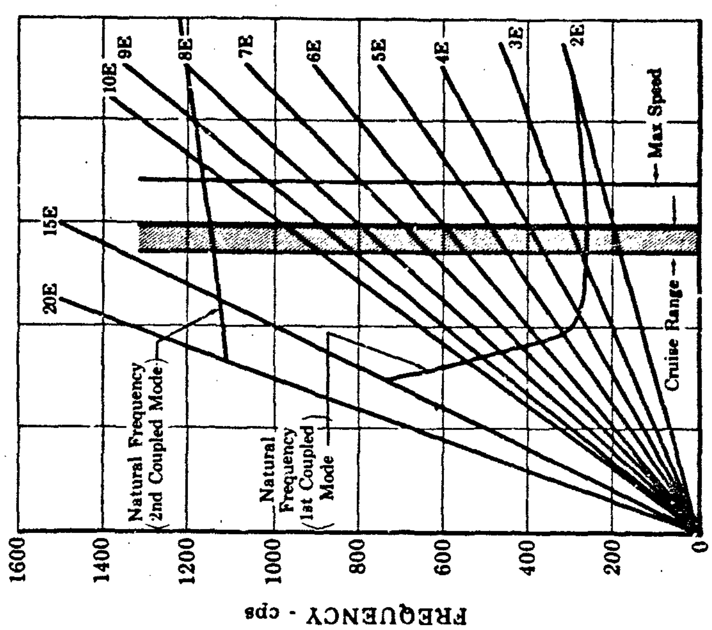

Figure 31. JTF17 2nd-Stage Turbine Resonant Speeds FD 16236 BIII

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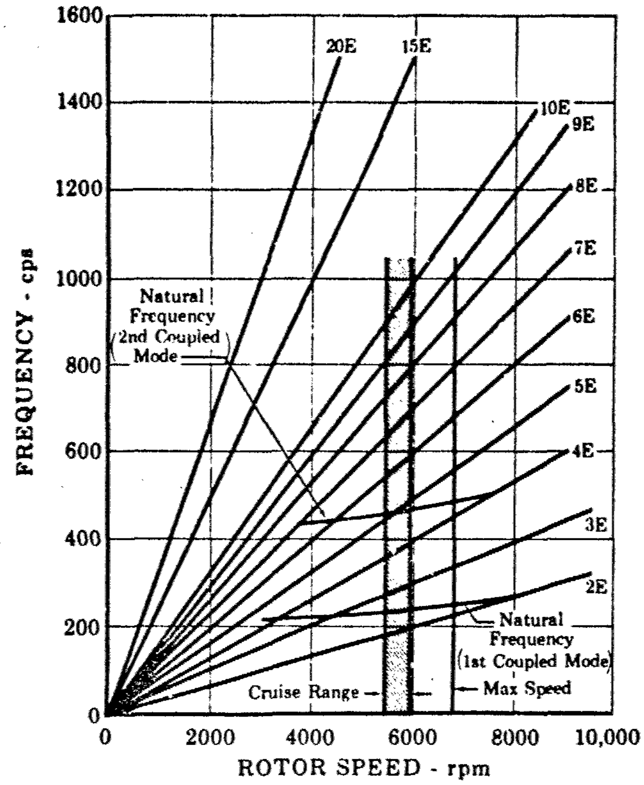

Figure 32. JTF17 3rd-Stage Turbine Resonant Speeds

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The 2nd- and 3rd-stage turbine blades use conventional tip shrouds for vibration damping. The damping effectiveness of the shroud is a function of several parameters, the most significant being the contact (notch) angle between adjacent shroud rubbing surfaces. As the notch angle is increased (contact surfaces approach an axial plane) the damping effectiveness diminishes because the contact surfaces become more nearly normal to the direction of vibration. The highest damping occurs when the contact surfaces are parallel to the direction of vibration. Because the friction between adjacent blades causes the damping, shroud wear is necessarily highest at low notch angles. Therefore, for the long life blades of the JTF17 engine, a compromise was made to increase the shroud notch angle to reduce shroud wear. The airfoils are designed to withstand the corresponding increase in vibratory stress. An improved hardfacing material (PWA 65) that is flame-deposited is used to reduce wear. This material can be stripped and the blade re-hardfaced as required.

Design studies have been made to apply a toggle weight damping system to the second stage. This wear-free system can be substituted for shrouds when further verified by test. However, since this design requires an extended neck of 20% of the blade length, it cannot be employed in the third stage for lack of available space. Other similar methods are being studied.

(3) Disks

Each of the three disks is designed for minimum weight consistent with reliability and life requirements. The first disk has an integral arm at the bore which eliminates the need of a separate shaft to secure it to the high pressure compressor shaft. This design has several advantages:

1. Better alignment due to a reduction in number of joints, which also improves balancing and maintenance
2. Better vibration characteristics
3. Avoidance of holes in high stress regions.

The 2nd-stage disk also avoids holes in high stress regions. It is attached to the low pressure rotor shaft by an integral arm rear cover-plate. The 3rd-stage disk, which is cooler than the other two, contains a conventional circle of bolts that secure it directly to the integral hub of the low rotor shaft. The bolt circle that secures the 3rd-stage disk to the low rotor shaft contains one offset hole to ensure proper reassembly orientation and to eliminate the need for rotor rebalance after replacement of this rotor stage.

The disks are cooled by compressor discharge air to maintain a low and uniform disk metal temperature. This improves low cycle fatigue life by minimizing disk temperature gradients due to gas stream temperature variations during engine transients.

Disk burst margin is defined as the predicted speed at which the disk will burst divided by the maximum operating speed. For the JTF17 engine, a burst margin of 1.3 was selected based on previous P&WA commercial engine experience. Pertinent disk stress data are shown in table 6.

Table 6. Turbine Disk Stress Data

	Stage 1	Stage 2	Stage 3
Bore tangential stress, psi	90,400	65,500	70,500
Average tangential stress, psi	74,200	59,300	59,300
Maximum radial stress, psi	74,200	59,300	59,300
Burst margin	1.30	1.30	1.30
0.2% yield margin	1.14	1.03	1.03
LCF cycle limit, cycles	100,000	100,000	100,000

The accuracy of prediction of disk burst speed has been substantiated by spin pit tests and experimental engine tests as was shown previously.

Figure 33 shows the cooling air and leakage flow paths through the turbine. Compressor discharge air bypasses the burner and is introduced to the chamber on the front side of the first disk through holes in the inner combustor case. This chamber serves as the source of all turbine rotor cooling air and operates at 95% of compressor discharge pressure.

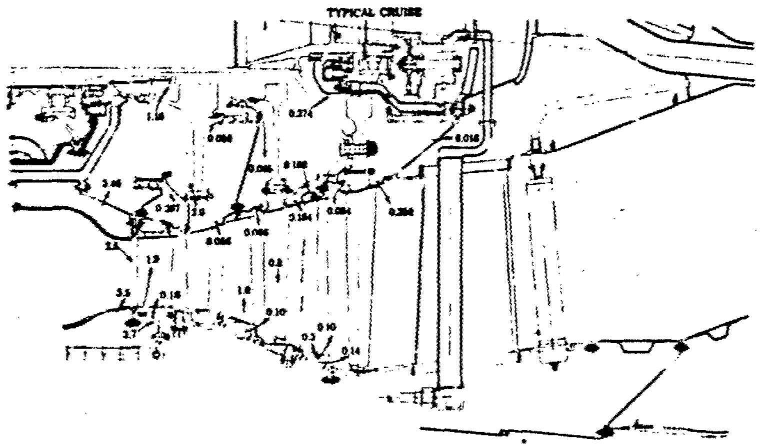

Figure 33. Turbine Cooling and Leakage Flow FD 17604 BIII

The labyrinth seals which are used to control leakage of turbine cooling air into the gas stream consist of a series of rotating stepped knife edges and matching abradable stationary shrouds. The combination of thin knife edges, small radial clearance, and an adequate expansion chamber has proved to be very effective in this type application. Both elements of the seal are readily replaceable.

The twin labyrinth seals between the 1st- and 2nd-stage disks minimize the leakage of cooling air into the hot gas stream. The pressure in the chambers around each disk is maintained above main stream pressure to prevent entry of main stream gases. By placing the feed path for blade cooling air on the back face of the second disk, supporting the disk on an integral arm coverplate, and incorporating twin labyrinth seals that are designed to be insensitive to transient thermal conditions, cooling air leakage is reduced to a practical minimum. Each seal is designed to be isolated from the thermal growth of any large-mass parts to which they are attached. A long cylinder or cone is used to reduce the displacement of the seal during transients, maintaining design clearance under all operating conditions.

Air to cool the attachment and airfoil of each 1st-stage blade flows through the scalloped coverplate flange around the bolts to the blade root. The passages in the extended neck of the blade are sized to provide the correct amount of air to each blade. This is a fail-safe feature that allows one or more of the airfoils to be damaged and yet not have all the cooling airflow through the damaged blade. Disk cooling air for the 2nd- and 3rd-stages is metered through holes in the 1st-stage disk integral arm. The 2nd-stage blade air follows a path up the back face of the disk similar to that of the first stage. The 3rd-stage air is metered through holes in the low rotor shaft integral arm.

(4) Shaft

The low pressure rotor incorporates a two-piece permanently assembled shaft, as illustrated in figure 28. One piece (a stub) forms the No. 4

bearing hub. It is pressed to the other piece and permanently riveted; however, these rivets take no driving torque. Positive torque transmission is provided through machined integral drive splines. Final machining of the shaft is accomplished after assembly. An access plate in the rear hub of the low turbine shaft permits radioisotope inspection of the entire engine without removal from the airframe. Replacement of the stub is provided for in event of damage in the No. 4 bearing compartment area. Turbine rotor shafting is designed using the critical speed criteria described in Volume III, Report B, Section II. Adequate radial clearance is provided between the high and low rotor shafts to prevent contact due to unbalance from up to 10% blade loss.

(5) Unit Construction

Unit construction is incorporated in the low turbine for ease of maintainability. This allows the low turbine to be removed and replaced without rebalancing, and tedious stage-by-stage teardown and reassembly are eliminated. This also permits rapid replacement of damaged 1st-stage blades, which can be removed from the rear after removal of the retaining nut.

b. Stationary Components

(1) Case

The turbine case holds the vanes in place and is the outer pressure vessel of the engine. It extends from the outer rear combustor case to the turbine exhaust case and is fabricated from a one-piece forging. The wall thicknesses of the case are set either by machining limitations or by blade containment requirements. Blade containment necessitates a 0.205-inch minimum wall thickness for the case surrounding the 1st-stage blades. The remainder of the case has a 0.070-inch thickness. A thickened exhaust case provides the 3rd-stage blade containment.

A detailed description of the analytical technique used in designing the case for blade containment is included in Volume III, Report B, Section II. The design criterion for blade containment is that the strain potential energy of the case shall exceed the kinetic energy of the failed blades. The case strain potential energy is determined from material physical property values modified by empirical factors derived from spin-pit and ballistic tests. Shear containment factors were also checked and found not limiting.

The case serves as a supply plenum to feed cooling air to the vanes through the annular space between the vane platforms and stationary shroud seals and the outer case. Case walls are convectively cooled so that the wall temperature does not exceed 1450°F. Bosses are provided on the turbine case to accommodate borescope inspection of the 1st- and 2nd-stage blades. Inspection of the 1st-stage vanes (if required more frequently than at hot section inspections) can be accomplished by borescope through the outer combustor and duct heater combustor support case. Points of attachment between the turbine vanes and case members are protected from wear by use of hardfaced surfaces. The 1st-stage vanes may be removed at hot section inspection.

(2) Tip Seals

The turbine case holds the stationary-shroud tip-seals in place around each rotor. These reduce turbine tip clearance and leakage, and increase turbine efficiency. Stationary-shroud seals consist of thermal-response rub-strips described in the Aerodynamic Design section. Points of contact between the shroud seals and the case are hardfaced to reduce wear.

(3) Vanes

All vanes are constructed in pairs with a common platform and attachment. This arrangement provides an important safety feature. In the event of burnthrough of an airfoil, the remaining airfoil portions are retained by the adjacent airfoil and platform and attachment structure. This system also reduces cooling air leakage between vane platforms.

In the 2nd- and 3rd-stage vanes, pairing of the vanes also reduces weight. The bending moments in the airfoils are reduced because the interconnecting inner platform converts the airfoils to a guided cantilever design. This allows the design of vanes of shorter chords. The larger outer platforms give a longer attachment wheelbase and reduce the loads induced in the outer support structure. Figure 34 shows a paired J58 2nd-stage vane.

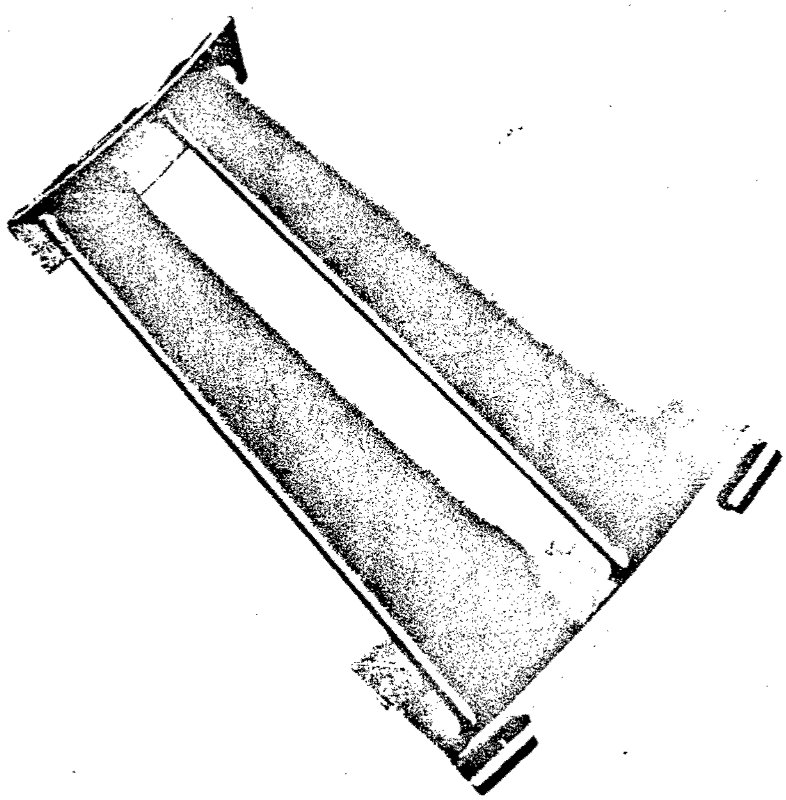

Figure 34. J58 2nd-Stage Paired Turbine Vane Showing Load Distribution

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(4) Turbine Exhaust Section

Aerodynamically, the turbine exhaust section removes the tangential swirl from the turbine exhaust. Mechanically, it acts as the No. 4 bearing support. It consists of an inner and outer case, the exit guide vanes, and the exhaust gas temperature probes. The outer turbine exhaust case is made from a one-piece forging.

There are 16 exit guide vanes to remove the tangential velocity in the turbine exhaust with a minimum pressure loss. These also serve as passages for oil and breather air for the No. 4 bearing compartment, and as the structural support of that compartment.

The inner and outer cases are weldments fabricated from AMS 5707 (Waspaloy). The 16 PWA 655 (Inconel 713) exit guide vanes are brazed into the cases with nickel alloy braze AMS 2675. Brazed joints are used because of their good heat conduction characteristics. During startup and shutdown, the vanes heat and cool more rapidly than the cases. This design minimizes the thermal gradient between the case and the vane.

Minimum weight is obtained by balancing the combined thermal stresses in the two cases and the vanes for optimum structural efficiency. The No. 4 bearing support and exit guide vane system are designed to sustain the load resulting from a turbine failure involving a 10% blade loss, without stressing the loaded elements beyond the yield strength of the material. The forward end of the outer turbine exhaust case is thickened to provide blade containment for the 3rd-stage turbine blades.

The turbine exhaust gas temperature probes are bolted to a pad on the inside of the outer turbine exhaust case and supported at the inner case by a spherical ball (figure 35). This construction technique permits removal of the probe from inside the engine duct without access through the airframe nacelle.

For maintenance purposes, the temperature probe outer case (figure 36) can be readily separated from the temperature sensing portion. The probe is vented through the spherical support ball at the inner end to a tube in the exhaust tailcone that exhausts at the throat of the primary nozzle. This configuration aspirates exhaust gas over the thermocouple junction, providing rapid response.

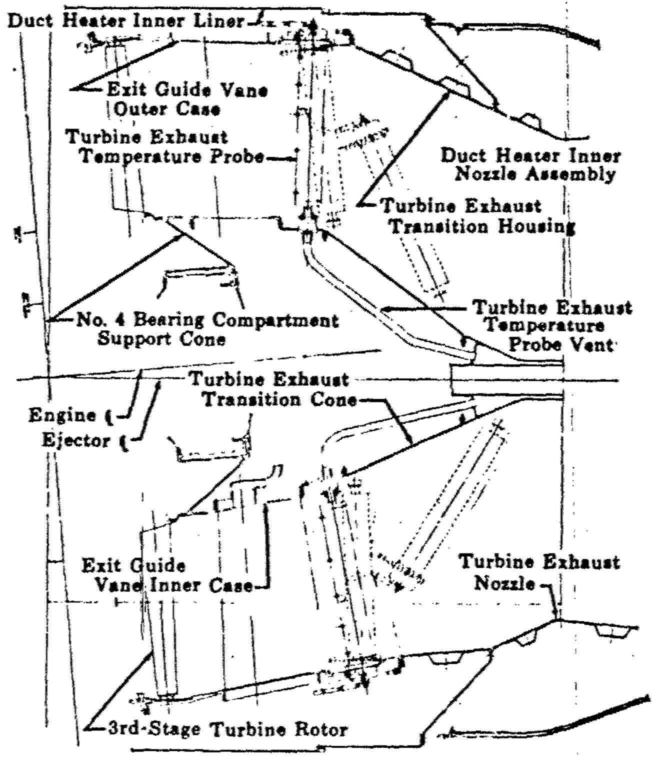


Figure 35. Turbine Exhaust Gas Temperature Probe Removal — FD 16226 BIII

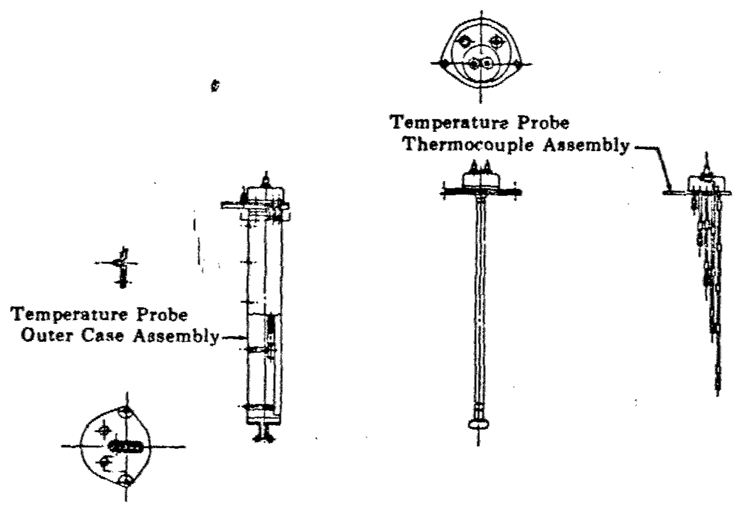


Figure 36. Turbine Temperature Measuring Probe — FD 16215 BIII

5. Product Assurance

Many of the design features of the JTF17 turbine have been incorporated specifically to enhance the maintainability, reliability, and safety of the engine. Also, special features are included in the design to minimize human errors in assembly, and value engineering criteria have been applied, for example, in the selection of materials and in the design of parts subject to wear and replacement. Some of the more important of these product assurance features and benefits are tabulated below.

a. Maintainability

1. The 1st-stage turbine vanes are replaceable without rotor removal.

2. Moment-weighted blades are used in all three stages to permit replacement in pairs without subsequent rebalancing.

3. The 1st-stage blades are replaceable without first disk removal after low turbine assembly removal.

4. The 1st-stage vanes, and blades, and 2nd-stage blades are inspected by borescope. The 3rd-stage blades are inspected by viewing through the exhaust nozzle.

5. Balanced unit construction of the low turbine allows assembly directly to the engine without rebalancing.

6. The 3rd-stage disk and blade assembly can be removed and replaced without subsequent low turbine rebalancing.

7. Labyrinth seals and associated abradable stationary-shrouds are readily replaceable.

8. Radioisotope probe access is provided through the rear hub.

9. Exhaust gas temperature probes are removable from inside exhaust nozzle, making access through aircraft nacelle unnecessary.

b. Reliability

1. Elimination of separate high rotor turbine hub reduces joints in the rotor system to improve balance and vibration characteristics, and eliminate holes in high stress regions of the disk.

2. The 2nd-stage disk support arm and rear coverplate eliminates holes in high stress areas of the disk.

3. All vanes are manufactured and assembled in pairs so that in the event of a single vane burnthrough the adjacent vane on the common platform retains the damaged airfoil.

4. All cooling air orifices are nonadjustable and not subject to assembly errors. Blade cooling air is orificed on the blade at the cooling air entrance. This allows airfoil damage without affecting the cooling flow to the remaining blades.

5. The twin labyrinth seal between the first and second disk is insensitive to transient thermal conditions to ensure a continuous flow of cooling air to the disks.

6. Adequate clearance is incorporated between the low and high rotor shafts to prevent contact in the event of up to 10% blade loss.

7. The turbine case and turbine exhaust are made from one-piece forgings avoiding potentially unreliable welds.

8. Convectively cooled vanes and blades are simple, rugged parts not sensitive to damage by foreign objects.

c. Value Engineering

1. Labyrinth seals and associated abradable stationary-shrouds are readily replaceable without involving large, expensive parts.

2. Points of attachment are protected from wear by hardfaced surfaces.

3. The lowest cost materials that could satisfy the service and temperature requirements were selected. Examples of material cost reductions incorporated in the JTF17 design that differ from the initial experimental engine are:

   a. Less costly AMS 5754 (Hastelloy X) in place of PWA 1004 (Waspaloy) for stationary blade shroud seals

   b. Maintenance costs are reduced by the elimination of separate high rotor turbine hub and associated bolts and nuts for attachment to first disk

   c. Inspection costs are reduced because of the access provisions provided for borescope inspection and radioisotope probing

   d. Balanced unit construction of the low turbine reduces maintenance costs by eliminating the necessity for rebalancing after assembly in the engine.

d. Human Engineering

1. Cooling air distribution errors due to improper assembly are eliminated by nonadjustable cooling air orifices.

2. The bolt circle of the 3rd-stage disk attachment to the hub contains one offset hole to ensure proper assembly.

3. The knife edge seals and associated stationary shrouds contain offset holes to ensure proper assembly.

4. The interstage stationary shroud diaphragm assemblies are designed to prevent improper assembly.

e. Safety Engineering

1. Access provisions are provided for borescope and radiography inspection.

2. 3rd-stage disk and blade assembly is provided with an offset hole to prevent unbalanced reassembly.

3. Radial and axial clearance is provided between the rotor and the stators to prevent rubbing during all operating conditions.

4. Any large axial rotor shifts will bring blades in contact with vanes before any disk contact is made.

5. Adequate clearance is designed between high and low rotor shafts to prevent contact in the event of up to 10% blade loss.

6. Blades and vanes are designed to prevent foreign object damage causing loss of cooling.

7. All hot section parts are cooled where required.

D. DUCT HEATER

1. Introduction

The high level of engine thrust required to meet the takeoff noise, transonic acceleration, and sonic-boom altitude requirements for the supersonic transport has resulted in a need for a thrust augmentation system for either turbojet or turbofan. The augmentation systems used in military service have not demonstrated the dependability and long life required for commercial airline service. One of the reasons we have proposed the turbofan engine for the SST is that it provides a solution to this fundamental problem.

Our first experiments ten years ago to augment a turbofan engine by burning in the fan duct were far from successful; the problems of burning relatively cool air at low enough duct pressures without excessive internal drag then appeared insurmountable. More recently, progress has been made in the TF30 engine by using aerodynamic flameholders in a coburner arrangement. Our tests indicated that this aerodynamic flame-holder was one possible approach to augmenting an SST turbofan, but high combustion efficiency was desired over a wider fuel/air ratio range than had been demonstrated by this means. Another requirement that was not demonstrated by any existing burner system was to be able to light the augmentor at low enough fuel/air ratios that no sensible pressure pulse would occur. This objective we felt necessary to minimize aerodynamic interactions between the aircraft supersonic inlet and the engine. The ram induction burner concept that we have included in the JTF17 engine design has met all our objectives:

1. Ignition and flame propagation at below 0.002 fuel/air ratio, well below the level that would result in a sensible pressure pulse.

2. High combustion efficiency, burning relative cool air over the full augmentation range required, up to 0.060 fuel/air ratio.

3. Low enough pressure loss to retain the low subsonic fuel consumption characteristic of the non-augmented turbofan.

This new augmentation concept has made practical the application of the turbofan engine to the supersonic transport. By so doing, it provides a unique solution to the problem of obtaining commercial airline dependability in a high-heat-release augmentor system. The burner liners and outer duct are cooled by fan discharge air, which is available in large quantities. The duct burner metal temperatures can thus be brought down to the level of those in combustion systems now operating continuously in commercial service. This benefit in durability also carries over into the aircraft nacelle structure, which "looks at" a cool outer duct, not at a cherry red afterburner duct. The fuel manifolds and nozzles are simple and also well-cooled by fan air so that they will not experience the coking and maintenance problems associated with fuel systems that live in a high

temperature turbine exhaust environment. The JTF17 duct heater design not only takes advantage of the long life inherent with reduced metal temperatures, but also incorporates such airline desired maintenance features as replaceable primary combustor modules, easily removable fuel nozzles, and small replaceable modular outer liner segments.

D. DUCT HEATER

2. Description

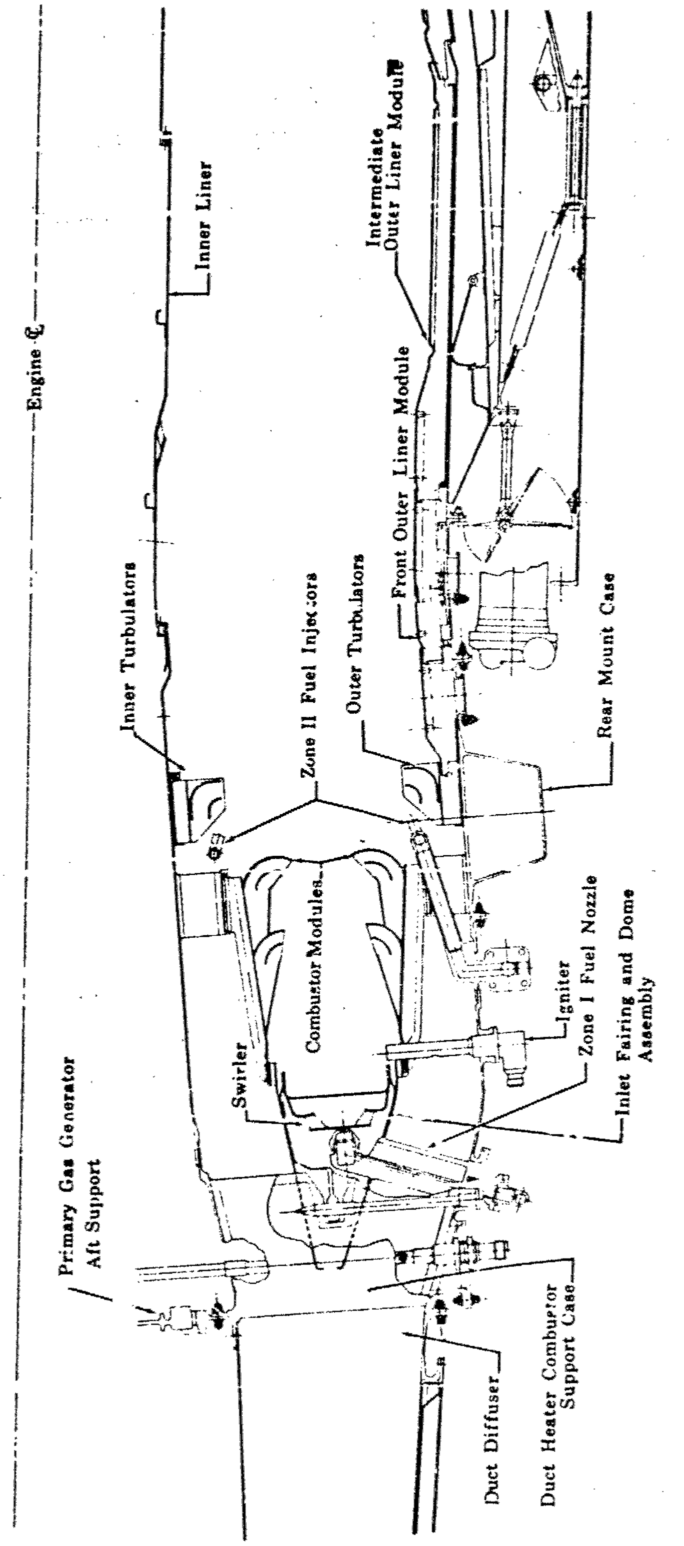
Figure 1 illustrates the two-zone annular ram-induction combustor (similiar in design to the primary combustor described in Section BIIB) and the following component parts:

1. Diffuser
2. Combustor support case and gas generator aft support
3. Inlet fairing, dome and swirlers
4. Fuel system and igniter
5. Combustor and liners.

The airflow path between the fan and the combustor is formed by the inner and outer walls of the diffuser and the combustor support case. Fan discharge air at a maximum of 660°F ensures that all components are adequately cooled to temperatures commensurate with extended service life. Fan discharge air enters the combustor through dome-mounted swirlers and ram-air scoops in modules that form the combustor walls. The swirlers provide a flow circulation region in which combustion with smooth ignition can take place. In the ram-induction combustion region (called Zone I), fuel is supplied by 40 uniformly-spaced nozzles located at the center of the annular combustor support case, and in the inlet fairing and dome assembly. Ignition in Zone I is achieved by electric spark igniters.

The Zone II combustion region at the combustor exit is used to supplement Zone I. Air is introduced into Zone II by 90-degree turbulators. Zone II fuel, which is introduced near both the inner and outer turbulators through 270 uniformly-spaced fuel injectors, ignites spontaneously from the Zone I combustion gases with no significant pressure fluctuation.

The gases from the duct heater are routed to the variable-area nozzle by the inner and outer liners and the case assemblies.



FD 17054
BII

Figure 1. Duct Heater Cross Section

3. Objectives and Requirements

The design objectives for the duct heater are consistent with those for the engine, as stated in the Introduction, Section I, of this report. Specific objectives for the design are to:

1. Furnish combustion gases to the nozzle at the desired average temperature and temperature profile
2. Obtain maximum heat release of the fuel in a minimum combustion chamber volume
3. Obtain maximum exit pressure by minimizing combustor pressure losses
4. Obtain ignition smoothly
5. Provide the capability for efficient maintainability by using parts that minimize assembly time and eliminate the requirement for special tools
6. To generate little or no smoke to minimize community air polution.

The mechanical design requirements for the duct heater are consistent with those stated in the Introduction, Section I of this report. In addition to meeting the necessary maintainability, reliability, and value engineering requirements for the individual duct heater components, the duct heater is designed to permit servicing of the components of the gas generator through the duct heater. This feature will be discussed in more detail in paragraph b.

Design requirements related to duct heater performance are summarized in table 1. In addition to these requirements, the duct heater is to operate over the fuel/air ratio range of 0.002-0.062 and ignition capability is required throughout the engine operating envelope.

Table 1. Duct Heater Performance Design Requirements

	SLTO	Cruise
Combustion efficiency, %	91.1	97.0
Temperature rise - ΔT, °F	2880	936
Total pressure loss - $\Delta P/P$, %	10.8	8.30
Fuel/air ratio - f/a	0.060	0.0153
Duct exit temperature, °F	3172	1596

The most important performance requirement is high combustion efficiency. The initial required (or goal) combustion efficiencies given in table 1 have already been achieved in the Phase II-C program during which a total of 421.5 hours of testing were accomplished with several duct heaters; combustion efficiencies higher than 95% were confirmed while operating in the relatively low-temperature, low-pressure environment of the duct fan. Based on this test experience, Pratt & Whitney Aircraft is confident of achieving a high-performance, long-life duct heater design for the JTF17 engine.

4. Design Approach

a. Detailed Description

A detailed description of the major components and the considerations that dictated the design approach in response to objectives and requirements of the duct heater follows.

(1) Diffuser

The diffuser joins the intermediate case to the duct heater combustor support case, as indicated in figure 2. The diffuser geometry is essentially the same as that used in the initial experimental engines. The diffuser is formed by the inner and outer cases which are connected by eight airfoil-shaped struts. These eight struts are continuations of the struts in the intermediate case and are used to route the gas-generator plumbing to the outside of the engine.

Eight round access ports are provided in the outer case to permit inspection and maintenance of the gas-generator components (specifically the gas-generator fuel nozzles and inner plumbing connections). These ports are located between accessory components and plumbing as illustrated in figures 2 and 3 and, thus, are readily accessible.

The inner diffuser wall has a boundary-layer bleed for the duct-heater-inner-wall cooling air. (Approximately 10% of the airflow is bled for this purpose.) The cooling air is bled from the inner wall at a location near the trailing edge of the struts. (See figure 2.)

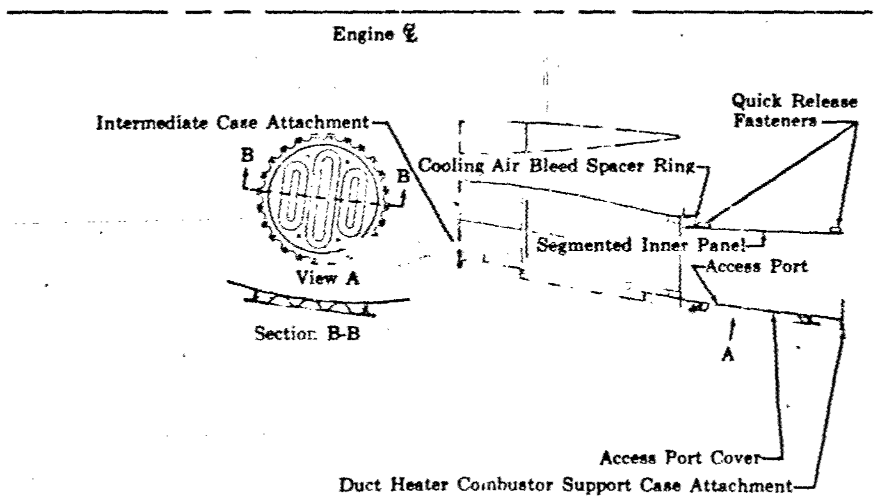


Figure 2. Duct Diffuser FD 16253 BII

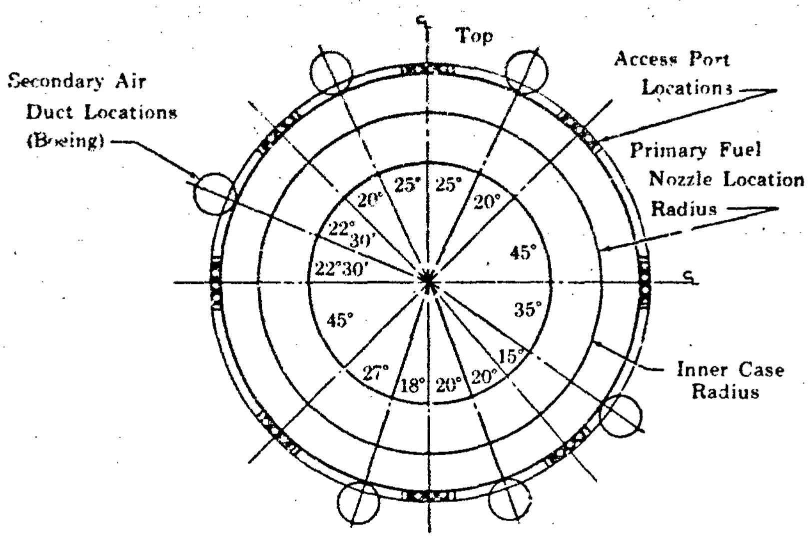


Figure 3. Duct Diffuser Case Access Port Location - Looking Aft FD 16252 BII

The rear inner duct case is split axially into four equal panels which have overlapping joints. The four panels are supported from the inner flange of the combustor support case. The panels are joined by quick-release fasteners to reduce the time required for servicing or inspection of the gas generator. The normal procedure to inspect the gas generator is to (1) remove the access-port covers, (2) unlatch the quick-release fasteners of the inner panel, and (3) slide the panel circumferentially out of the way. (See figure 2.)

Butt-weld construction, proven in previous production engines, is used throughout the fan diffuser to ensure reliability and repairability as well as to provide readily inspectable weld joints with reduced stress concentration. AMS 4910 (A-110 titanium) alloy is used for all of the fan diffuser parts. The case is elastic-buckling-limited and is sized for a 30% buckling margin. This is an established criterion proven by commercial engine experience.

In the diffuser, the velocity of the fan air at cruise conditions is reduced from an inlet Mach number of 0.529 to an exit Mach number of 0.211. Table 2 summarizes significant diffuser flow parameters for cruise as well as other pertinent flight operating points.

Table 2. Design Flow Parameters

	SLTO	Mach No. = 2.7 Alt = 65,000 ft
Inlet Mach Number	0.449	0.529
Inlet Velocity, ft/sec	635	838
Exit Mach Number	0.194	0.211
Total Pressure Loss, $\Delta P/P_T$ - %	0.9	1.36

The pressure losses in the JTF17 diffuser are extremely low. Test results showing the total pressure loss for the diffuser are given in figure 4. The test results have also shown that total pressure losses will increase with a distorted diffuser inlet profile. Based on initial experimental engine tests, the inlet profile is essentially flat at all flight conditions tested, thus guaranteeing minimum diffuser pressure losses. Representative diffuser inlet and exit (combustor inlet) Mach number profiles determined from full-scale tests, are shown in figure 5.

Pressure-loading and airflow distribution at the flight points specified in the RFP are shown in figure 6.

(2) Combustor Support Case and Gas Generator Aft Support

The combustor support case (figure 7), which is made of PWA 1202 (titanium MST811), provides the support for the combustor and transfers part of the gas-generator load to the outer cases. Butt-welded construction throughout the basic structure minimizes weld shrinkage, distortion, and built-in stress concentration points. This method of construction greatly facilitates inspection and quality control of the welds and provides reliability and structural integrity of the assembly.

The inner and outer walls of the combustor support case are connected by eight equally-spaced airfoil-shaped struts. Stiffening rings are used at the leading and trailing edges of the struts to distribute the strut loads into the case. The major loads are caused by (1) maneuvers, (2) thermal gradients between the inner and outer walls, (3) gas pressure, (4) combustor blowoff load, and (5) loads from the gas generator. The outer case is elastic-buckling-limited and is sized for a maximum stress of 22,900 psi. This design provides a 30% buckling margin, which is a criteria consistent with commercial engine designs.

The gas generator has two supports; the aft support is attached to the gas generator diffuser case and the combustor support case as shown in figure 7. The "load-path" is from the gas-generator diffuser case, to the aft support, through the combustor support case, and to the engine rear mount. The support arrangement permits the use of lightweight titanium in the intermediate case because it reduces the loads and deflections in the case.

The limiting loads on the aft support will occur during an aircraft maneuver. P&WA commercial engine experience has proven that design stress values of 2/3 the ultimate strength are acceptable criteria for stress-limited short-time loads.

The material selection for the inner portion of the aft support is PWA 1009 (Inconel 718). Strength-to-weight ratio at maximum operating temperature dictates the choice of material. The material choice for the outer portion of a support is AMS 5616 (Greek Ascoloy), which has thermal-expansion compatibility with titanium as well as good strength-to-weight ratio. Figure 7 shows the configuration.

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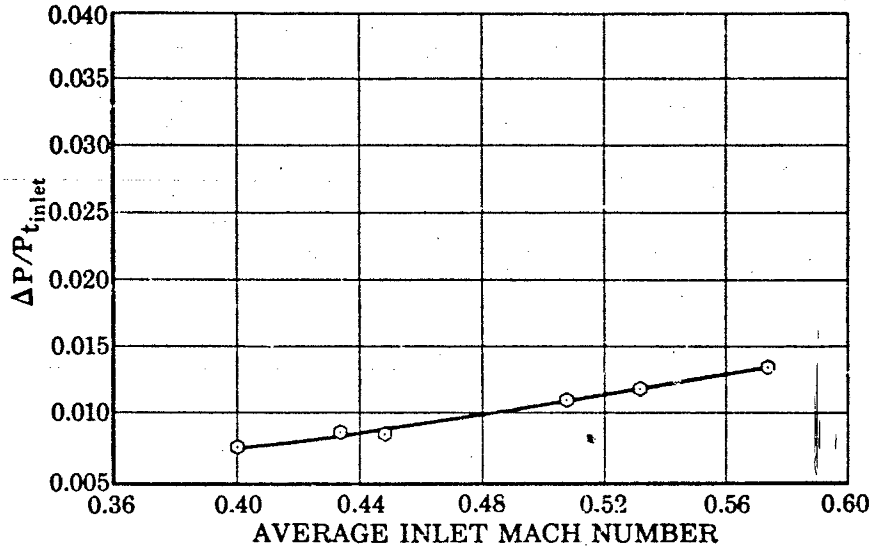

Figure 4. JTF17 Full-Scale Duct Heater Rig Diffuser Total Pressure Loss vs Inlet Mach No. FD 16255 BII

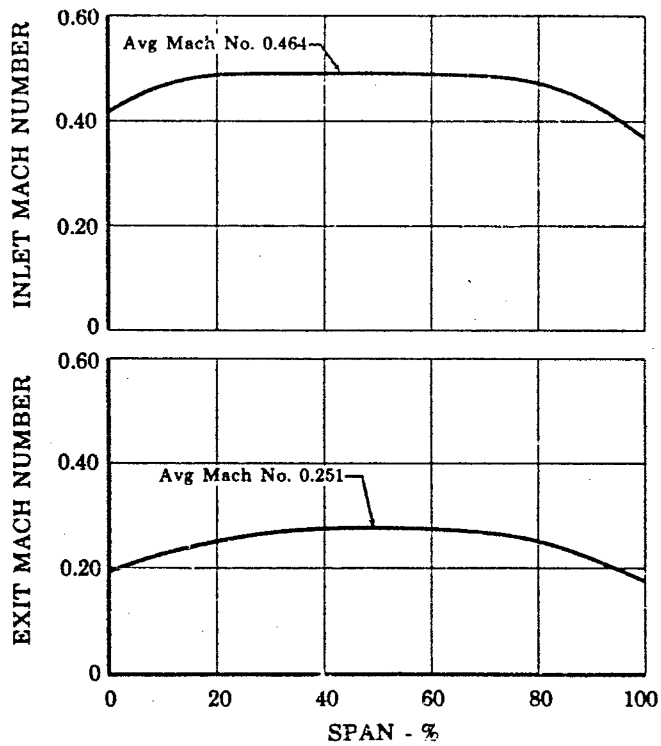

Figure 5. Average Mach No. Profiles at Diffuser Inlet and Exit FD 16254 BII

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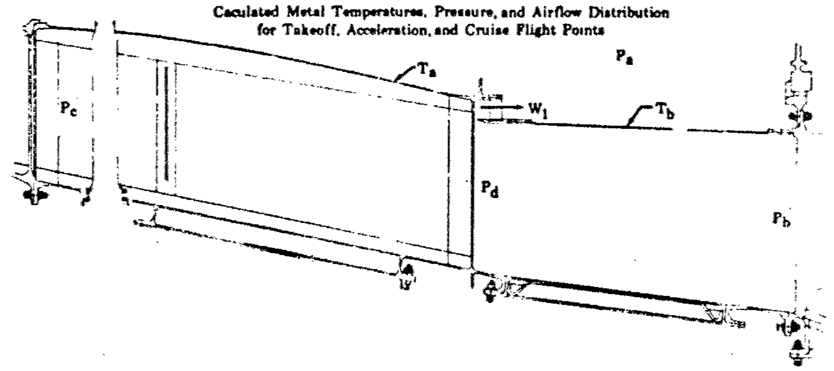

	Max D/H SLTO	Min D/H SLTO	Max D/H M 1.5 and 45K ft	Min D/H M 1.5 and 45K ft	Max D/H M 2.7 and 65K ft	Min D/H M 2.7 and 65K ft
T_a - °F	300	300	322	322	656	656
T_b - °F	302	302	323	323	657	657
W_1 Duct Flow - %	10.4	6.6	10.6	7.6	9.9	7.8
P_a - psia	41.3	42.4	18.9	19.3	24.6	24.9
P_b - psia	42.0	42.0	19.2	19.2	24.7	24.7
P_c - psia	38.2	38.2	17.0	17.0	21.4	21.4
P_d - psia	41.6	41.6	19.1	19.1	24.4	24.4

Figure 6. Duct Heater Diffuser Calculated Metal Temperature Pressure, and Airflow Distribution — FD 17050 BII

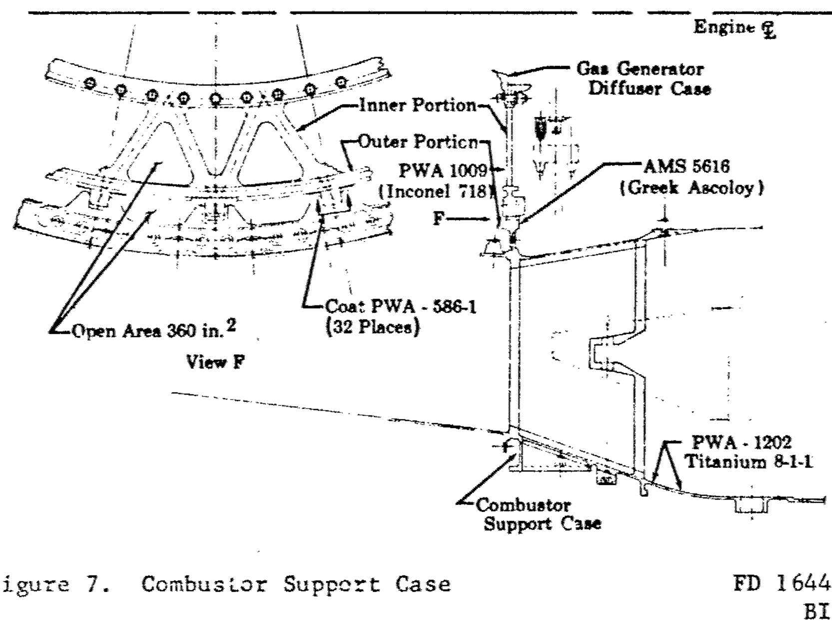

Figure 7. Combustor Support Case — FD 16443 BII

The radial square-spline (shown in figure 7) provides quick assembly or disassembly of the duct heater and gas generator in the same manner as accomplished on the initial experimental engine. At engine assembly, the duct heater from the combustor support case rearward is installed as a major subassembly around the gas generator. The splines transfer the load of the gas generator into the duct heater cases while allowing

for radial and axial expansions. To improve wear resistance, the outer spline teeth are coated with an antigalling dry lubricant compound, Specification PWA 586-1.

(3) Inlet Fairing, Dome, and Swirlers

The inlet fairing and dome assembly (figures 1 and 8) is identical to that used in the initial experimental engines. The inlet fairing channels the air to the inner and outer combustor annuli. Air admitted through the front of the fairing is diffused to provide a high static pressure differential across the dome. This ensures good swirler and dome-cooling airflows at all flight conditions.

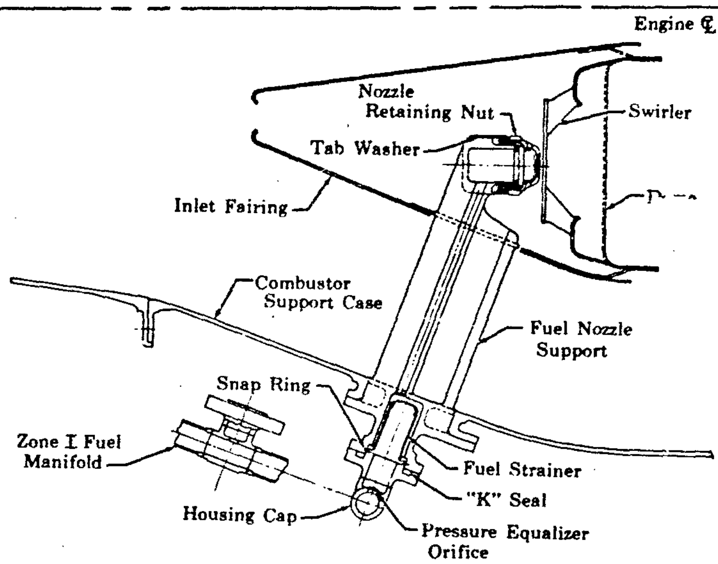

Figure 8. Duct Heater Zone I Fuel System

FD 17570
BII

The inlet fairing assembly is a full annular structure fabricated from two sheet metal cones. It has 40 equally-spaced support ribs. The fairing is cut out to clear the eight airfoil-shaped struts in the duct heater support case. Notches in the strut trailing edges (shown in figure 7) accept lugs on the inlet fairing dome assembly. Retaining pins are used to attach the lugs to the struts. This method of attachment allows relative component part thermal expansion while holding the fairing concentric in the duct.

The ribs stiffness will allow a maximum pressure differential of 26 psi across the fairing, which is considerably higher than the maximum anticipated pressure of 7.5 psi. This maximum pressure would occur only during the duct nozzle failure modes when the gas flow in the duct diffuser becomes supersonic.

Inlet fairing aerodynamic characteristics have been visually observed using a water table. Positive flow to the swirler ports is confirmed by the streamlines depicted in figure 9.

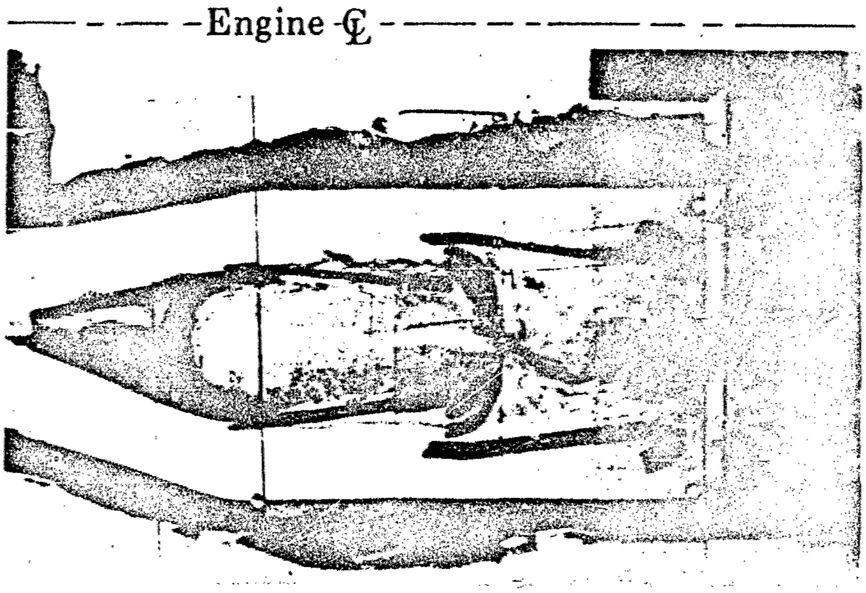

Figure 9. Water Table Analogy Streamlines Feeding Swirlers FD 17049 BII

The swirler (figure 10) blading has an average pitch-to-chord ratio of 0.7, which is consistent with P&WA experience. It provides the high-swirl velocity necessary to promote circulation by inducing a low-pressure region in the center of the combustor, and therefore good fuel/air mixing. Based on previous commercial engine experience, AMS 5382 (Stellite 31) was selected for the swirlers because this alloy has good oxidation resistance at elevated temperatures and can be precision cast and thus provides a cost saving over fabricated sheet metal swirlers.

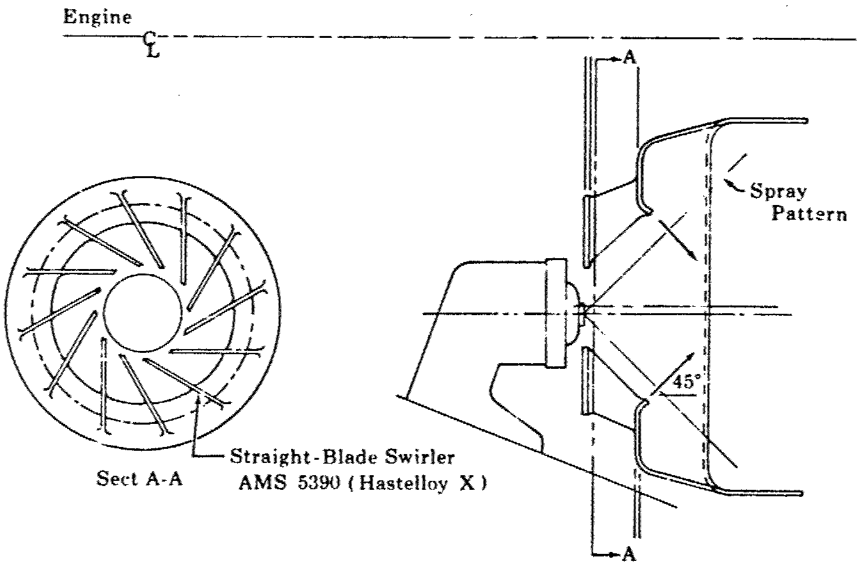

Figure 10. Duct Heater Swirler Arrangement FD 16240 BII

A total of 421.5 hours of engine and full-scale rig testing has shown adequate dome cooling and cleanliness. Figure 11 shows the excellent condition of the dome assembly; no evidence of overheating existed.

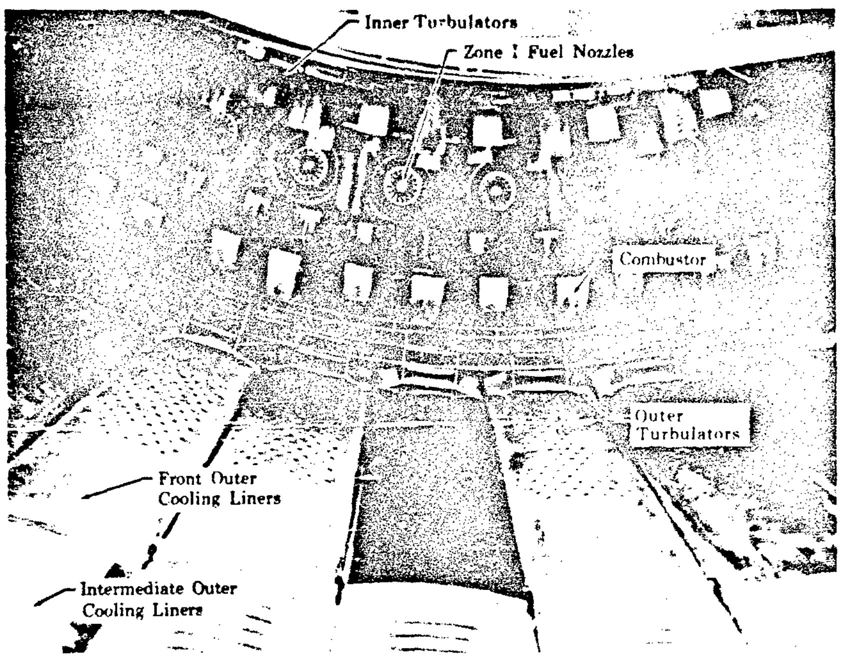

Figure 11. Annular Duct Heater After Cruise Testing — FD 15313 BII

(4) Fuel System and Igniter

The duct heater has separate fuel injection systems for Zone I and Zone II. The adequacy of the two-zone injection system has been demonstrated in tests. Figures 12 and 13 present performance data obtained in Phase II-C tests. As shown, the Zone I system has already demonstrated the capability of producing more than 95% combustion efficiency at cruise conditions. The Zone II system, which supplements the Zone I during aircraft takeoff and acceleration, has demonstrated the capability to produce more than 93% combustion efficiency.

The ignition system is composed of two 4-joule, low-tension, surface-gap spark plugs. The igniter plugs are cooled by aspirating fan discharge air. The adequacy of this system has been demonstrated in sea level and altitude full-scale duct heater tests. Ignition is accomplished with a slight rise in duct heater pressure, as shown in figures 14 and 15. Successful lightoffs were obtained with combustor fuel/air ratios in the 0.0014 to 0.0048 range. The phasing-in of the Zone II fuel in incremental changes produced no pressure discontinuity (figure 16).

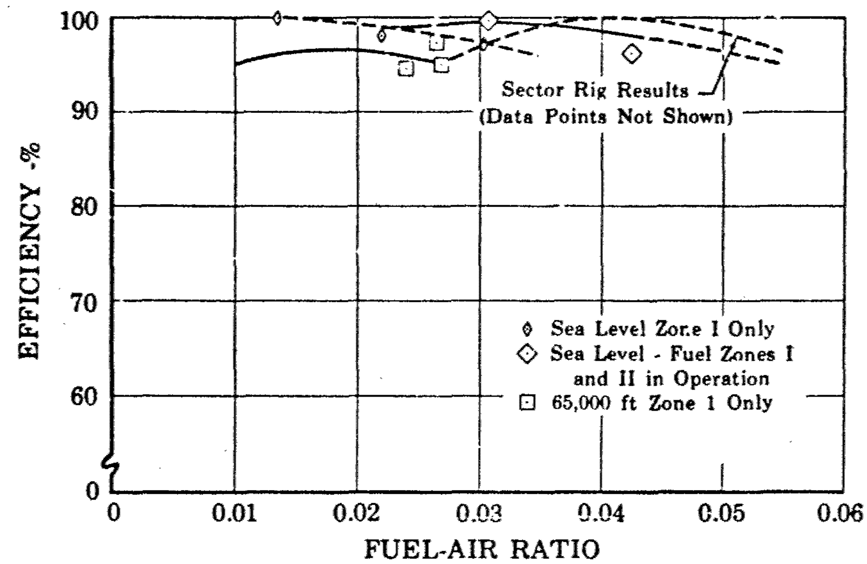

Figure 12. JTF17 Full-Scale Duct Heater Rig Efficiency vs F/A Ratio FD 16250 BII

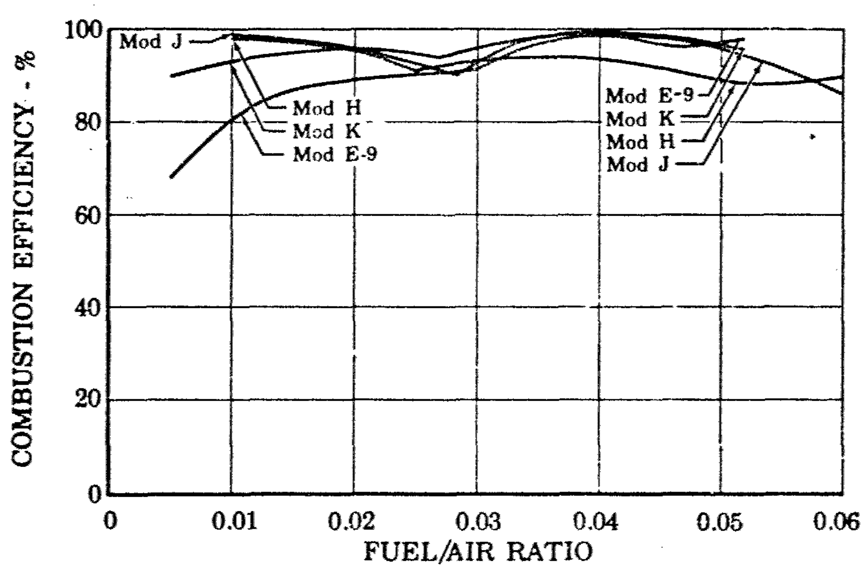

Figure 13. Duct Heater Combustion Efficiency at Takeoff Conditions FD 16244 BII

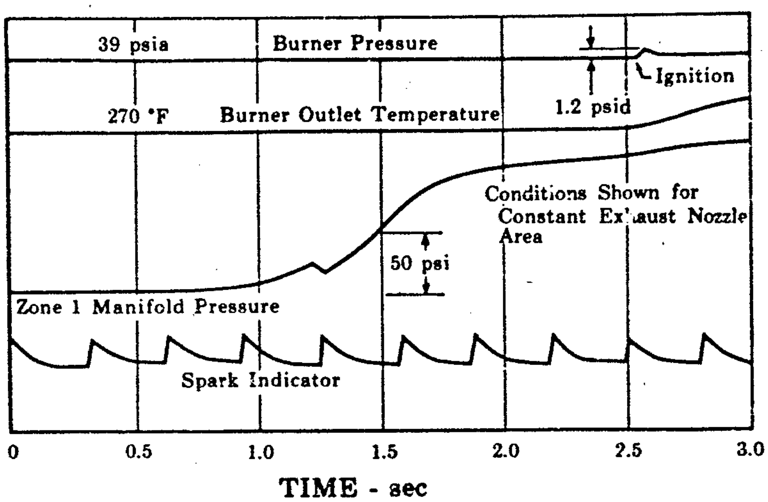

Figure 14. Full-Scale Annular Duct Heater Rig - Sea Level Ignition Test Results

FD 15304A
BII

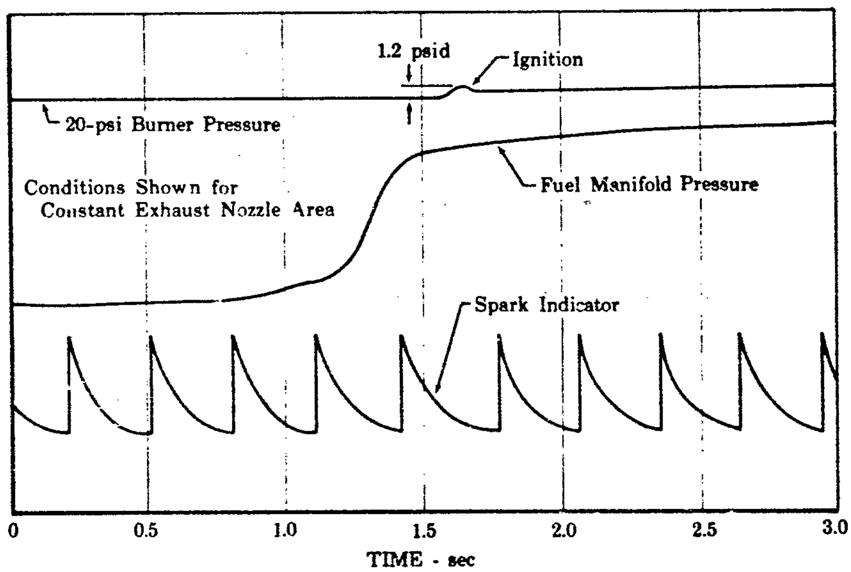

Figure 15. Full-Scale Duct Heater Rig - Cruise Ignition Test Results

FD 15303A
BII

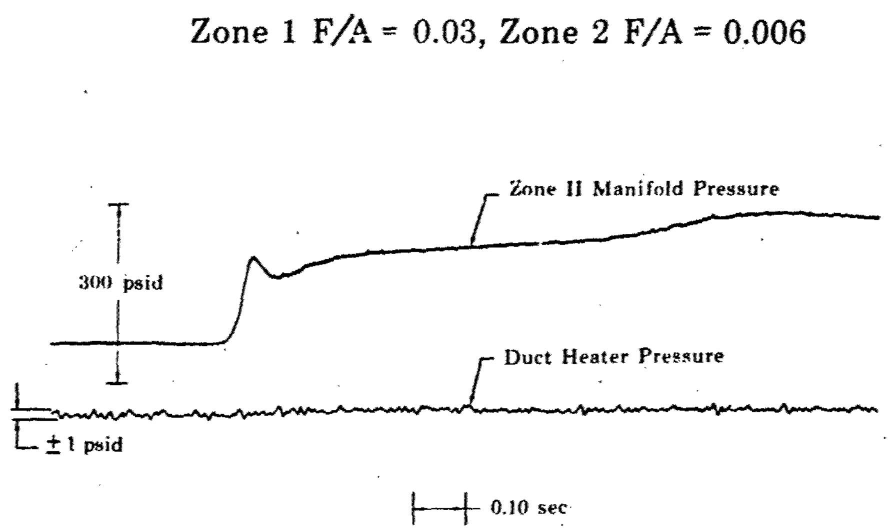

Figure 16. Rate of Duct Heater Pressure Rise With Addition of Zone II Fuel (Rig Data) FD 14605E BII

Overall duct fuel system flow range is determined from the range of fuel/air ratio requirements. The minimum fuel/air ratio has been set at 0.002 for ignition. At this level there will be (1) negligible fan duct pressure and airflow upset during duct heater lightoff, (2) no change in aircraft inlet setting during duct heater lightoff, and (3) smooth thrust transition at lightoff.

The maximum required fuel/air ratio is 0.060, and it occurs during acceleration at a sea level Mach number of 0.7. From the operating envelope, the minimum and maximum fan airflows are 75 lb/sec and 476 lb/sec, respectively. From the above, the required maximum-to-minimum fuel flow ratio is determined as follows:

$$\frac{\text{Maximum Fuel Flow}}{\text{Minimum Fuel Flow}} = \frac{0.060}{0.002} \times \frac{476}{75} = 190.5$$

The maximum available pressure drop based on fuel pump design pressures and line losses is estimated to be 800 psi. The lowest allowable pressure drop with which adequate fuel atomization can be achieved, is 15 psi. The maximum-to-minimum fuel flow ratio available from a single set of fixed geometry nozzles is the square root of the ratio of the maximum pressure drop to the minimum pressure drop ($\sqrt{(800/15)}$) or 7.4. Comparing this ratio with the required ratio indicates that a fixed geometry system would not be adequate without using undesirable high pump discharge pressures. Furthermore, it can be shown in a similar manner that different sizes (but fixed area) nozzles for Zone I and Zone II would also not suffice and therefore both Zone I and Zone II fuel injection systems use nozzles having variable-area features.

The Zone I fuel system is composed of 40 individually removable dual-orifice nozzles (fixed primary, variable secondary). The fuel nozzle selection is based upon J58 engine experience with this type nozzle. Identical fuel nozzles are also used in the JTF17 primary combustor. These fuel nozzles will provide spray characteristics (i.e., atomization and spray angle) that have demonstrated combustion stability and high performance in the J58 and other P&WA engines.

A Zone I nozzle installation is shown in figure 8. As shown, the fuel nozzle cartridge is inserted into a machined cavity in the support; a series of holes allows cooling air to be aspirated inwardly to wash the nozzle face and prevent fuel coking and carbon accumulation. Each fuel nozzle is protected from contamination by a strainer located in the nozzle housing. The fuel strainers are cylindrical and are made from perforated sheet metal having 0.012-in. diameter holes, which are smaller than the holes in the fuel nozzle. They are similar to the J58 fuel strainers. Each strainer is retained by a snap ring so that no contaminants can enter the nozzle housing while the Zone I fuel manifold is removed. Metal K-type seals are used at all the mechanical joints. Experience on the J58 engine has indicated a very high reliability for these seals in alternating fuel flowing and nonflowing applications.

The Zone I fuel system has 6 cluster manifolds, each having either 6 or 7 nozzles, depending on cluster location. The cluster system is used because of weight considerations and simplicity. In a given cluster the nozzles are arranged in series. To equalize the pressure at all nozzles, each nozzle cap has an orifice of different size (figure 8). The nozzle caps and connecting plumbing are welded together to form a cluster manifold. Welding the caps into a cluster assembly prevents inadvertent installation of the orifices in the wrong sequence or location.

The Zone I system maximum-to-minimum fuel flow ratio capability is 125. The associated system flow and injector pressure drop characteristics are shown in figure 17. The high value of maximum-to-minimum fuel flow ratio capability designed into the Zone I system provides margin for development and optimization of Zone I/Zone II fuel flow splits for maximum duct heater performance.

The Zone II fuel system adds fuel to the combustor bypass air. This system is composed of 10 manifold segments (five inner and five outer) that have a total of 270 injectors. Figure 18 shows a cross section of a Zone II fuel injector. Fuel pressure deflects the diaphragm, thereby providing the required variable flow characteristic. Slots in the seat impart swirl, resulting in a hollow cone spray. Fuel flow-pressure characteristics are shown in figure 19. The individual injectors are threaded into sockets welded to the tubing segments. This design allows the injector elements to be removed individually for ease of replacement or repair.

An outer manifold segment consists of a centrally located inlet fitting, tubing, injectors, and end plugs (figure 20). Because of the degree of reliability and durability attained with butt-welded joints in the J58 afterburning sprayring, butt-welded construction is used in

all Zone II manifold segments. The fuel inlet fitting protrudes through, and is bolted to pads machined on the outer case. The fittings are the straight-through banjo-type derived from the J58 afterburner sprayrings. Metal K-type seals are used in Zone II fittings. Spherical plugs, threaded into the ends of the segments, are retained in brackets that form the segments into a complete ring (figure 21).

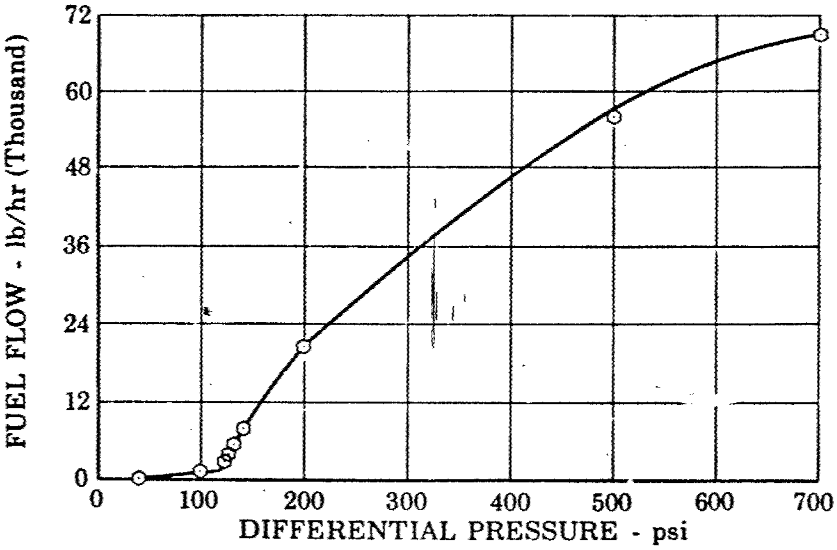

Figure 17. Zone I Duct Heater Nozzle Total Fuel Flow vs Pressure FD 16249 BII

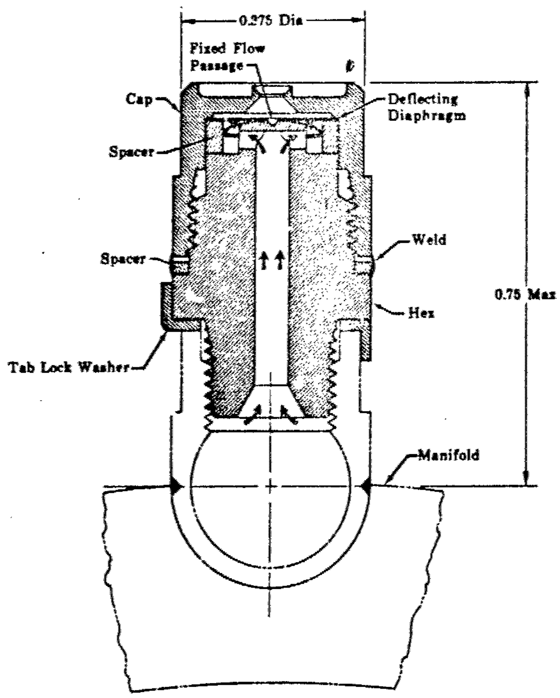

Figure 18. Zone II Duct Heater Fuel Injector FD 14324A BII

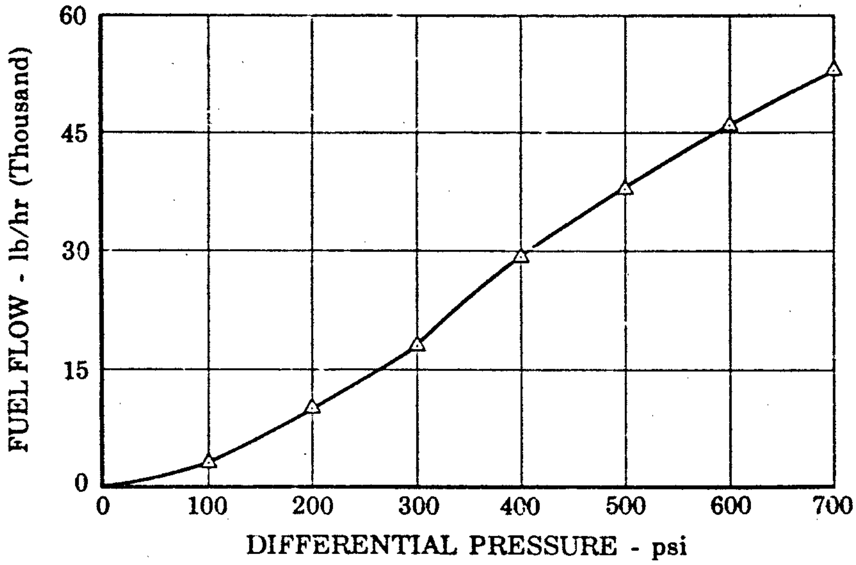

Figure 19. Zone II Duct Heater Total Fuel Flow FD 16248 BII

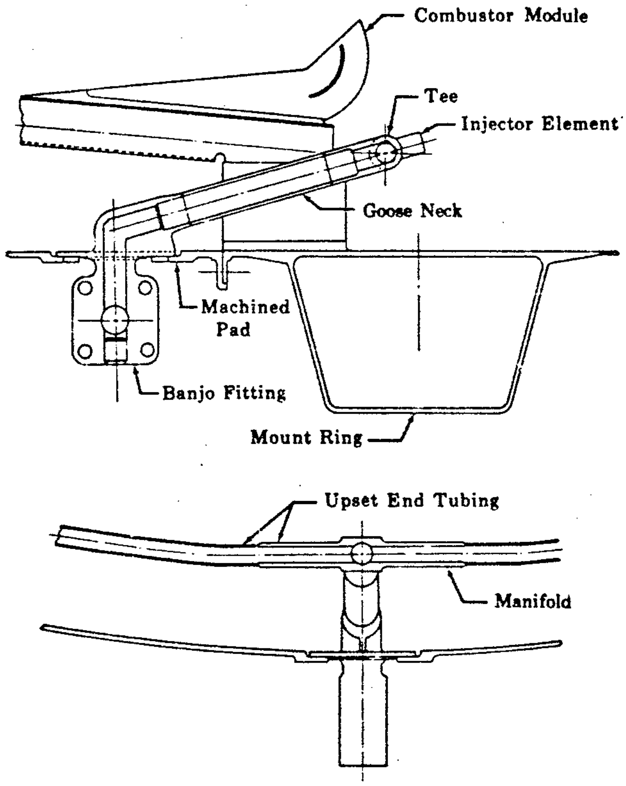

Figure 20. Zone II Fuel Manifold FD 16247 BII

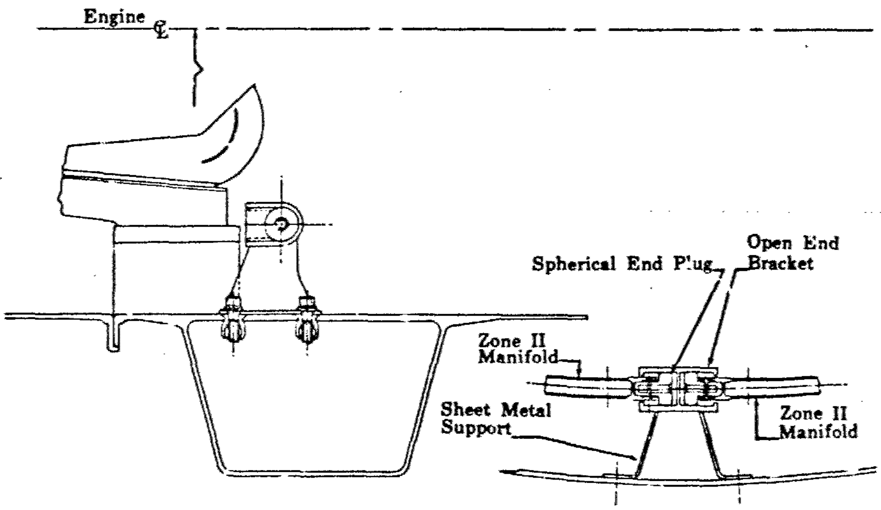

Figure 21. Zone II Fuel System OD Bracket FD 17051
BII

The inner manifolds are similar to the outer manifolds (figure 22). Each consists of the inlet fitting, 3/8-in. tubing, injectors, and end plugs. The fuel is supplied to the inner manifolds by tubes routed through the combustor support case struts at four locations. The internal feed lines are then routed along the inner fan duct case to fittings that project through machined pads in the duct wall. The manifold segments are bolted to fittings from inside the fan duct. Similar brackets are then used at the segment ends.

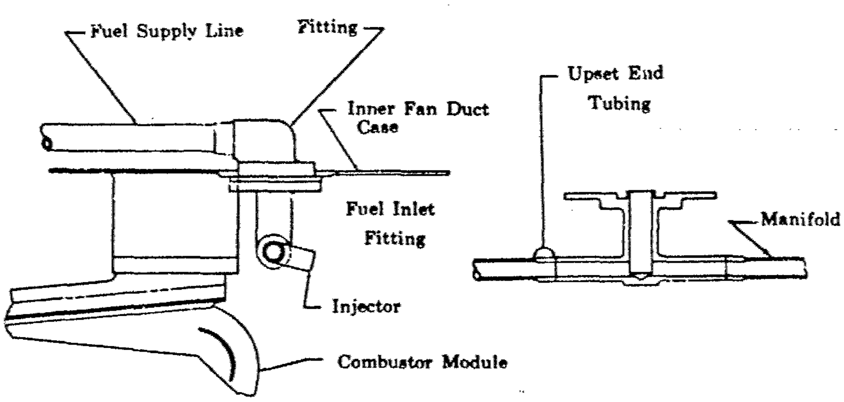

Figure 22. Zone II Fuel Manifold ID FD 16245
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The selection of materials for the Zone II fuel system is based on successful J58 manifold experience.

(5) Combustors and Liners

The Zone I portion of the duct heater is of a ram-induction design. Scoops with turning vanes, swirlers, and slots, direct air into the combustor. A Zone II fuel system adds fuel to the Zone I bypass air, some of which passes through 90-degree turbulators to provide mixing. (See figures 1 and 11.)

The duct heater overall burning length, from the Zone I fuel nozzles to the duct exhaust nozzle, is the same as that in the initial experimental engine. However, the JTF17 engine Zone I combustor is 6.8 inches shorter while Zone II is longer (by 6.8 inches).

The JTF17 engine Zone I combustor has two sets of scoops, staggered axially. The rear set of scoops has been overturned 30 degrees to inject air forward and toward the dome. The overturned scoops improve circulation and result in good mixing of fuel, air and fuel/air vapors. Furthermore, test results with overturned tubular scoops in the Model J combustor have shown increased fuel/air operational limits as well as improved combustion efficiency. (See figure 13.)

A geometric scaling method developed by P&WA was used in sizing the initial experimental engine duct heater, as well as that for the JTF17 duct heater. The open-area distribution of the Model E-9 was used as the basis for design of the Zone I JTF17 combustor. (See figure 23.)

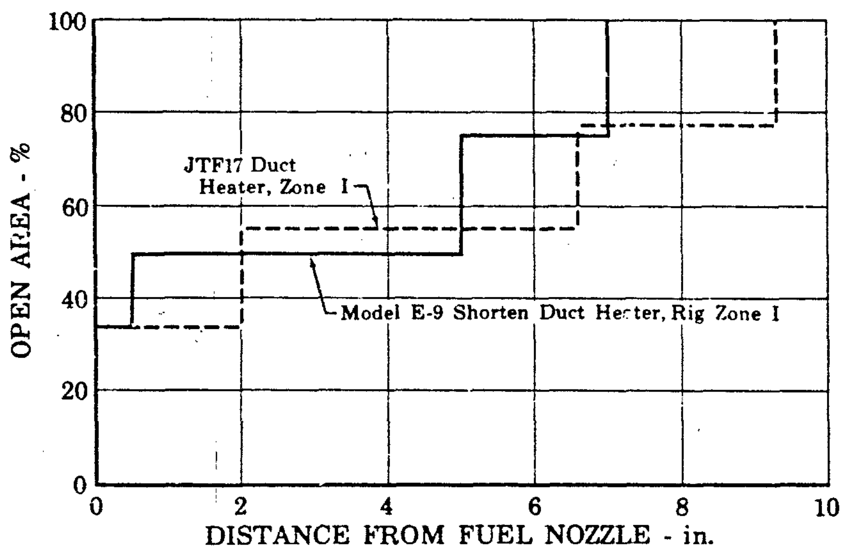

Figure 23. Zone I Combustor - Percent Open Area vs Burning Length From Fuel Nozzle — FD 16239 BII

Figure 24 shows the exit temperature profile of the full annular duct heater during Zone I operation only. The temperature profile is peaked in the center. Improved mixing of burner bypass air with the combustion gases of Zone I would flatten the temperature profile and improve thrust equivalent combustion efficiency.* In the JTF17 a set of 90-degree turbulators (rather than the 35-degree turbulators) has been mounted immediately behind the Zone II fuel manifolds to increase mixing of the Zone II air with Zone I combustion gases, and therefore flatten the temperature profile.

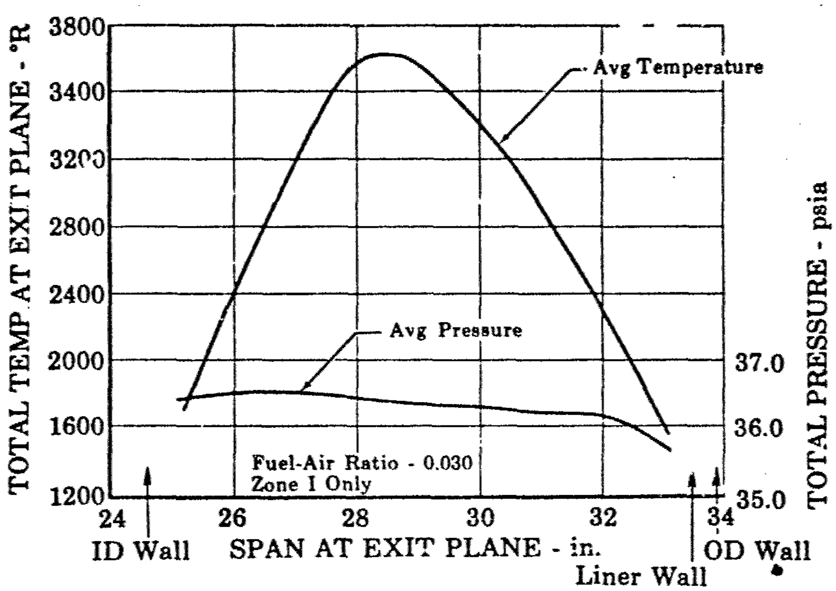

Figure 24. JTF17 Full-Scale Duct Heater Rig Radial Temperature

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The combustor is designed to have good maintainability by utilization of a module concept. (See figure 25.) Two scoops and a base plate make up one module, which can be individually removed while leaving the inlet fairing and Zone I fuel nozzles intact. The inner and outer walls of the combustor consists of 80 total modules.

The pressure loads on the modules are transmitted into the cooler and stronger inner- and outer-duct cases, thus eliminating separate structural rings. With the individual module concept, the scoops can be weld-repaired or the module replaced individually.

The modules (figure 25) are shaped like circular arcs, so that the pressure loads are resisted by hoop stresses in the module base plates. The scoops act structurally like square-cross-section beams providing longitudinal stiffness to the module base plates.

*Thrust equivalent combustion efficiency is explained in Report A, Section III, paragraph D.

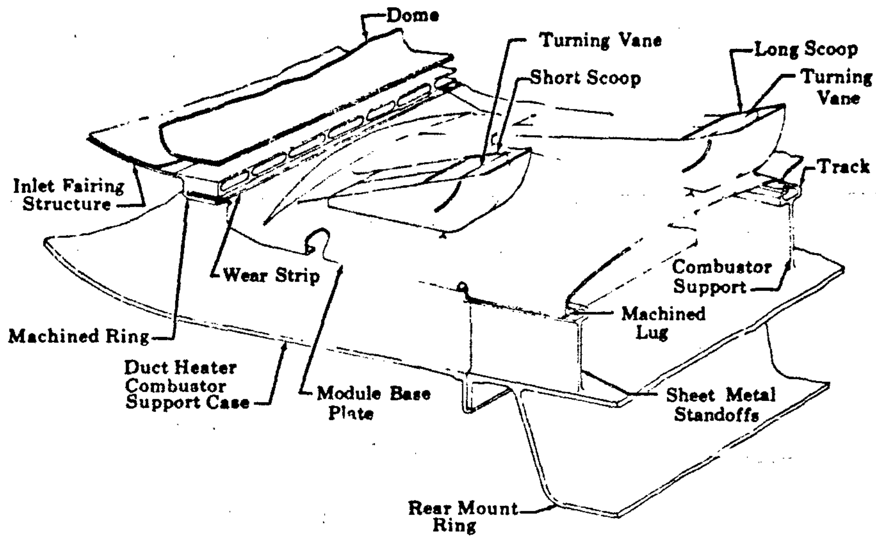

Figure 25. Duct Heater Combustor Module FD 16408 BII

A module base plate is made from a sheet metal stamping. To each base plate two scoops are butt-welded to raise "footprints" drawn during the forming operation. The scoops are made from two stampings that are butt-welded together. The scoop turning vanes are nicrobrazed (per Specification AMS 2675) in place. Machined lugs are butt-welded to the turned-down sides of the base plate. A wear strip is seam-welded to the forward edge of the base plate. The lugs and wear strips are made of Haynes L605 because of good wear characteristics proven in J58 engine operation. The mating surface on the inlet fairing and dome assembly, and the inner and outer cases are made of AMS 5759 (Haynes L605).

AMS 5536 (Hastelloy X) sheet was selected for the scoop and module base plate because of its excellent oxidation resistance, good creep properties at eleveated temperatures, and availability. The use of fan discharge air for cooling maintains metal temperatures at a low operating level. This air is approximately 900°F cooler than the J58 afterburner cooling air at cruise conditions. Rig testing indicated module base plate temperatures of about 1350°F and maximum temperatures at the scoop tips of about 1500°F.

The rear supports for the modules consist of 40 tracks mounted on sheet metal standoffs that are butt-welded to attachments machined into the inner and outer cases. The rear supports are readily repairable because the tracks are butt-welded to sheet metal standoffs. The modules are inserted so that the front portion engages the machined ring on the inlet fairing and the lug slides into the tracks; the modules are retained by pins. Figure 25 shows the outer module configuration.

The duct heater outer liners, made of AMS 5536 (Hastelloy X), serve two primary functions. First, being convectively cooled thermal barriers, they contain the high temperatures of the combustion gases. Secondly, they provide acoustic absorption, which stabilizes combustion and reduces engine noise.

Acoustic absorption is accomplished by the forward portion of the outer cooling liner. This liner is composed of 60 pressure-loaded catenary segments illustrated in figure 26. The acoustic liners are perforated with 0.2-inch diameter by 0.3-inch long standpipes. These perforated liners are separated by a volume of gas from the engine cases. The mass of gas in the liner apertures and the volume of gas in the cavity behind the liners form an oscillatory system that is analogous to a spring-mass system, which has a frequency response like a Helmholtz resonator. At low incident pressure wave amplitudes, energy dissipation (and therefore sound absorption) is predominantly caused by viscous losses occurring as a result of oscillation of the gas in the liner apertures. At larger amplitudes, turbulence and circulation caused by higher particle velocities in the liner apertures effectively control the absorption characteristics. Figure 27 illustrates the absorbing liner principles.

The variation of the absorption coefficient with frequency for the JTF17 engine liner is shown in figure 28. The liner absorption coefficients are higher in the region of fan-blade-passing frequencies than in the combustion instability frequency range.

The module concept is also used for the outer liner. As in the combustor, each module is made of a sheet metal stamping. The modules are retained in tracks, which are welded to the duct outer case. Machined lugs are butt-welded to the edges of the modules (figure 26) forming longitudinal beads, which fit into the tracks.

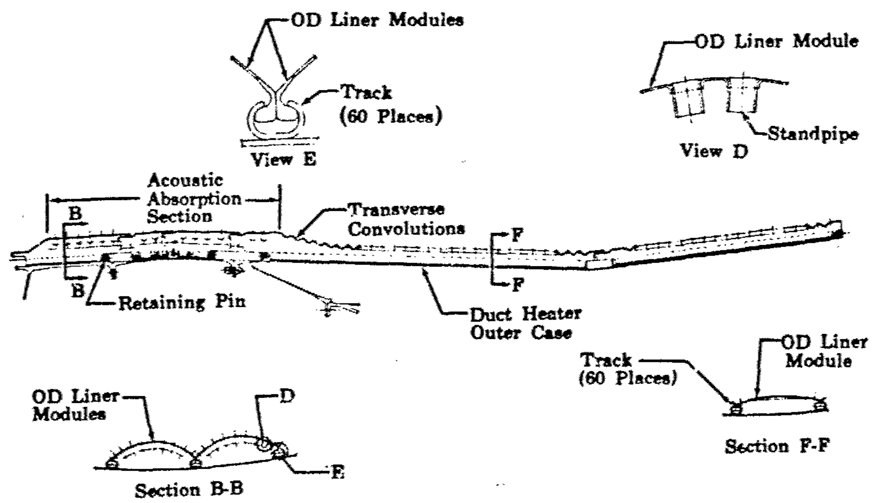

Figure 26. Outer Liner Module

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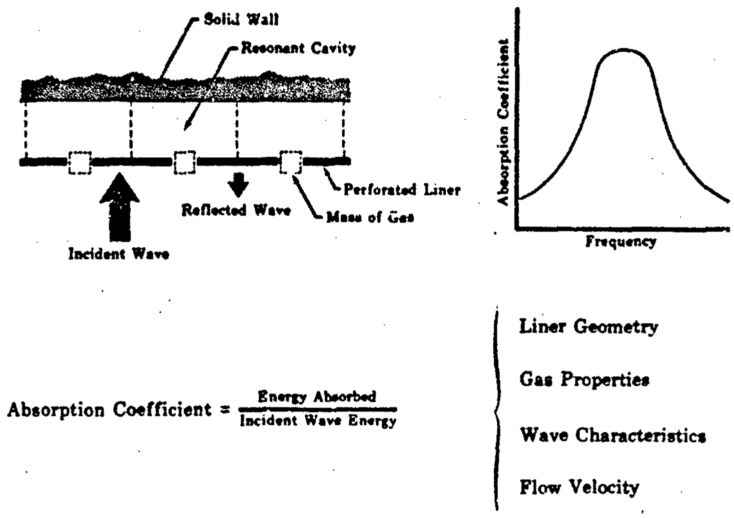

Figure 27. Absorbing Liner Principles FD 7363A BII

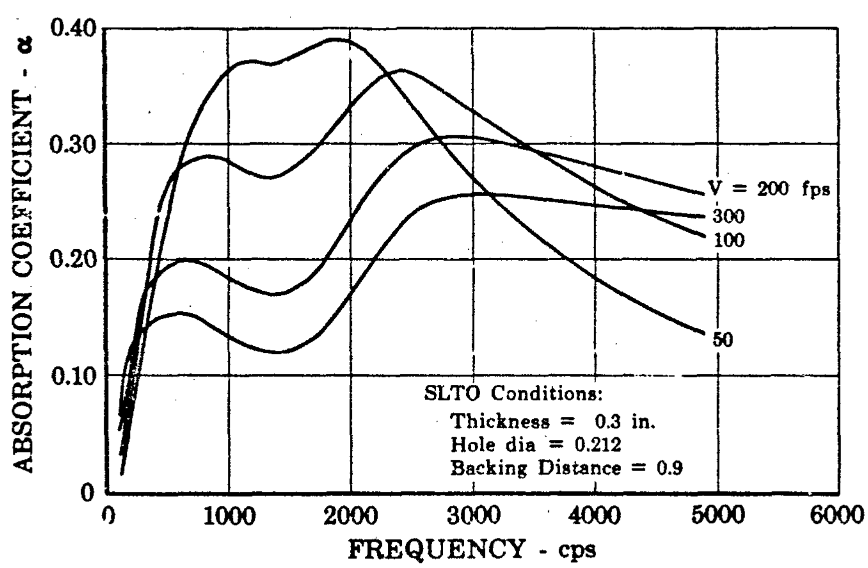

Figure 28. Variation of Absorption Coefficient With Frequency as a Function of Velocity for the JTF17 Duct Heater FD 162[illegible]1 BII

The liner modules have transverse convolutions (see figures 11 and 26) to provide longitudinal flexibility and to relieve thermal stress. Because of the modular construction concept, a 1.0% creep design criterion is used. Axial movement of the liners is restricted by pins installed in the ends of the tracks. The modules are interchangeable and offer all the maintainability advantages of the duct heater and primary combustor modules. These are (1) low cost, replacement, and repair; (2) easy handling; and (3) compact storage of spares.

The inner liner, which is made of AMS 5536 (Hastelloy X), forms a duct heater inner flowpath wall. The limiting criterion for the inner liner is 30% buckling margin for the pressure loads. This criterion is based on previous production engine experience. The inner liner uses film cooling slots similar to all P&WA production engines.

The metal temperature predictions, pressure levels, and airflow distributions for all duct heater components are shown in figure 29. The duct heater outer case metal temperatures are presented as a function of case heat rejection for the flight points specified in the RFP. This method of tabulation is utilized because of airframe nacelle variations.

b. Product Assurance Considerations

Specific maintainability, reliability, and value engineering advantages are listed below. These advantages were discussed in more detail within the design approach section.

(1) Maintainability

1. Servicing of the gas generator is accomplished by 8 access ports accessibly located between external components and plumbing.

2. The use of quick-release fasteners in the diffuser inner wall assembly reduces the time required for servicing of the gas generator.

3. Quick assembly or disassembly of the duct heater as a unit is provided by sliding the duct heater rearward on the radial square spline of the gas-generator aft support.

4. The replacement of the fuel nozzles in Zone I can be accomplished without removal of the combustor.

5. The replacement of the fuel injectors in Zone II can be accomplished individually.

6. The inner and outer combustor modular construction approach permits modules of Zone I to be replaced individually without removal of Zone I fuel nozzles.

7. The modular construction of the outer liner permits Zone II modules to be replaced individually.

(2) Reliability

1. Diffusion rates for the diffuser are based on P&WA commercial engine designs.

2. Butt-welded struts of the type used in the combustor support case have been successfully demonstrated on other P&WA commercial engine cases.

3. All-metal K-type seals, which are used for fuel flowing and nonflowing applications in the fuel system, have been used in the J58 engine with complete success.

4. Pressurizing valves are incorporated in the fuel nozzle housing as close as possible to the discharge orifices to avoid fuel coking.

5. Adequate fuel passage sizes (based on commercial engine experience) are used to ensure minimum susceptibility to contamination and sticking of the variable-area valves in the fuel nozzles.

6. Thermal stresses are minimized by using the modular construction for the Zone I annular combustor; the modules are suspended individually and allowed to grow to relieve stresses when the duct heater is in operation.

7. Component cooling by relatively cool fan discharge air (having a maximum temperature of 660°F), ensures component temperatures at values commensurate with long life.

8. Butt-welded construction throughout the basic structure of the diffuser, combustor support case, combustor scoop modules, and Zone II fuel manifolds facilitates inspection and quality control and provides reliability and structural integrity of the assemblies.

9. Full-scale test results of the duct heater combustion system demonstrated ignition system reliability.

10. Zone I fuel nozzle strainers are located as close as possible to the nozzle orifices to prevent contamination.

11. Zone I fuel nozzle strainers are retained in the nozzle housing during disassembly of the fuel manifold which protects the fuel nozzles from contamination during handling.

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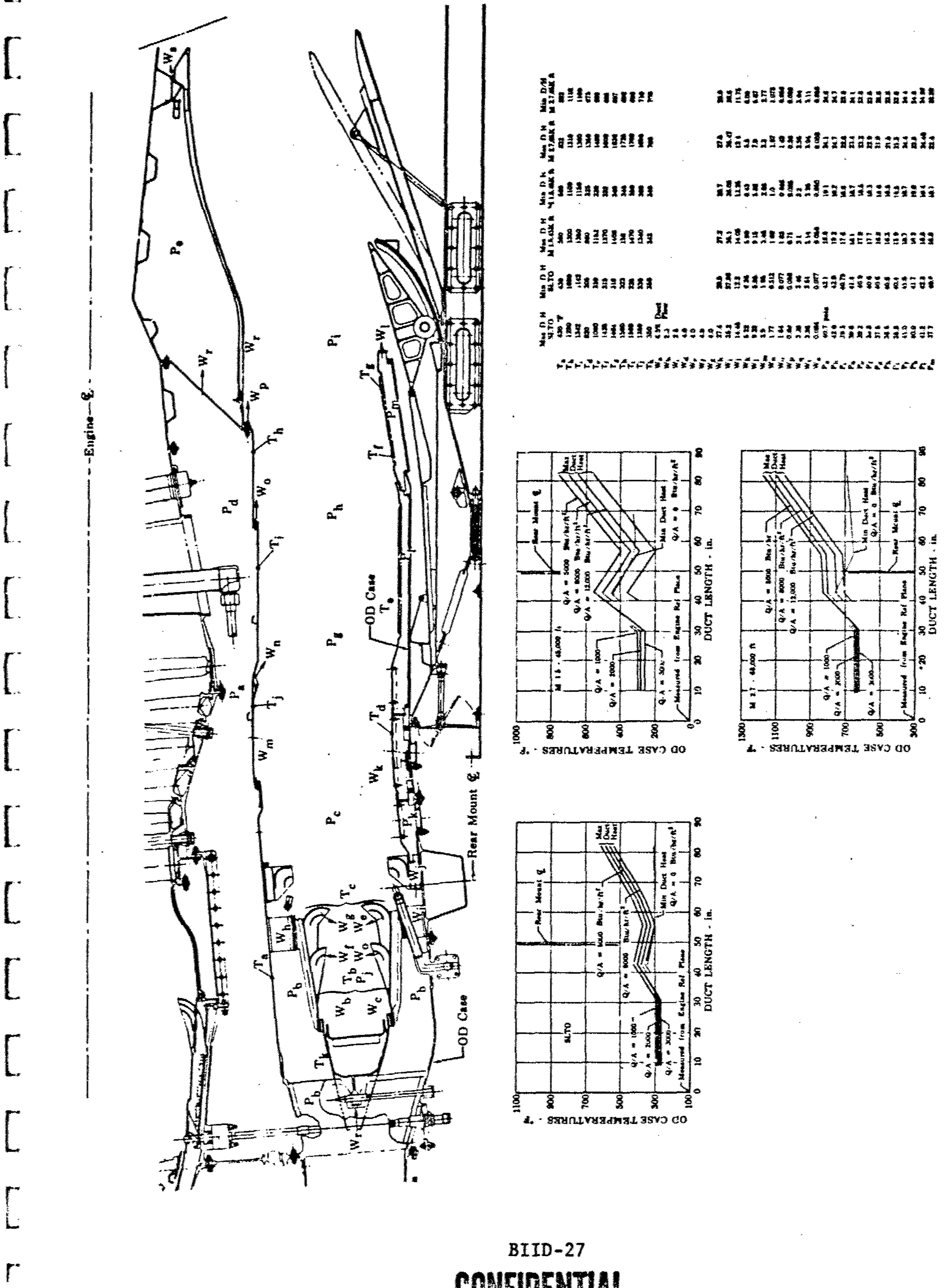

Figure 29. Predicted Metal Temperature, Pressure Levels, and Airflow Distributions for all JTF17 Duct Heater Components

(3) Value Engineering

1. Low cooling air temperatures in the duct heater permit the use of relatively low-cost commercially available materials.

2. Cast swirlers represent a cost reduction over fabricated sheet metal types.

3. Zone I fuel nozzles are identical to the primary combustor nozzles.

4. Modular construction of the Zone I combustor and Zone II outer liner permits replacement or repair of these component parts economically.

5. Ground handling costs are minimized by using modular construction, which simplifies repair or replacement of component parts.

6. Cost and strength-to-weight ratio comparisons were made for all materials chosen for the duct heater.

7. Zone I fuel nozzle housings are castings that result in cost savings over a forged or fabricated configuration.

8. Modular construction of both Zone I and Zone II combustors allows the use of small sheet metal stampings.

c. Materials Summary

Material choices for the duct heater components are based on demonstrated mechanical properties, erosion and corrosion resistance, and service life in previous P&WA production engines.

Diffuser Case	
Sheet Metal Parts	AMS 4910 (Titanium A-110)
Machined Parts	AMS 4966 (Titanium A-110)
Combustor Support Case	PWA 1202 (Titanium MST 811)
Gas Generator Aft Support	
Inner Portion	PWA 1009 (Inconel 718)
Outer Portion	AMS 5616 (Greek Ascoloy)
Inlet Fairing, Dome, and Swirlers	
Sheet Metal Parts	AMS 5536 (Hastelloy X)
Machined Ring	AMS 5759 (Haynes L605)
Swirlers	AMS 5382 (Stellite 31)
Fuel System Zone I	
Nozzle Support	AMS 5362 (Stainless Steel 347)

Fuel System Zone II	
Tubing	PWA 1060 (Inconel)
Machined Fitting	AMS 5665 (Inconel)
Machined End Plug	AMS 5759 (Haynes L605)
Combustor Scoop Module	
Sheet Metal	AMS 5536 (Hastelloy X)
Machined Lugs	AMS 5759 (Haynes L605)
Wear Strips	AMS 5537 (Haynes L605)
Inner and Outer Liners	AMS 5536 (Hastelloy X)

E. EXHAUST NOZZLE

1. General Description

The JTF17 exhaust nozzle system is composed of three basic sections. These are: (1) the gas generator fixed convergent-divergent exhaust nozzle; (2) the coannular duct variable-throat-area plug nozzle; and, (3) the reverser-suppressor section which functions as the final divergent nozzle for both coannular exhaust streams. This section of the proposal will discuss only the first two of these components. The reverser-suppressor is discussed separately in Paragraph F.

2. Objectives and Requirements

The JTF17 exhaust nozzle system is designed to be compatible with the performance requirements of the combined reverser-suppressor nozzle system as defined in Volume III, Report A, Paragraph E. It provides a variable-area duct exhaust nozzle to maintain the desired fan operating point over the full range of duct augmentation, and provides a convergent-divergent gas generator exhaust nozzle for aerodynamic compatibility between the duct and gas generator streams. The design proposed has been selected to provide:

1. Exhaust nozzles that are mechanically and aerodynamically stable in all modes of operation and provide a duct nozzle response rate compatible with engine cycle and safety requirements.
2. Exhaust nozzles that have extended service life and reliability and incorporate maintainability features permitting attainment of these objectives.
3. A system that requires a minimum of external actuation force to reduce size, weight, and complexity of the hydraulic actuation system.
4. A gas generator exhaust nozzle contour that, above idle power settings, provides a choked exhaust nozzle throat and subsequent partial expansion of the gas generator flow.
5. A variable-throat-area duct nozzle that is free of hysteresis, is insensitive to unequal loadings, and that remains concentric with the fixed center plug. This is necessary to provide a repeatable and accurate input into the unitized engine control. This control feedback system is discussed in Volume III, Report B, Section III.
6. The duct nozzle response rates required to maintain control of the fan operating point during rapid changes of duct heater augmentation.
7. A positive aerodynamic opening force on the duct nozzle to ensure its opening in the event of loss of hydraulic power. Failure mode and effect analysis has determined this to be the desired failure mode to provide the largest available engine thrust over most of the operating envelope. The opening force must be sufficient to overcome system friction.
8. Fire safety at all times by ensuring that there is no external leakage around actuator rod seals.

9. Extended service life by using design concepts, materials, and coatings that have been proved effective in equal or more severe service operation.
10. For easy inspection, replacement and/or repair of parts that are subject to wear and exposure to the exhaust gas streams.
11. A nearly "balanced" flap nozzle to permit use of a minimum size and number of hydraulic actuators, a minimum size hydraulic pump, and low hydraulic pressure. This is desirable in order to reduce weight and complexity and to increase system reliability and durability.

3. Design Approach

(a) General

The exhaust nozzle system has been designed to incorporate development and service experience gained from the J58 engine program.

The duct exhaust nozzle relies very heavily on the design and experience gained in the development, production and service use of the J58 engine afterburner nozzle. To date, the J58 balanced flap nozzle has accumulated approximately 20,000 hours of development engine testing. Of this, over 7700 hours have been at conditions that equal or exceed those expected on the JTF17. In addition, a substantial number of service flight hours have been, and are being, accumulated at these more severe conditions.

The gas generator and duct exhaust nozzles are shown in figure 1.

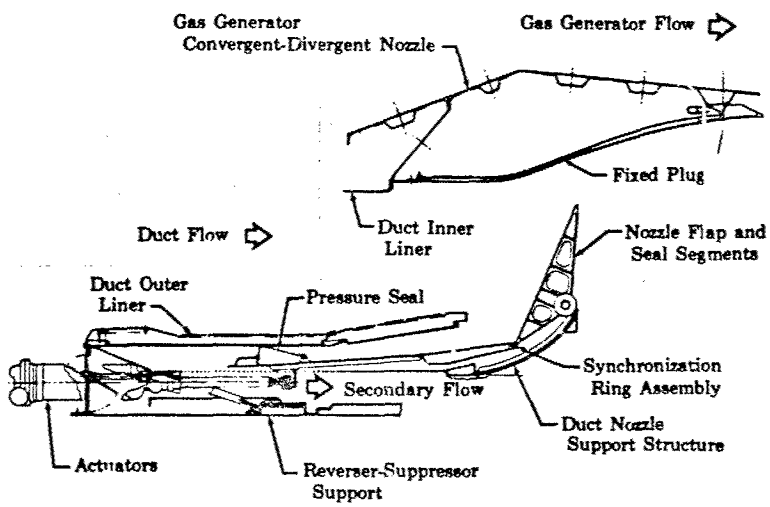

Figure 1. Exhaust Nozzle System FD 16997
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The gas generator exhaust nozzle is a reinforced, one-piece, sheet metal assembly that is formed to create a convergent passage, a nozzle throat, and a short divergent passage.

The duct exhaust nozzle consists of:

1. Main Structure
   The main structure, shown in figure 2, is a one-piece shell weldment with integral tracks to position both the synchronization ring and nozzle flaps.

2. Synchronization Ring
   The synchronization ring, shown in figure 3, is a slightly conical, sheet metal weldment reinforced to provide required stiffness and to distribute actuator loading. The ring is supported in the main structure by adjustable guide rollers.

3. Nozzle Flaps and Seals
   These components are shown in figure 4. Both are precision cast units that are hinge-mounted to the synchronization ring.

4. Synchronization Ring Pressure Seal
   Backflow between the synchronization ring and outer duct wall is prevented by a series of pressure balanced sectors that form a dynamic ring seal as illustrated by figure 5.

5. Plug
   The fan duct nozzle plug is located at the aft end of the duct as shown in figure 1. It is a double wall, convectively cooled weldment and serves as the initial expansion surface for the duct exhaust stream.

6. Actuators and Mounting
   Four hydraulic actuators, as shown in figure 6, are mounted to the nozzle support structure through a simple replaceable trunion as shown in figure 7. These are double acting, continuously cooled actuators that serve to position the nozzle synchronization ring and attached flap and seal segments and thus control and establish the variable nozzle area.

(b) Detailed Description

Gas Generator Exhaust Nozzle

In designing the gas generator fixed exhaust nozzle, three approaches were evaluated: (1) ring-reinforced single wall, (2) double-wall convectively cooled, (3) non-reinforced single wall. Of these three approaches, the ring-reinforced single wall structure proved lightest, and was used for that reason. All approaches were structurally adequate. The double-wall, convectively cooled was the least desirable in weight and cost.

The nozzle structure is straightforward in concept and execution and has no expected development problems. It is based on design buckling margins proved to be sound in J58 externally pressurized structures.

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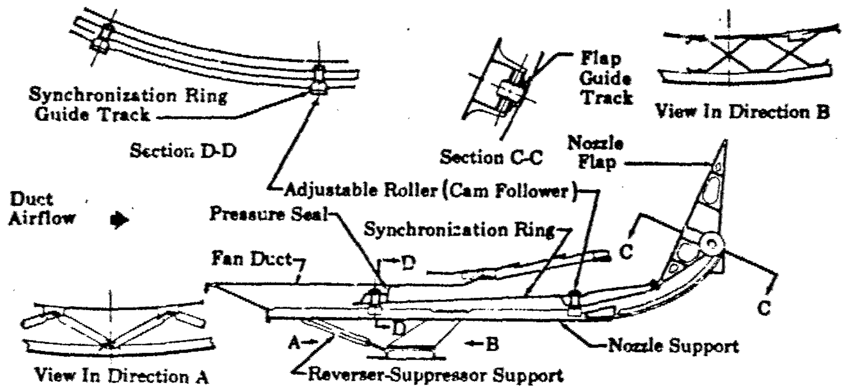

Figure 2. Variable Nozzle Configuration FD 16409 BIIE

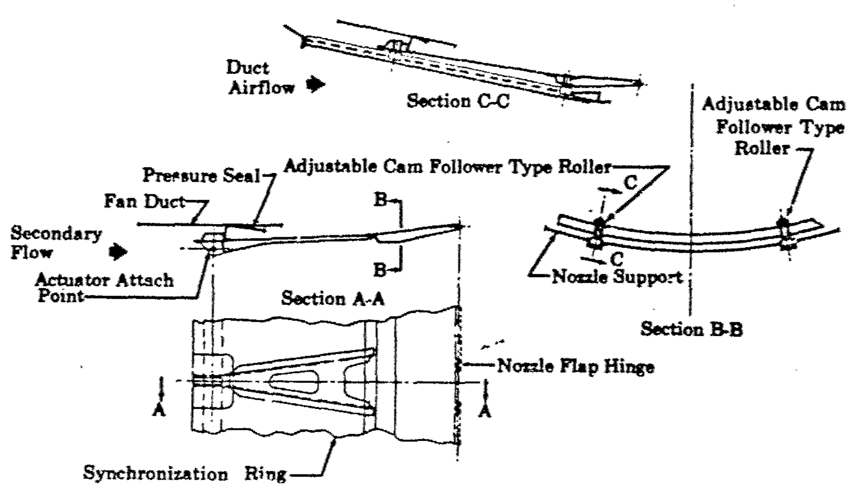

Figure 3. Synchronization Ring FD 16393 BIIE

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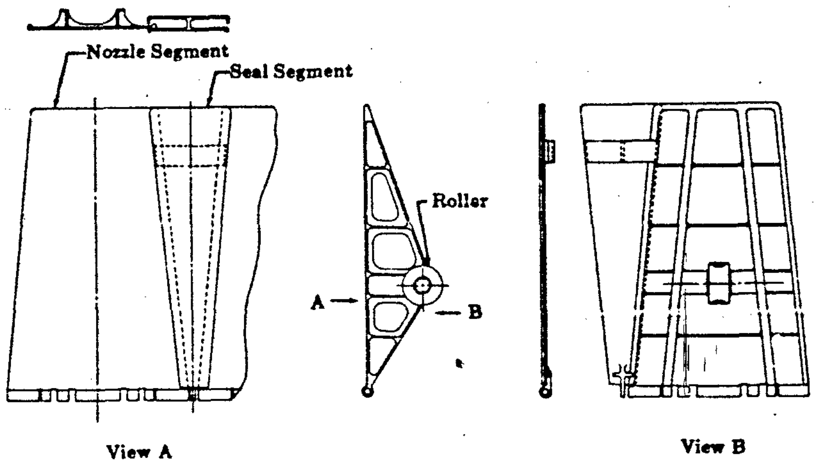

Figure 4. Typical Duct Nozzle and Seal Segments FD 16389 BIIE

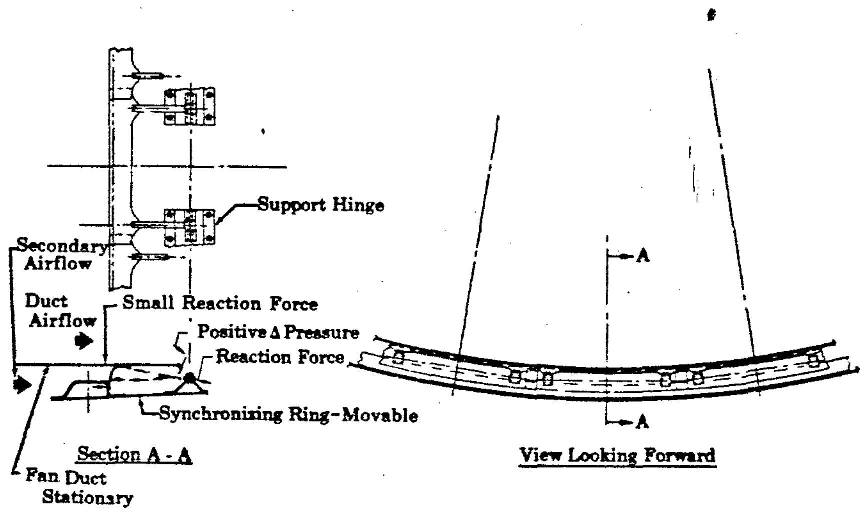

Figure 5. Pressure Seal Arrangement FD 16397 BIIE

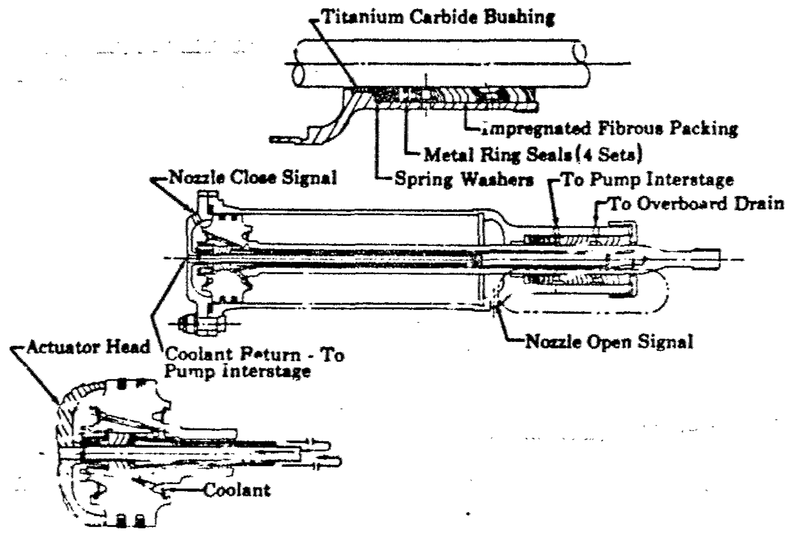

Figure 6. Duct Nozzle Actuator FD 16395 BIIE

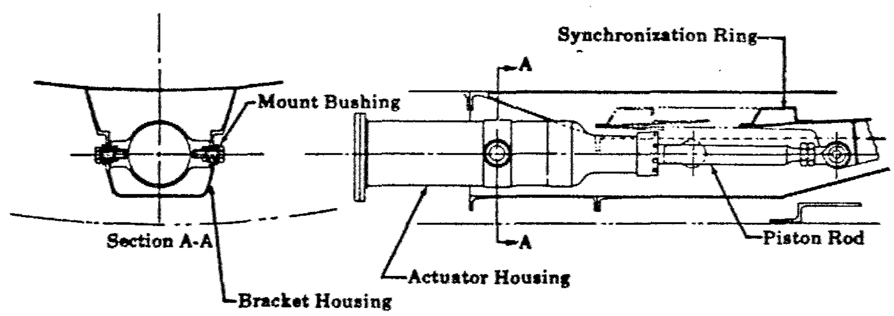

Figure 7. Duct Nozzle Actuator Mounting FD 16396 BIIE

Duct Exhaust Nozzle

The main structure contains integral tracks to support and position the synchronization ring and flap segments. The nozzle flap tracks are contoured to control radial position of the flap tips to form the nozzle variable-area throat. The tracks for the synchronization ring are embedded so that cocking or unbalance loads exerted by the synchronization ring rollers will be distributed tangentially into the structure shell without generating local bending loads that, if present, could result in a shortened fatigue life.

To provide for long wearing surfaces, both the synchronization ring support tracks and the flap segment positioning tracks are fitted with wear resistant liners. Three methods of attachment are being investi-

gated: (1) co-extrusion of liner and tracks, (2) diffusion bonding, (3) mechanical attachment. In addition, the use of various flame deposited coatings will be evaluated for this application.

The lattice structural support for the reverser-suppressor was selected to provide a distributed load path around the nozzle and reverser-suppressor structure and at the same time provide a minimum flow area of 500 sq. in. for secondary airflow. This airflow becomes the cooling medium for the reverser-suppressor during cruise operation.

Synchronization Ring

This ring structure is patterned after that used in the J58 engine. The adjustable roller supports are the same as used in the J58.

Nozzle Flaps and Seals

Both precision cast units closely follow J58 design practice. Both are hinge-mounted to the synchronization ring and are loosely held together and to the structure tracks during non-operating periods by an integral clip arrangement. During operation, aerodynamic loading forces the seal against the flap and the flap roller against the structure. The many hours of J58 operation have demonstrated that PWA 47 coating is an effective wear preventative in the flap and seal overlap areas.

Analytical studies indicate that the maximum differential pressure loading on the variable flaps is 36.7 psi at Mach 2.7 and 53,500 ft. altitude. The force balance per flap for this condition is illustrated in figure 8. Note that the center of pressure of the flap is offset from the roller reaction point to furnish a positive unbalance force to open the nozzle in the event of hydraulic system failure. This force varies in magnitude with operating condition, but remains consistent in direction.

The maximum calculated force required to move the nozzle at this maximum load condition is 14,400 pounds. This results from a combination of pressure differential and system friction. The force available from each of the four actuators at a piston ΔP of 1100 psi is 4400 lb for a total of 17,600 lb. The required nozzle area variation from 3 to 12 square feet is provided for the full range of nonaugmented to full augmented operation.

The highest temperatures in the variable nozzle do not occur at full augmentation conditions where the nozzle is full open, but rather at a lower level of augmentation where the nozzle is less than full open and the exhaust stream impingement angles are greater.

Cooling is accomplished using fan discharge air as the cooling medium. Figure 9 is a schematic of the boundary conditions at the most severe operating point (Mach 2.7, 68,500 ft partial D/H). The corresponding metal temperatures are shown in figure 10.

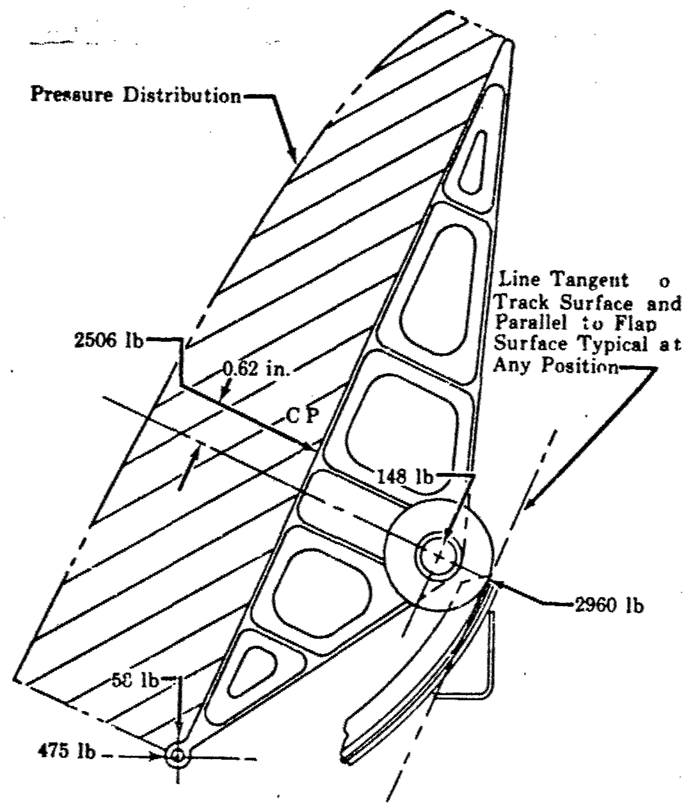

Figure 8. Nozzle Flap Loading

FD 16392
BIIE

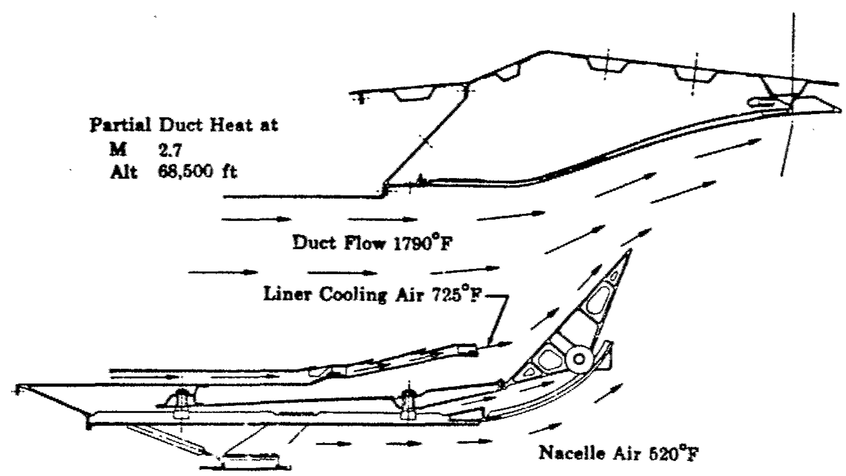

Figure 9. Nozzle Flow Schematic

FD 16881
BIIE

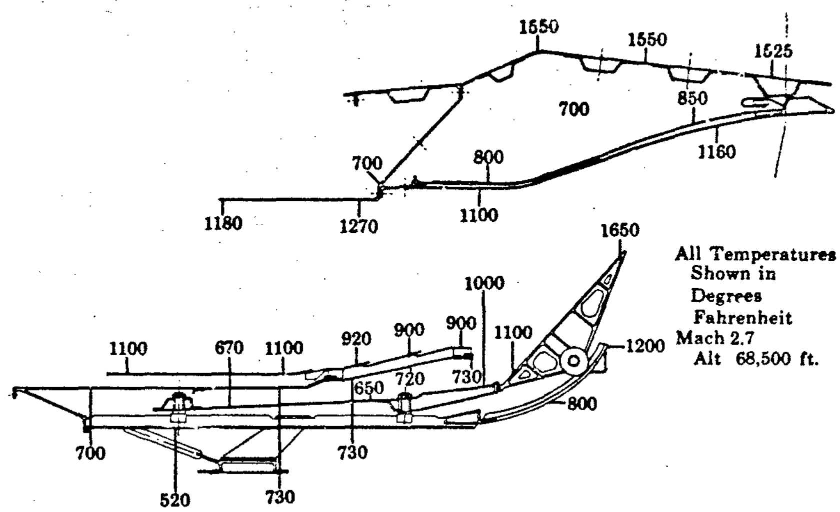

Figure 10. Metal Temperature at Cruise With Partial Duct Heat

FD 16882
BIIE

These temperatures are predicted on the basis of J58 temperature calculations and test data correlations. Actual temperature measurements of J58 flaps have confirmed their predicted operating temperature of 1300-1900°F. Using the same prediction methods, the JTF17 flaps will operate approximately 250°F cooler.

Nozzle Plug

Both film cooled and convectively cooled structures were studied for this component. The double wall, convectively cooled unit was selected because of its more efficient use of cooling air and because it provides a smooth contour in the supersonic convergent-divergent nozzle. The cooling annulus is approximately 1/4 inch in height. Maximum temperatures on the inner plug occur at full augmentation and are shown in Table 1.

Table 1. Maximum Plug Wall Temperatures, Cooling Flow, and Coolant Static Pressures at Various Flight Conditions

FLIGHT CONDITION	MAX. WALL TEMP. °F	COOLING FLOW % DUCT AIRFLOW	COOLANT STATIC PRESS. psia
Sea Level Static	1500	3.13	36
M 0.5-15,000 ft. altitude	1500	3.0	25
M 1.5-45,000 ft. altitude	1550	3.13	15
M 2.7-65,000 ft. altitude	1700	3.17	22

Nozzle Actuation

The nozzle actuators incorporate cooling features and rod sealing concepts that were developed for, and are used in, the J58 engine at higher temperatures (Mach 3+) and pressures (2500 psi in J58, 1500 psi in JTF17). The rod sealing concept subjects the fibrous packing seals

only to low pressure. The internal actuator pressure is reduced before contacting the seals by a series of metallic piston rings. The leakage flow across these rings is routed back to main fuel pump interstage cavity pressure of approximately 150 psi.

Any leakage through the initial packing set (150 psi ΔP) is routed to the overboard drain system. This concept provides fire safety by ensuring that no fuel will leak into the engine compartment through the rod seals. As previously mentioned, the constant circulation of fuel throughout the actuator housing and rod will prevent localized overheating of fuel or metal and prevent thermal shocking of the seal package.

The net hydraulic pressure across the actuator piston is 1100 psi while the actuator is moving, and 1500 psi while stationary. These pressures provide the required area change response rate of 12 square feet per second.

The actuators are supplied with a constant cooling flow of fuel through the housing and through the length of the actuator rod as shown in figure 6. The most severe metal and fluid temperatures predicted are indicated in figure 11. Temperature predictions are based on correlated J58 calculations and data.

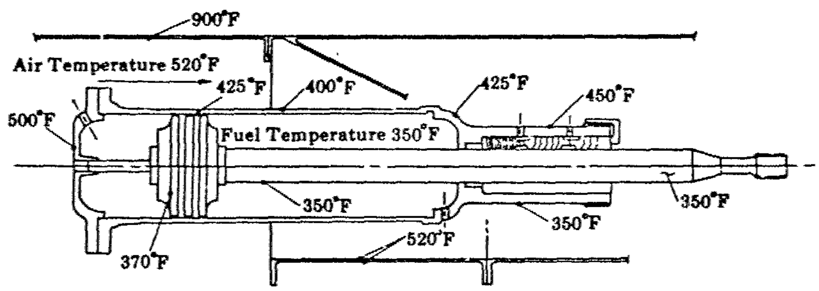

Figure 11. Maximum Duct Heater Nozzle Temperatures — FD 16391 BIIE

As with the variable nozzle, the actuator design relies heavily upon J58 development experience. Design features and material selections have been retained. The cooled rod design is a significant factor in obtaining extended service life of the actuator seals and ensuring fire safety.

During Phase III, studies will be made of the feasibility of using titanium actuators. This will require the development and use of either wear resistant liners or development of wear resistant coatings for the cylinder bore. Both approaches will be evaluated.

(c) Phase II-C Exhaust Nozzle System

The duct exhaust nozzle described in the Phase II-B report was similar in function, but differed in mechanics, from the variable exhaust nozzle defined in this report. The Phase II-B report defined a "translating shroud" reverser-suppressor. The nozzle assembly was designed to be compatible with this particular reverser-suppressor.

In that design the variable area was achieved by rotating the flaps and seals about a fixed hinge point through the use of a system of linkages. That particular nozzle design was completed and tested on the JTF17A engine and used during the initial test program. The nozzle performed satisfactorily and met all design requirements. Later Phase II-C testing will be accomplished with the "balanced flap" nozzle system; this testing will be summarized in the final Phase II-C report.

(d) Material Selection

Material and coating selections are identical to those used in the J58 engine for comparable parts. The exceptions are the use of titanium for the main structure and synchronization ring because of a less stringent operating environment.

Titanium was selected for these components based on successful use in the YF12 and SR-71 airframe-mounted ejector for the J58 engine. The lower operating temperatures predicted for these JTF17 components permit the use of this lighter weight material. J58 temperature predictions have been correlated with test data and instill a high level of confidence in JTF17 Predictions.

The plug nozzle materials selected were based on use of these materials in J58 combustion chambers (Hastelloy X) and transition duct (L-605 with PWA 53-21 coating). These materials and coatings provide proved durability and oxidation resistance at high temperature conditions.

MATERIALS SUMMARY

	Part Name	Material	Specification
1.	Gas Generator Exhaust Nozzle	Waspaloy	AMS 5544
2.	Duct Exhaust Nozzle Flug	L-605	AMS 5537
	and Coating	Nickel-Aluminum	PWA 53-21
3.	Plug Liner	Hastelloy X	AMS 5536
4.	Synchronization Ring	Titanium	AMS 4916
5.	Synchronization Ring Pressure Seal	Titanium	AMS 4916
6.	Nozzle Actuator Housing	Greek Ascoloy	AMS 5616
7.	Actuator Piston and Piston Rod	Greek Ascoloy	AMS 5616
8.	Piston Rod Hardface	Tungsten Carbide	PWA 46
9.	Piston Rod Guide Bushing	Titanium Carbide	
10.	Piston Rings	Cast Iron	
11.	Flaps and Seals	Inconel 100	AMS 5397
12.	Nozzle Flap Rollers	Waspaloy	AMS 5709
13.	Roller Shaft	Inconel 718	PWA 1016
14.	Coating-Flap and Seal	Aluminum-Silicon Coating	PWA 47
15.	Flap Cam Track	Titanium	AMS 4916
16.	Flap Cam Track Liner	L-605	AMS 5759

4. Product Assurance Considerations

(a) Maintainability

The nozzles have been designed to provide for ease of maintenance as illustrated by:

1. The complete duct exhaust nozzle can be removed or installed as a unit.
2. Nozzle assembly and adjustment may be accomplished on the engine or as a separate sub-assembly and this setup is not affected by installation or removal of the complete unit.
3. Adjustments are provided to facilitate attachment of the actuators to the synchronization ring and to permit removal of backlash (or looseness) in the synchronization ring rollers.
4. Individual component parts (flaps, seals, rollers, actuators) are replaceable without machining, welding, selective assembly, shimming or ground spacers being required.
5. Actuator trunions are fitted with replaceable bushings.
6. The duct exhaust nozzle plug is removable without any additional engine disassembly. It is necessary to remove this plug only to remove the gas generator exhaust nozzle assembly.

(b) Reliability

Reliability and extended service life are ensured by:

1. Using fan discharge air as the basic cooling medium. This permits operational metal temperatures well below those already proved in J58 development and service experience.

2. Blow-off and radial aerodynamic loadings on the flaps are resisted by tangential and axial stresses in the basic fixed structure and not by high hydraulic forces or by localized bending loads which could shorten fatigue life.
3. Adjustable synchronization ring rollers remove looseness and backlash from the system and provide accurate nozzle positioning and feedback to the control system.
4. The synchronization ring, rollers, and support are designed to withstand stalled actuator cocking loads without overstress of any part.
5. Low actuation forces that result in a hydraulic system with low pressure requirements, and low-stress actuator mounting and attachment points.
6. Using proved J58 actuator rod seal concepts that prevent full pressure drop across any one set of seals and subject non-metallic seals to low pressure differentials only.
7. Continuous cooling flow circulation of fuel through the actuator housing and rod. This prevents stagnation and overheating of the fuel and also prevents heating of the actuator rod with subsequent thermal shock to the rod seal package when extended and retracted.
8. Wear resistant liners for the flap tracks and synchronization ring tracks.
9. Wear and oxidation resistant coatings for the actuator rod, seal, and flap surfaces, and plug outer surface.
10. Wear resistant materials for the actuator, rod end bushings, and flap and synchronization ring rollers.

(c) Safety

Flight safety features that have been incorporated are:

1. Use of low temperature cooling air.
2. Rapid nozzle response rates to prevent fan overspeed, fan over-pressure and surge, or nozzle overtemperature.
3. Positive leakproof actuator sealing based on concepts proved under more stringent conditions.
4. Unbalance to open in event of hydraulic loss.
5. Use of fuel as hydraulic fluid to simplify the system by the elimination of hydraulic system coolers and a limited capacity tank.

F. REVERSER-SUPPRESSOR

1. Description

The JTF17 reverser-suppressor, shown in figure 1, consists of a main structure, floating tertiary doors, 3-position rotating clamshells, and floating trailing edge flaps.

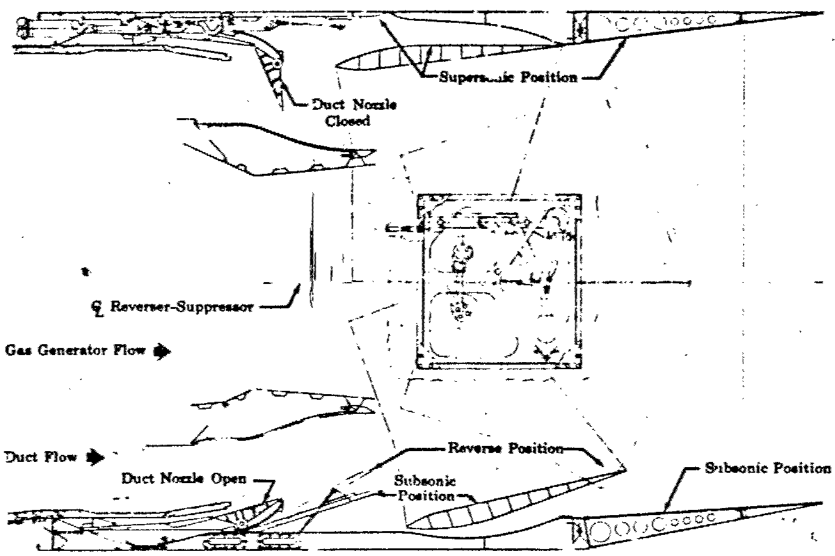

Figure 1. JTF17 Reverser-Suppressor FD 16416 BIIF

This unit forms the final expansion section of the exhaust system for the coannular duct and gas generator exhaust streams and also provides for redirecting and targeting exhaust gas flow for thrust reversal, for airframe external nacelle contours, and for suppression of engine exhaust noise. The reverser-suppressor is aerodynamically actuated in all forward flight modes and hydraulically actuated into reverse.

The aerodynamic performance of this system is discussed separately in Report A of this Volume. Noise suppression is discussed in Report C.

2. Design Objectives and Requirements

The design selected provides an exhaust system that will meet or exceed performance objectives in both forward and reverse flight modes, will provide flight safety in all modes of operation, and will possess extended service life potential. The components of the reverser-suppressor have been designed to provide:

1. The variable exit area of the exhaust system that is necessary to efficiently utilize the variations in pressure ratio produced during different portions of the flight regime
2. Internal and external flow contours that will permit attainment of the performance goals required of the exhaust system to meet engine performance objectives

3. The required exit-area variations in a design that will be aerodynamically and mechanically stable in all modes of operation
4. For reverse flow of the exhaust gases during reverse operation with targeting compatible with the airframe installation
5. Actuation rates for the reverser that are consistent with control system requirements
6. Exhaust noise suppression
7. Extended service life and reliability by utilizing design concepts, maintainability features, and materials that have proved effective in equal, or more severe, service operation.

3. Design Approach

a. Main Structure

The semimonocoque basic structural shell and internal structure of the reverser-suppressor, illustrated in figure 2, is fabricated from titanium for minimum weight. Major stresses are limited to compressive and shear buckling. The primary support panels are simple box-beam type construction. Both skins are buckling-limited in compression because both positive and negative bending moments are imposed. The aft cylindrical portion of the reverser-suppressor incorporates an inner and outer stressed skin structure that reduces bending moments in the support rings and bulkheads by providing shear panels on the inner and outer diameters. The complete unit can be removed from (or installed on) the engine by removing (or installing) 34 bolts and disconnecting (or connecting) three hydraulic lines and the power lever interlock.

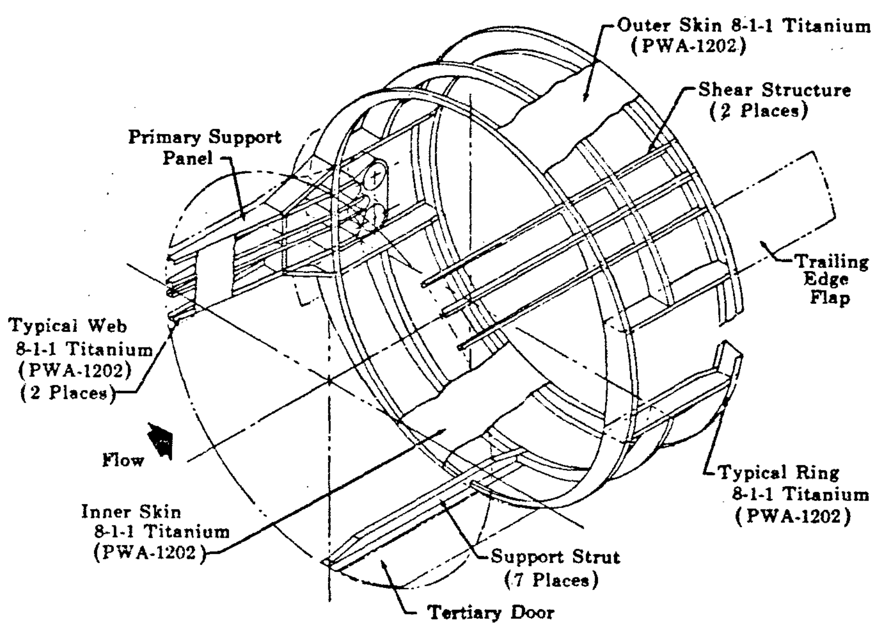

Figure 2. Reverser-Suppressor Main Structure

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BIIF

b. Tertiary Doors and Synchronization System

The 11 double-hinged tertiary doors, which are aerodynamically actuated, admit tertiary air into the reverser-suppressor during subsonic flight to prevent overexpansion of the exhaust stream and consequent losses in performance. The tertiary doors, shown in figure 3, are fabricated from titanium honeycomb panels for minimum weight. Resistance-welded honeycomb is used, rather than conventional brazed panels, because of superior fatigue properties and adaptability to the thermal stresses encountered during flight conditions. The greatest temperature gradient occurs during reversing when the doors serve as shrouds, directing the flow of exhaust gases outward and forward to provide reverse thrust. (See figure 12c.)

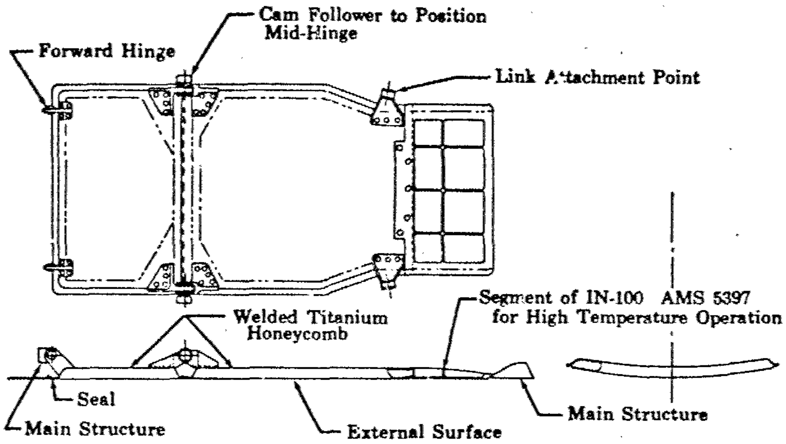

Figure 3. Tertiary Door

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BIIF

The trailing edge of each door is cast from a heat-resistant, nickel-base alloy and is mechanically attached to the door. This tip provides erosion and thermal protection to the door in the area nearest the hot exhaust gases.

At supersonic cruise, an outward pressure difference of about 10 psi across the doors provides a uniformly distributed load on the door, pushing it tightly against the support struts. Seals are provided around the edge of each door and along the hinge line to minimize secondary air leakage and resultant performance loss. The door seal at the hinge allows the door to pivot from the closed position to the maximum of 30 degrees open. The hinge seal, similar at both hinges, is formed to maintain a smooth contour on the outside of the door and is shaped so that the pressure tends to force the seal harder against the mating surface. The seal strips can be easily removed and replaced without removing the door.

All door hardware, such as hinges and brackets, is made of forged titanium and is mechanically attached to the doors. The spherical joint at each pivot provides a hinge that is self-aligning and not subject to binding due to structural deflections or normal manufacturing tolerance. Any hinge section, ball joint, or link attachment can be easily removed or replaced without distrubing the basic system.

The tertiary door interconnecting saddle system, shown in figure 4, consists of a pair of saddles at each support strut. Each pair of saddles is guided by a shared set of rollers in tracks in each strut. Each saddle contains a cam track to support and guide the midhinge of the adjacent tertiary door and is connected to the aft end of the door by a link. Another link, connected to the forward end of each saddle, ties the saddle system to the torque tube system. The torque tube system consists of a torque tube with bellcranks at each end and is mounted between the struts at the leading edge of each door. The saddle system and the torque tube system combine to assure complete synchronization of the tertiary doors with each other through their full range of travel. This system assures that all doors will be opened and locked into position as the system is actuated into reverse. All torque tube mounts and link connection points have ball joints to eliminate binding due to misalignment; this also provides tight connections to reduce linkage end play. The rollers that guide the saddle system and door midhinge are designed to withstand the imposed loads and temperatures without lubrication. All ball joints and rollers are identical, except for detail sizing, to those developed for, and used in, current J58 production engine exhaust nozzles. All saddles, links, and tracks are designed to be easily inspected or replaced while the reverser-suppressor is mounted on the airframe.

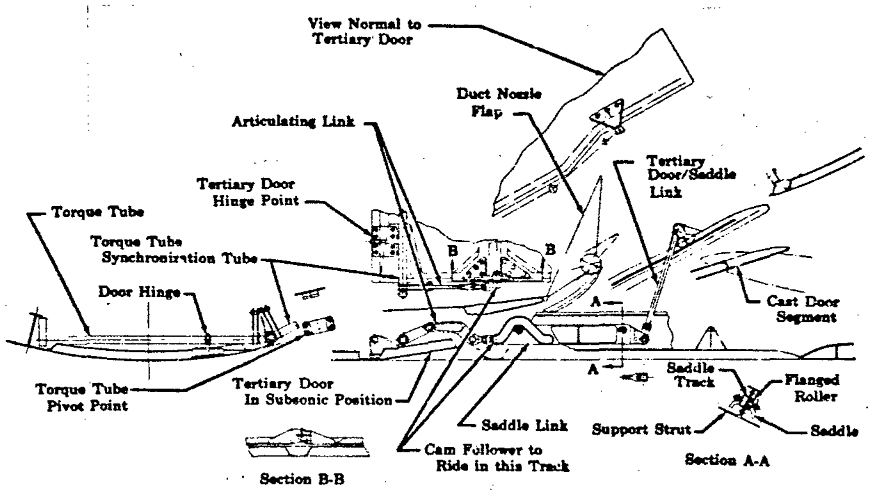

Figure 4. Tertiary Door Interconnecting Saddle System — FD 16858 BIIF

c. Clamshell Mounting and Actuation

The 3-position rotating clamshells form the aerodynamic flow contours during forward flight and, when hydraulically rotated, provide blockage for reverse thrust. The JTF17 engine clamshells are formed by inner and outer skins that are positioned and stiffened by Z- and channel-section members, which are attached to forged pivot supports, as shown in figure 5. The clamshell is fabricated from nickel-base alloy, which is easily welded and requires no subsequent heat-treatment for strength.

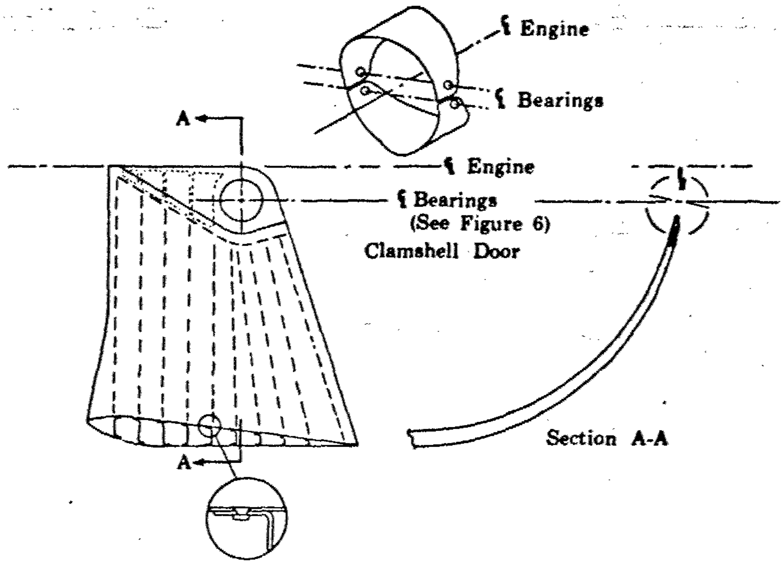

Figure 5. Clamshell Construction FD 16434
BIIF

Each clamshell is supported by, and driven through, two pivots, which are shown in figure 6. The clamshell pivot point is located to pressure balance the clamshell so that minimum size actuators are required for reversing. A double-face coupling is provided between the clamshell pivot and the actuation arm to permit accurate adjustment of the clamshell position relative to the clamshell-tertiary door interlock system (which is described later in this section).

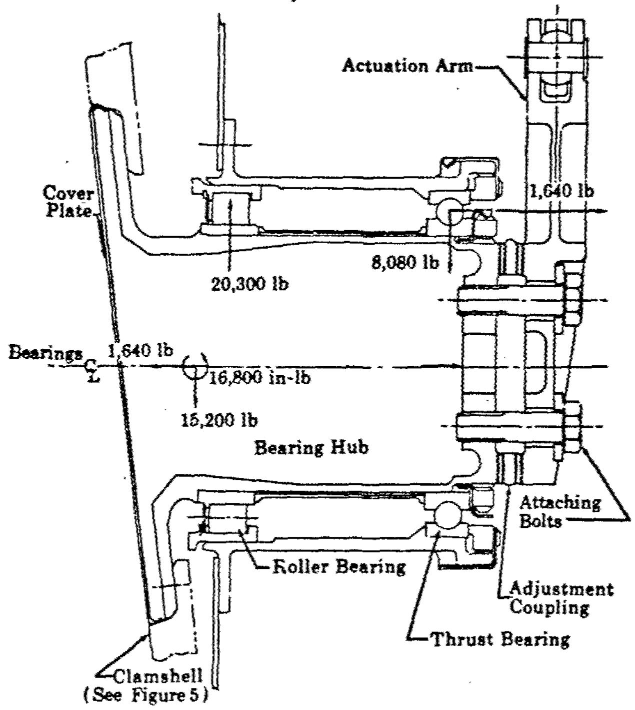

Figure 6. Clamshell Pivot FD 16426
BIIF

Bearings for the clamshell pivot assembly are designed to operate dry at high temperatures (stabilized at 1000°F). The roller bearing has a metallic seal to prevent foreign material from entering the pivot assembly. The compartment enclosing the pivot assembly is maintained at secondary air pressure, which is greater than the static pressure of the exhaust stream, to assure that no exhaust gases leak into the pivot assembly. A sheet metal coverplate is provided over the hub-to-clamshell attachment point to shield the attachment bolts from exhaust gases and to provide a smooth inner clamshell surface. The pivot assembly is similar to that used successfully in JT3D engines in commercial service, except that higher temperature bearings are used and the pivots straddle the horizontal, rather than the vertical, centerline of the engine. This position will prevent accumulation of water, solvents, or fuel in the bearing area during cleaning, operation, or exposure of the engine.

The clamshell hubs and bearings are easily removed or inspected by removing the inside coverplate and outside access panel. The clamshell can be removed or left in place, as desired. All hubs and bearings are interchangeable, thus eliminating matched sets or shims.

Two hydraulic actuators, based on the proved J58 design and shown in figure 7, are used to rotate the clamshells to the reverse position. During all other flight modes the actuators are not pressurized and are supplied with only low-pressure fuel cooling flow. One actuator is located within each of the two clamshell tertiary door interlock modules (which are described later in this section). This location protects the actuators from high temperatures and foreign matter that would tend to shorten the life of the piston rod seals. The actuators are exposed to engine secondary air with a maximum temperature of 700°F. Easy access is provided to the actuators by removing the outer access panel.

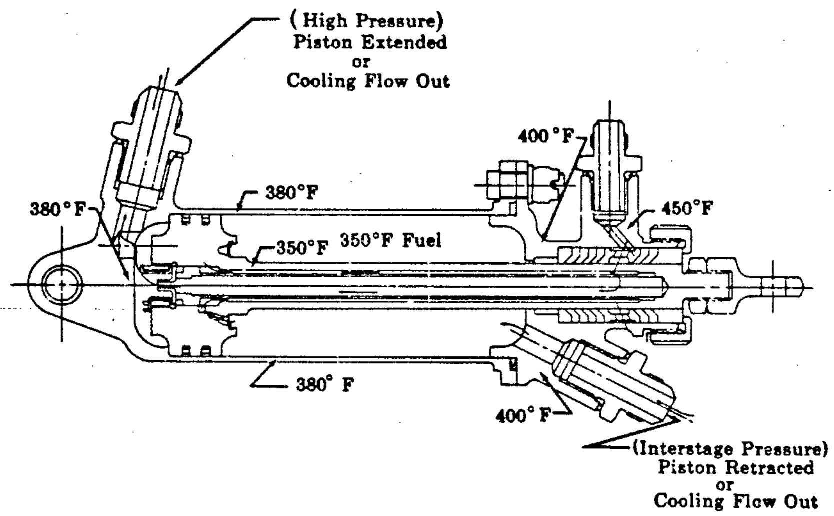

Figure 7. Clamshell Hydraulic Actuator FD 16422 BIIF

The actuators are operated by a hydraulic ΔP of 1100 psi across the piston. This high pressure is at the head of the piston only; the

actuators are returned to normal position by the main fuel pump interstage pressure of 400 psi. This interstage pressure is the only pressure acting on the actuator rod seals and is used to cool the inner piston and rod through the use of internal ports and a metering orifice. Thus, a reliable, leak-proof seal has been designed by using only low pressure on the rod seal end of the actuator.

Actuator cooling is provided when the piston rod is in either the extended or normal position. When the piston rod is extended (high pressure on head end of piston), the cooling flow passes through the metering orifice, piston, and rod and exits through the actuator-close port. When the piston rod is retracted, the flow is in the opposite direction, exiting through the actuator-open port. Three actuator fittings are used (piston extended, piston retracted, and seal drain). The piston rod is retracted at all times, except during thrust reversal, to provide actuator rod protection and prolong seal life.

d. Trailing Edge Flaps

The 16 free-floating trailing edge flaps, shown in figure 8, and the interleaving seals are fabricated from titanium alloy and nickel alloy. The heat-resistant nickel alloy is used on the inner (hot) side of both the flaps and longitudinal seals. The titanium alloy is used on the outer side and for the ribs. Insulation trapped between the inner skin and ribs protects the titanium ribs from direct contact with the hot inner skin.

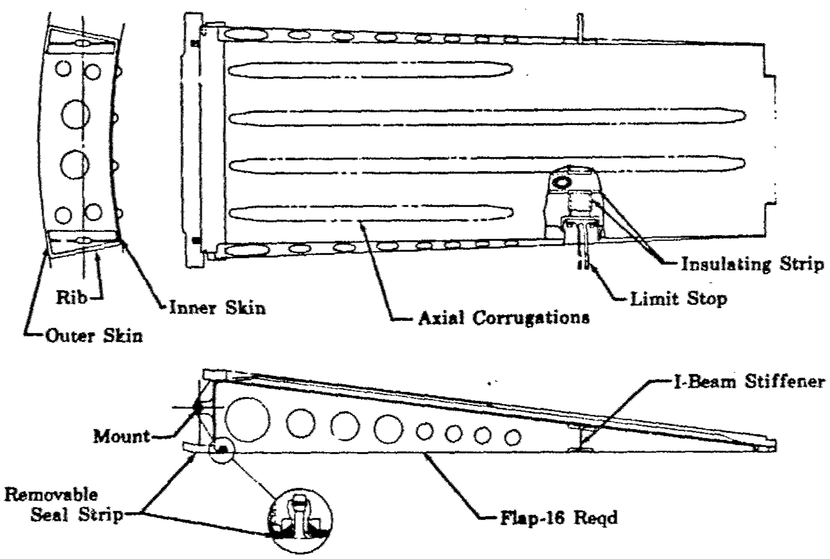

Figure 8. Trailing Edge Flap

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Differential thermal expansion between the two skins in the longitudinal direction is provided for by spacers that are riveted to the inner skin and attached through slots to the ribs. Expansion in the circumferential direction is accounted for by longitudinal corrugations between the ribs. This concept of unrestrained and corrugated inner

skins is the result of development for the J58 engine and is currently in service. Riveting the inner skin also provides easy removal for repair or replacement.

Hinges for the flaps and longitudinal seals are provided with replaceable bushings of wear-resistant alloy. The outer hinge seal is removable for inspection of the flap hinges.

The range through which the flaps and seals can move is limited by a stop mechanism, which is shown in figure 9. In this arrangement, a stop pin and bushing is retained in the seal segment and is passed through slotted brackets that are attached to each of the adjacent flap segments. The flaps and seals are free to move between the open and closed positions until the pin bottoms in the slotted brackets. The brackets, pin, and bushing are readily replaceable.

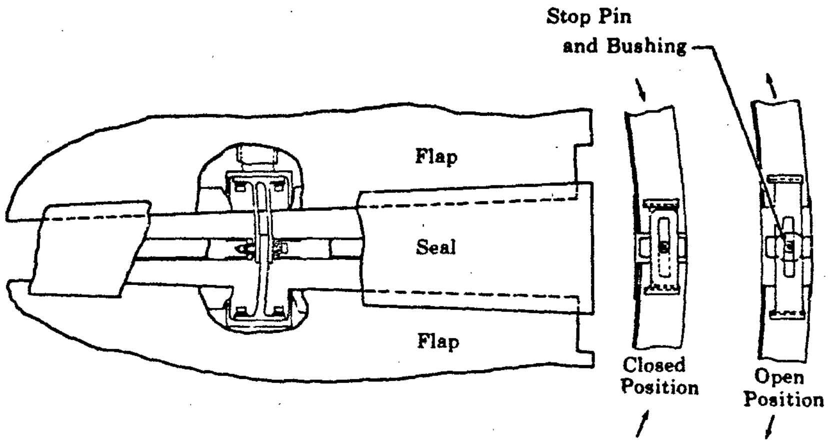

Figure 9. Flap and Seal Stop Mechanism FD 16425 BIIF

e. Clamshell-Tertiary Door Interlock Module

A mechanical interlock and synchronizing system, shown in figure 10, connects the tertiary doors with the clamshells. This system is incorporated into the reverser-suppressor to provide a fully aerodynamic-actuated system in forward flight, while retaining pilot control during reversing modes. The force required to position the clamshell doors during all forward flight conditions is supplied by aerodynamic loading on the tertiary doors. A reverser actuation system is integrated into the interlock system and overrides the aerodynamic system when actuated. This design eliminates the need for pilot action or for complex Mach number/altitude control systems to operate the reverser-suppressor during all forward flight modes. Reversing can be achieved anywhere within the reverser envelope; the reverser system is locked out at other flight conditions to prevent inadvertent reversing at high Mach numbers.

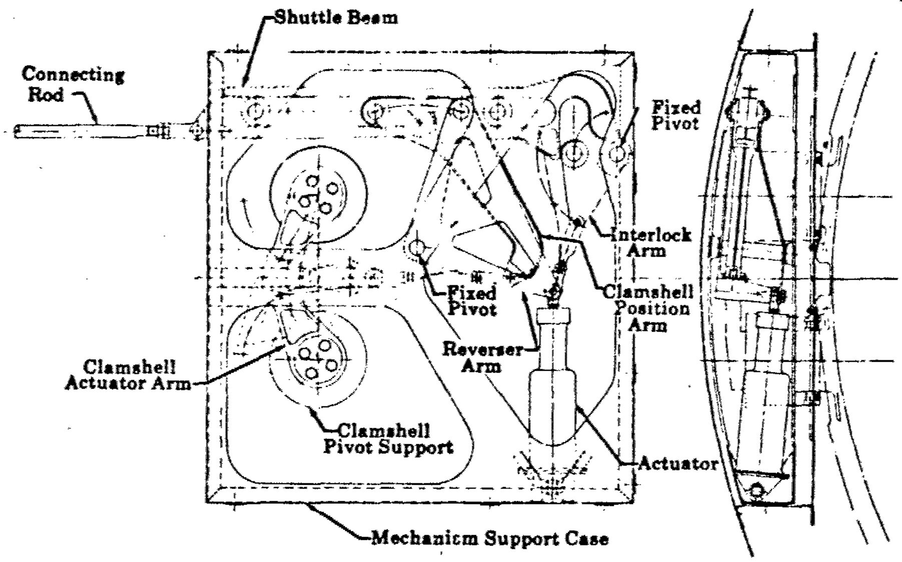

Figure 10. Clamshell-Tertiary Door Interlock System

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BIIF

The interlock module contains the mechanical and hydraulic system. The module can be removed or installed by disconnecting the one connecting rod, the hydraulic actuator plumbing lines, and the bolts that mount the module to the main structure. The modular design permits bench assembly, testing, and checkout prior to installation. In addition, all parts are easily accessible for inspection or replacement while installed on the reverser-suppressor by the removal of the outer panel cover.

As in the tertiary door interconnecting saddle system, all links have ball joints and all rollers are designed to operate without lubrication; all are based on J58 engine development and service experience. All pivot points have removable bushings for replacement at scheduled overhaul, if required.

f. Throttle-Reverser Interlock Assembly

The throttle-reverser-interlock assembly, shown in figure 11, serves to interlock the engine fuel control (throttle) and the thrust reverser doors so that:

1. In forward flight position, inadvertent motion of the clamshells to the reverse position returns the throttle to engine idle.
2. It is not possible to apply forward thrust unless the clamshells are in forward flight position.
3. It is not possible to apply reverse throttle without the clamshells being shifted to the reverse position.
4. In the reverse position, inadvertent motion of the clamshells to the forward flight condition returns the throttle to engine idle.

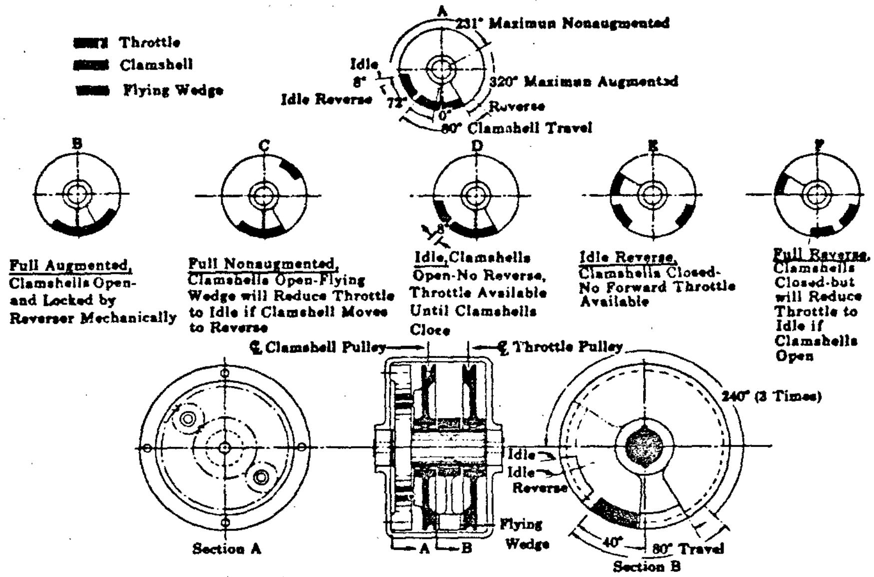


Figure 11. Throttle-Reverser Interlock Assembly FD 16436 BIIF

These requirements are met by a combination of stops and gearing arranged and connected to the items to be interlocked so that the completed unit is a compact assembly approximately 4.5 inches in diameter and 3 inches thick. The interlock assembly, which is self-contained, shielded from dirt and foreign matter, and easy to service or replace, consists of a housing with the following parts:

1. A throttle pulley that incorporates a stop and thus matches the stop motion to the throttle motion. As shown on the interlock diagram (figure 11a), zero is the full reverse position, 72 degrees is idle position, 231 degrees is the maximum nonaugmented power setting, and 320 degrees is the maximum augmented power setting.
2. A clamshell pulley that is coordinated with the motion of the clamshell by cables and links and incorporates a stop. This pulley is mounted on a shaft with the throttle pulley so that the stops on both pulleys rotate in the same plane and, at certain positions, interact with each other.
3. A flying wedge that is mounted between the throttle pulley and clamshell pulley and on the same shaft. The wedge rotates in the same plane with and interacts with the throttle and clamshell stops. The wedge is geared to the clamshell pulley by planetary gearing to amplify its motion to fit the requirements.

At the clamshell open position, the stop is 8 degrees from the throttle stop in the idle position (figure 11d). This 8-degree motion of the throttle stop allows the clamshells to close and, by means of linkage, moves the clamshell stop 80 degress counterclockwise (figure 11e). If the clamshells do not close, the stop remains in position and the throttle can no longer continue in the reverse thrust direction.

In the full-reverse condition (figure 11f), the clamshell stop is clear of the throttle stop, but any motion of the clamshell to open causes the clamshell stop to move 80 degrees clockwise, contacting the throttle stop and pushing it to the idle position.

In forward flight, the flying wedge is at the 320-degree position (figures 11b and 11c), clear of all forward-flight throttle stop positions. Motion of the clamshell stop to reverse position (80 degrees) causes the wedge to move 240 degrees counterclockwise (figure 11e), sweeping the entire forward-flight throttle stop area, and moves the throttle stop to idle. When the clamshells are closed, the flying wedge is at the throttle stop idle position, thus making it impossible to move the throttle in the forward thrust direction unless the clamshells are first opened.

4. Operational Sequence

The JTF17 reverser-suppressor operates through three basic modes, as illustrated in figure 12.

a. Subsonic Flight Operation (figure 12a)

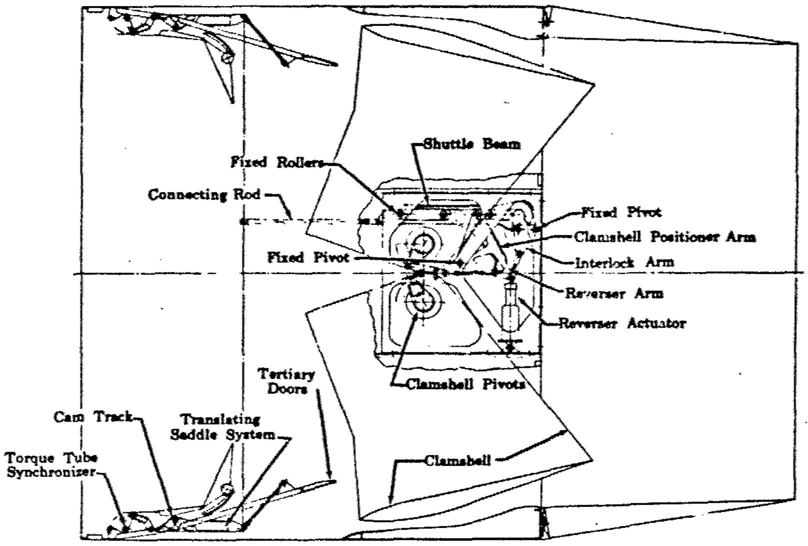


Figure 12a. Operational Sequence of the JTF17 Reverser-Suppressor FD 16442 BIIF

Pressures within the reverser-suppressor are below ambient at all subsonic conditions and at certain low supersonic conditions during descent with the engine at idle. This negative pressure forces the tertiary doors inward, permitting tertiary airflow into the reverser-suppressor to fill the exit area and prevent overexpansion of the engine exhaust stream and resultant performance losses.

As they move inward, the tertiary doors move an interconnecting saddle system and attached shuttle beam aft until the shuttle bottoms. This limits the maximum inward swing of the tertiary doors during forward flight.

A cam track on each of the saddles guides a midhinge of the tertiary air doors and aligns the doors for the proper tertiary flow lead-in angle (9 degrees). The clamshells are positioned for maximum subsonic nozzle performance by the clamshell tertiary door interlock system.

As tertiary flow requirements vary, the tertiary doors move between the maximum inward (open) position and the closed position. The doors are synchronized to move as a unit to eliminate aerodynamic cycling of the individual doors due to asymmetric environmental pressures that result from angle-of-attack variations or gust loads. No control system or pilot attention is required for any of these operations.

At this flight condition, aerodynamic pressures force the trailing edge flaps to the closed position.

b. Supersonic Flight Operation (figure 12b)

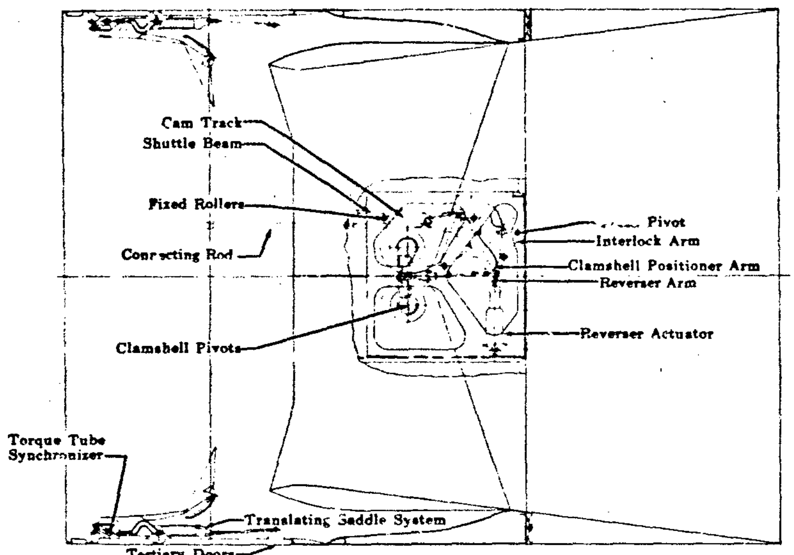


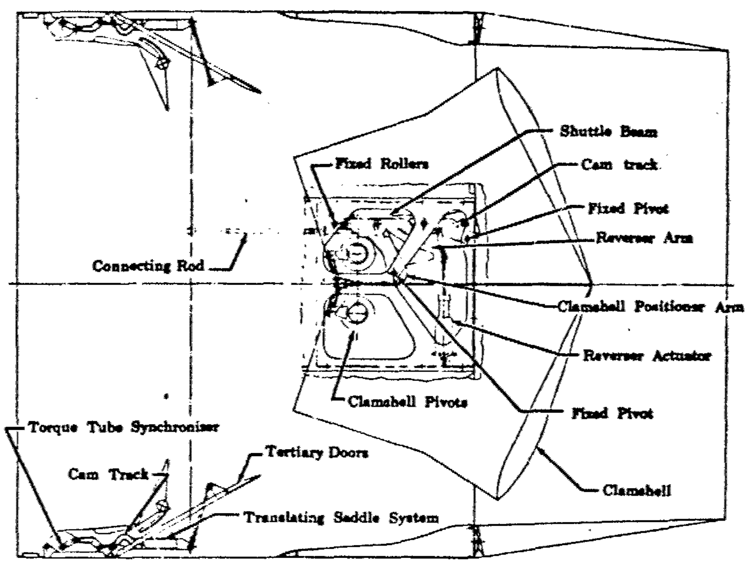
Figure 12b. Operational Sequence of the JTF17 Reverser-Suppressor FD 15442 BIIF

Above certain altitudes, Mach numbers, and power lever settings, pressures within the reverser-suppressor rise above ambient and move the tertiary air doors toward the closed position. The translating saddle system and the connected shuttle beams move forward and, during the last portion of this travel, rotate the clamshells to the supersonic cruise position. This rotation changes the inner shroud configuration to a divergent nozzle for maximum supersonic performance. Because of the toggle action of the links to the saddle system, slight internal pressures securely hold the tertiary doors shut and the clamshell in the cruise position.

The aft end of the shuttle beam moves mechanically to block the reverser actuator from extending and thus prevents inadvertent reverser operation at supersonic conditions. This entire operation is aerodynamically actuated and no pilot attention or control system is required to accomplish any portion of the operation.

At this flight condition, aerodynamic pressure forces the trailing edge flaps to the open position.

c. Reverser Operation (figure 12c)



REVERSE POSITION

Figure 12c. Operational Sequence of the JTF17 Reverser-Suppressor

FD 16442
BIIF

The reverser can be actuated at any time during subsonic flight within the reverser operating envelope. Hydraulic pressure, applied to the reverser actuators, rotates the clamshell to the reverse position by use of a toggle linkage system. The same motion moves the shuttle beam and translating saddle system aft to position and lock the tertiary doors in a reverse configuration.

Forces are balanced in the system so that in hydraulic failure, aerodynamic forces move the clamshells back to subsonic position, unlocking the tertiary air doors and permitting the doors to assume a free-floating, forward flight position. This assures that hydraulic failure cannot compromise forward flight operation.

5. Airframe Installation

The basic concepts of the reverser-suppressor are the same for the two airframes. Differences in configuration are required because of differences in nacelle contours, as shown in figure 13.

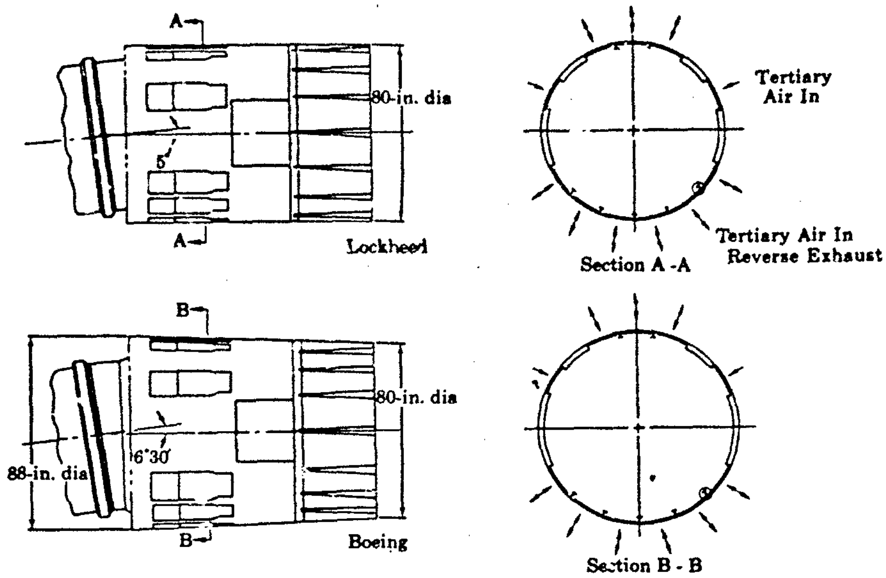


Figure 13. Comparison of Boeing and Lockheed Reverser-Suppressor FD 16569 BIIF

6. Reverse Backflow Prevention

During reverse operation, the blocked and redirected exhaust gases must be prevented from backflowing within the engine compartment. This is accomplished in the Lockheed installation by free-floating flapper doors located approximately 11 inches aft of the engine rear mount ring between the engine outer case and the nacelle, as illustrated in figure 14. There are 32 flapper doors located around the circumference of the engine, with doors omitted at the 45-degree locations to provide space for the duct exhaust nozzle actuators and hydraulic lines. The doors are statically balanced about an off-center hinge line to provide unequal area distribution. When acted upon by the exhaust stream, the unequal area provides the force necessary to close the doors. Stops are provided for each door to limit the open position to approximately 60 degrees, thus assuring that the doors will always be in an easy closing position. In the open position, the flow area through the doors is 550 in^2. This area is provided to permit the flow of secondary air to cool the reverser-suppressor during supersonic operation. Each door operates independently of the others and can easily be removed without disturbing the mounting or alignment of the other doors.

In the Boeing installation, secondary air is ducted from the inlet through six bypass ducts. In this installation, valves in the bypass duct tubes prevent backflow of exhaust gases. These valves are part of the bypass-cooling air control system.

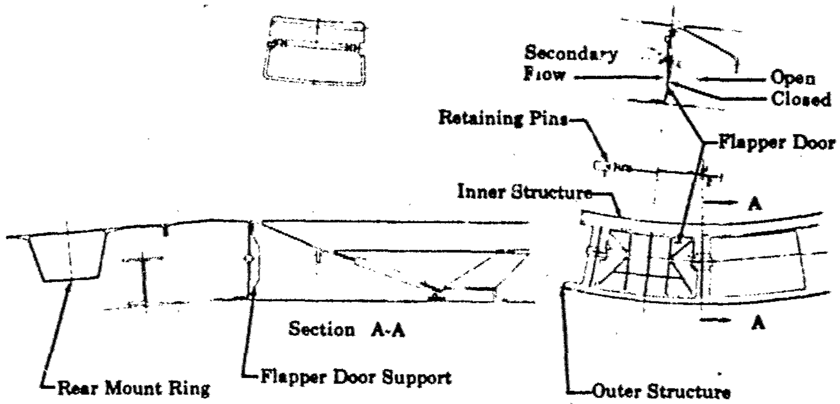


Figure 14. Free Floating Secondary Air Flapper Doors FD 16451 BIIF

7. Material Selection

Material selections have been made based on experience in comparable or similar components in other engines or aircraft. Titanium was selected for the basic structure of the reverser-suppressor based on successful use of this material in the YF-12 and SR-71 airframe-mounted blow-in door ejector for the J58 engine. This unit operates through a broader speed and altitude range than will be experienced on the JTF17.

Hastelloy X was selected for the clamshells and for the trailing edge flap inner skins based on demonstrated ease of repairability and oxidation resistance when used in hot section components of the J58 and present commercial engines. Hastelloy X has no heat-treat requirement and is not subject to aging when maintained at temperature for long periods. Inconel 625 is being investigated as an alternative material for these locations. This material retains the above desirable features and has a higher allowable stress, but has not yet been incorporated because of lack of sufficient test experience.

Inconel 718 has been used in other applications where high temperature strength is required (clamshell pivot). This material is used as basic structure for the J58 engine diffuser case and thrust mount and for the J58 rotor thrust bearing mount. This material is also used as basic structural material in the production TF30 engine.

The use of L-605 as a high temperature wear resistant material is common practice throughout Pratt & Whitney Aircraft engines and industry in general.

Materials and coatings selected for the actuators are identical to those used in J58 afterburner nozzle actuators at higher pressures and temperatures.

Table 1 summarizes the selection of materials.

Table 1. Summary of Selection of Materials

Unit	Material	Criteria
1. Basic Structure	Aluminum-coated PWA 1202 and PWA 1204 (811 Titanium)	Corrosion protection and weight
2. Tertiary Doors	Titanium honeycomb Face Sheets-PWA 1202 (811) Core-AMS 4901 (Commercially Pure)	Weight
3. Trailing Edge Flaps		
Outer Skin and Ribs	Aluminum-coated PWA 1202 (811 Titanium)	Corrosion protection and weight
Inner Skin	AMS 5536 (Hastelloy X)	High temperature oxidation resistance, weldable, non-heat-treatable.
Stops	AMS 5709 (Waspaloy)	Strength and wear
4. Clamshells	AMS 5536 (Hastelloy X)	High temperature oxidation resistance, weldable, non-heat-treatable.
Pivot Hub	PWA 1010 (Inco 718)	Strength
5. Seals	AMS 5537 (L-605)	Wear
6. Cam and Roller Tracks	AMS 5537 (L-605)	Wear
7. Actuators		
Housing	AMS 5616 (Greek Ascoloy)	Strength and wear
Piston Rod	AMS 5616 (Greek Ascoloy)	Strength and wear
Piston Rod Hardface	PWA 46 (Tungsten Carbide)	Wear
Piston Rod Guide Bushing	Titanium Carbide	Wear
8. Hinge Bushings	AMS 5616 (Greek Ascoloy)	Wear

8. Design Criteria

The reverser-suppressor structure is designed for the following criteria:

1. Maximum ΔP (internal to external)
   Across tertiary air doors and support struts = 9.85 psi
   At reverse, ΔP across tertiary air doors = 6.8 psi
   At reverse, ΔP across support struts = 10.0 psi
2. Gust loading
   Vertical = 0.85 psi
   Horizontal = 1.5 psi

3. Maneuver g-loads - see figures 15 and 16.
4. Creep limit of 0.5% maximum consistent with 50,000 hours airframe life as defined in Section I of this Volume.
5. Buckling margin of 1.3 minimum for either plastic or elastic buckling. For hardware such as rods where eccentricities are well defined, the minimum margin is 1.15.
6. Maximum combined stresses are limited to the lesser of 100% of yield stress or 2/3 of ultimate stress.

The stress factors listed above are selected based on their successful use in structural components of the J58 engine and are consistent with the extended life and reliability requirements of the JTF17 engine.

The operating pressures, gust, and maneuver loads are consistent with engine performance and airframe installation requirements.

9. Actuation Response

Calculated clamshell response to the hydraulic reverse signal is shown in figure 17. Actuation is most rapid during emergency conditions such as aborted takeoffs when engine rpm is high and the hydraulic pump output is maximum. Clamshell rotational speed, shown in figure 18, indicates the inherent snubbing effect of the linkage between the actuator and clamshell system, slowing the clamshell speed as full reverse position is reached. Maximum calculated actuator loads, shown in figure 19, are based on calculated clamshell torque loads and correlated data from wind tunnel scale model reverser testing.

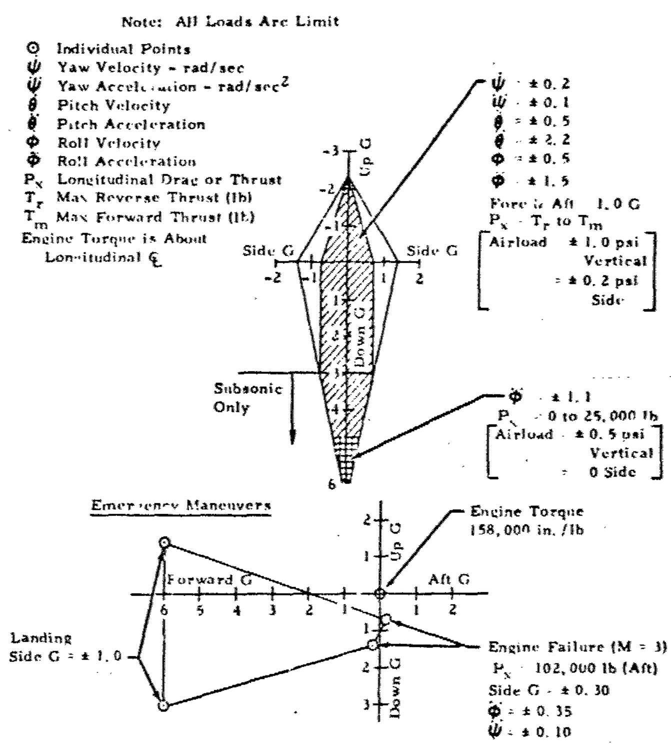


Figure 15. JTF17 Maneuver Load Diagram - Lockheed Installation FD 16285 BIIF

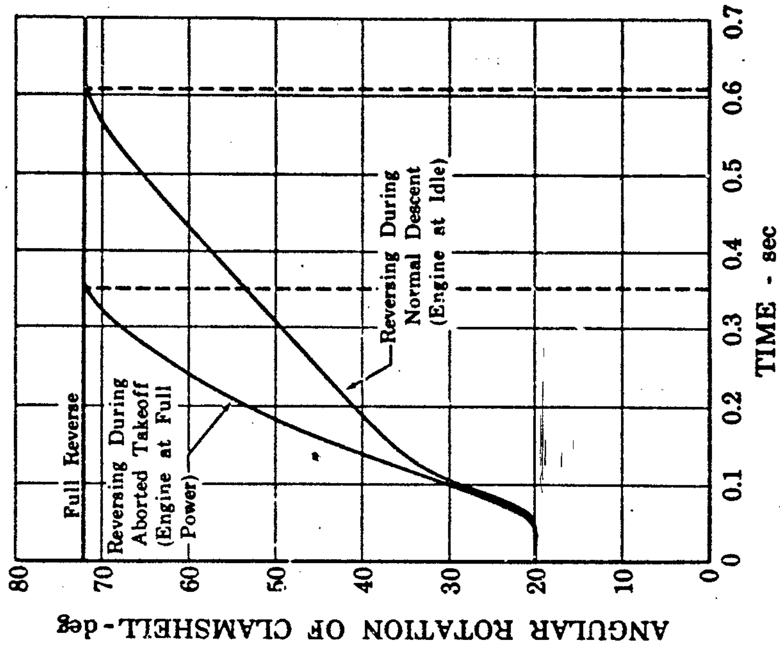


Figure 17. Clamshell Rotation vs Time FD 16405
BIIF

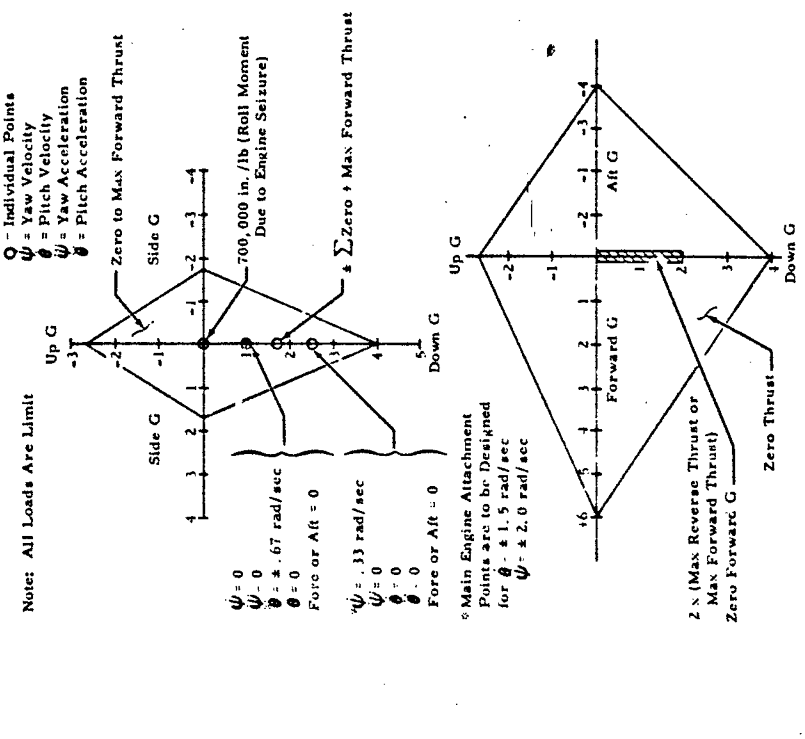


Figure 16. JTF17 Maneuver Load Diagram - FD 16287
Boeing Installation BIIF

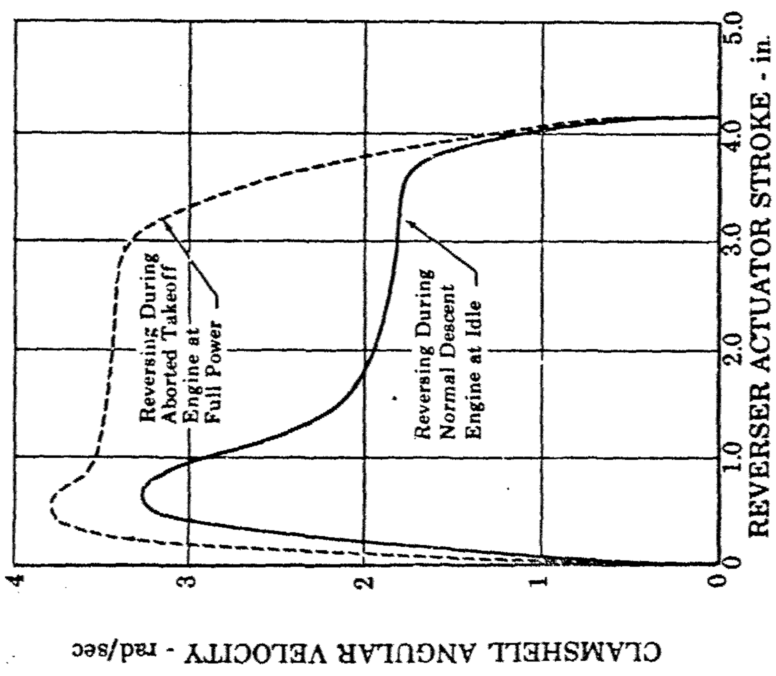


Figure 18. Clamshell Angular Velocity vs Actuator Stroke FD 16404 BIIF

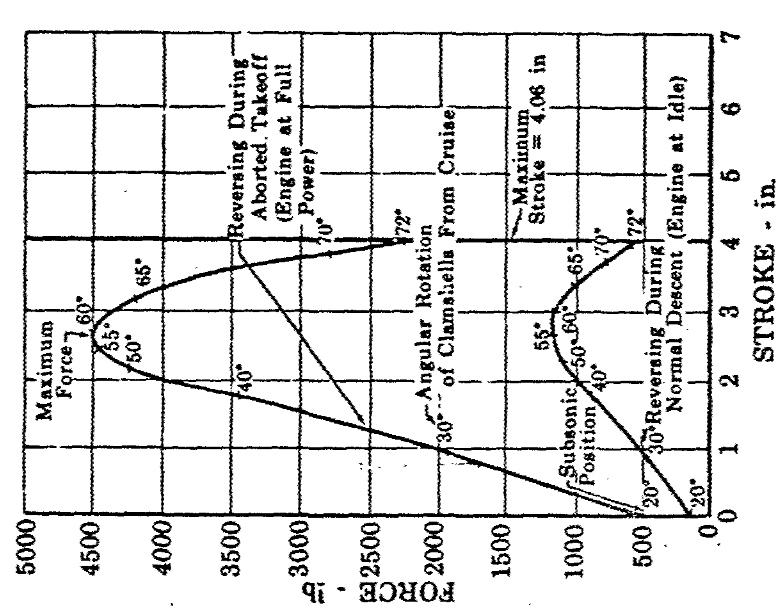


Figure 19. Clamshell Actuator Force vs Stroke FD 16043 BIIF

10. Operating Temperatures

Predicted average metal temperatures in the reverser-suppressor in the three flight modes are shown in figure 20.

Predicted reverser actuator metal, fluid, and environmental temperatures are shown in figure 7. Cooling flow circulation routes are also indicated. These predictions are made using analytical methods that were developed and used to predict J58 engine metal and flow temperatures and that were later verified with correlating data.

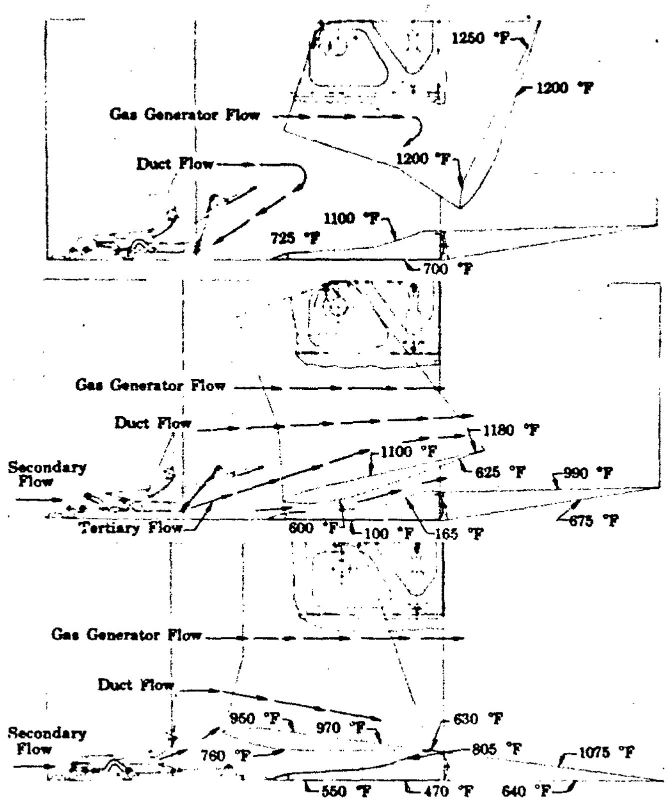


Figure 20. Thrust Reverser System Metal Temperatures FD 16498 BIIF

11. Operating Loads

The maximum calculated aerodynamic pressures on the clamshells during reverse thrust are shown in figure 21. These maximum pressures result in the clamshell pivot loads shown in figure 6. Clamshell interlock loads during reverse are shown in figure 22.

The thrust reverser is designed to operate with a maximum ΔP of 12.7 psi across the clamshells.

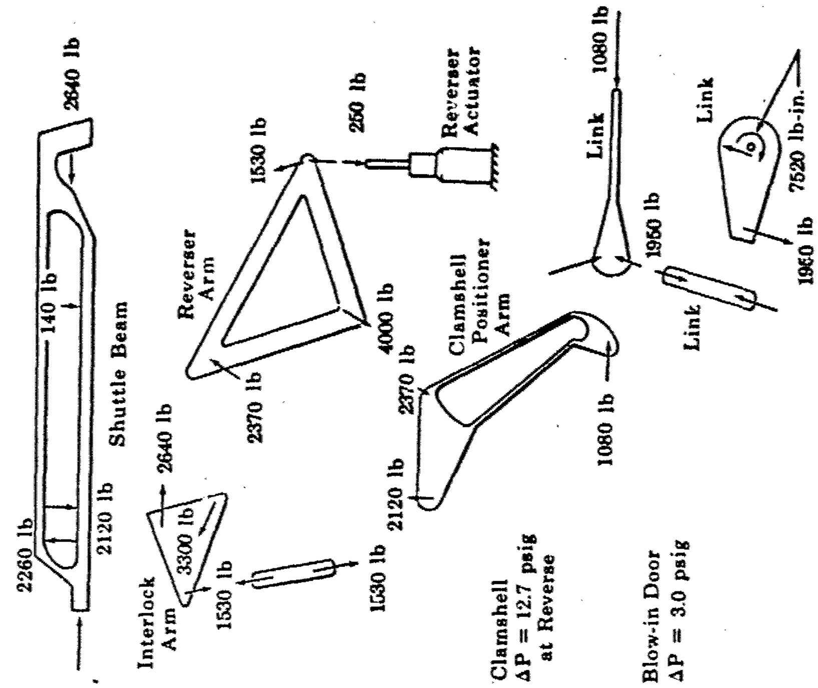


Figure 22. Clamshell Interlock Loads During FD 16432 Reverse BIIF

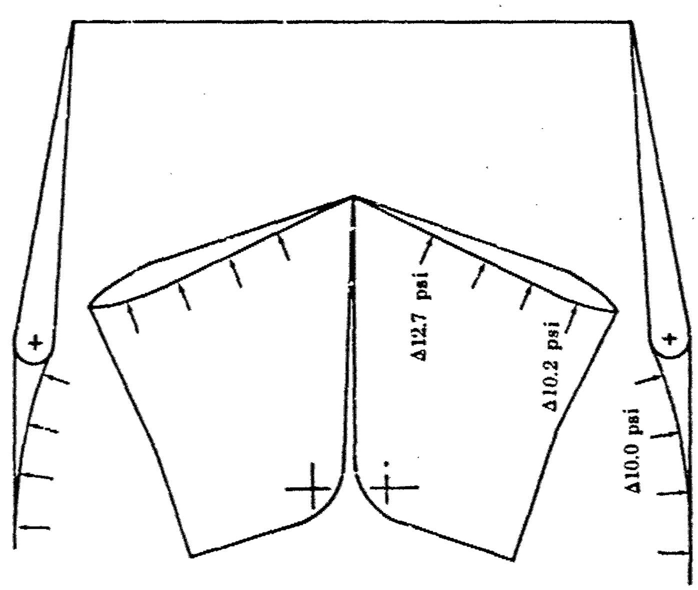


Figure 21. Aerodynamic ΔP on Clamshells at FD 16427 Reverse Condition BIIF

12. Phase II-C

During Phase II-C, design of the translating shroud reverser-suppressor proposed in Phase II-B was completed along with the alternative design study of the clamshell reverser-suppressor. The clamshell approach resulted in less weight, smaller size, and higher potential life. Effort was terminated on the development of the translating shroud concept and experimental parts procured to the clamshell design. Testing of this design will be initiated during September 1966.

13. Phase III Objectives

During Phase III, the objectives will be to develop the reverser-suppressor to meet mechanical, performance, and sound suppression objectives. The final installation configuration will be evaluated for mechanics and performance. These test plans are discussed in Report E.

During Phase III, it is planned to continue wind tunnel testing to optimize aerodynamic contours. Report E will cover detail plans.

Comprehensive studies of sound suppression, as discussed in Report C, will be conducted and results will be reflected in design modifications, as required.

Other titanium alloys and protective coatings will be evaluated to provide improved corrosion resistance and/or protection, as necessary.

The feasibility of using titanium actuators will be studied. This will require the development and use of either a wear-resistant liner or development of wear-resistant coatings for the cylinder bore. Both approaches will be evaluated.

14. Growth

Growth potential is inherent in the design of the reverser-suppressor. With some weight penalty to withstand higher pressures and with selective substitution of some materials to withstand higher temperatures, the unit has the potential of providing greater reverse thrust. A potential also exists, with some weight and control system penalty, for providing for an intermediate null thrust operating mode of the reverser if this should become an airframe requirement.

15. Product Assurance Considerations

The reverser-suppressor is designed to provide ease of access and maintenance as follows:

1. The complete reverser-suppressor assembly can be installed or removed without disassembly other than disconnecting the hydraulic supply lines and throttle interlock mechanism.
2. Individual components subject to wear or possible damage can be removed and replaced without major engine disassembly.
3. External access panels are provided for the clamshell-tertiary door interlocks.

4. Hydraulic reverser actuator system and clamshell-tertiary door interlock system is modular and is mounted in a separable framework that can be installed, removed, or bench-tested as a package.
5. Lifting provisions are incorporated into the basic reverser-suppressor to provide ease of handling.
6. Like components are interchangeable, thus eliminating the need for fixtures, jigs, or other devices to install or replace components.

The reverser-suppressor is designed for reliability and fail-safe operation under all conditions. The features that accomplish this include the following items:

1. Throttle-reverser interlock system
2. Elimination of high pressure on actuator seals
3. Pressurized actuator required only for thrust reversing
4. Aerodynamic balancing of clamshells to reduce actuator forces and assure fail-safe positioning in event of hydraulic failure
5. Integral, aerodynamically-actuated system to provide all forward flight configurations
6. Automatic mechanical lockout of reverser system at high Mach numbers.

Providing an extended service life for the reverser-suppressor has been the basic guideline for the total design. To achieve long service life, the following design philosophy was applied:

1. All main structural members are designed for compression or shear buckling stresses. These are well below allowable yield stresses, thereby eliminating or reducing the creep life and fatigue considerations for metals at elevated temperatures.
2. The inside or hot surface of the reverser-suppressor is made of Hastelloy X for maximum life at elevated temperature and for ease of weld-repair of damaged parts.
3. The three basic modes of the reverser-suppressor (subsonic, supersonic, and reverse) are basically aerodynamically and mechanically stable and do not require continuous or oscillatory motion to achieve maximum performance. Each mode change is smooth and positive, and only four motions occur during a typical flight. This results in a reduction of wear on the moving parts. In addition, all mode changes, except aborted takeoff, occur during periods of low load conditions, a feature that increases the dynamic life of each of the moving parts.
4. All parts subject to wear are separable from the reverser-suppressor structure, allowing use of special high-strength, wear-resistant materials without compromising the main structural integrity.

16. Background Exhaust System Studies

During the early study phases of the supersonic transport engine program, Pratt & Whitney Aircraft evaluated several different exhaust system concepts. The more promising types evaluated included the plug nozzle, the long actuated flap variable nozzle, and the blow-in door ejector.

a. Plug Nozzle

The plug nozzle can provide aerodynamically-adjusting performance characteristics. In this type nozzle the throat is annular and supersonic expansion of the exhaust gases takes place along a central plug. There is an aerodynamic adjustment to ambient pressure accompanied by an appropriate variation in flow area.

The nozzle flow field is, however, strongly influenced and sensitive to local pressure variations surrounding the nozzle. The presence of an external air stream flowing over a base or boattail will produce lower than ambient pressures and thus induce overexpansion of the exhaust stream and attendant internal drag losses.

The plug nozzle also has serious mechanical and cooling problems when designed for an augmented engine with attendant requirements for a variable throat area. This is because (1) large throat area variation requirements result in long actuation strokes, (2) there are high actuation loads, and (3) large quantities of cooling air are required. Thrust reversing a coannular plug flow is limited to an umbrella type which is incompatible with airframe requirements.

b. Long Flap Nozzle

The long-flap variable nozzle system, shown in figure 23, is composed of a series of long, overlapping flaps that are mechanically actuated. A control system is necessary to sense and position the flaps according to power setting and flight Mach number.

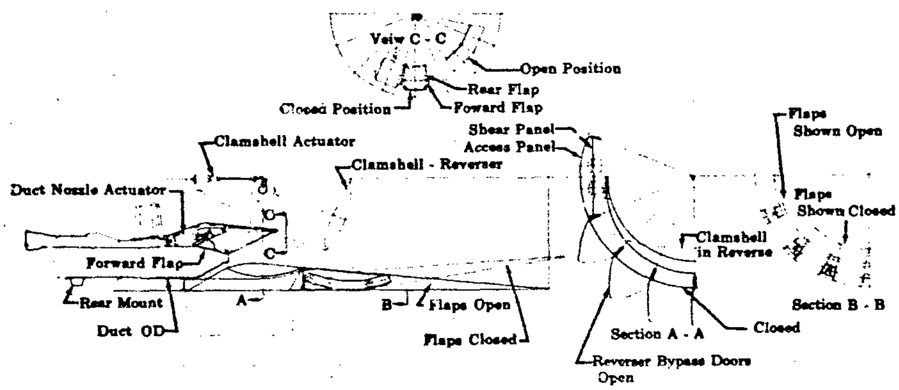


Figure 23. Long Flap Variable Nozzle System FD 16444 BIIF

The position of these flaps determines both the nozzle exit area and the external boattail geometry. The flaps thereby control both the internal performance level and external boattail drag losses. Both flap length and amount of travel are large, as dictated by these performance requirements. The need for secondary airflow is high because of the large scrubbing area to be cooled and because of leakage through long-flap seals. Actuation and attachment loads of the flaps are high because of flap length and area exposed to differential pressures. To fit this type system with a suitable reverser, it is necessary to provide a variable

inner plug to control duct nozzle area. The weight, mechanical complexity, and control system required for the long-flap nozzle make it unattractive for the supersonic transport in comparison to the blow-in door ejector approach.

c. Blow-in Door Ejector

The blow-in door ejector is an infinitely variable, self-actuated nozzle that aerodynamically adjusts to the correct expansion ratio as engine pressure ratio and flight Mach number vary.

The major portion of the ejector is fixed and the movable portions are short and self-adjusting, making the nozzle light, reliable, and efficient. Effective thrust reverser mechanisms can be incorporated with minimum complexity, utilizing the existing blow-in tertiary doors as reverser flow doors. Wind tunnel testing on the blow-in door concept was initiated in 1956. Since that time over 12,000 hours of wind tunnel testing have been accomplished. In addition, J58 engine flight experience with a blow-in door ejector has been, and is being, obtained through speed and altitude ranges which exceed those of the supersonic transport.

d. Performance Comparison

A comparison of the performance of the long-flap nozzle and the plug nozzle relative to the blow-in door ejector is shown in figure 24.

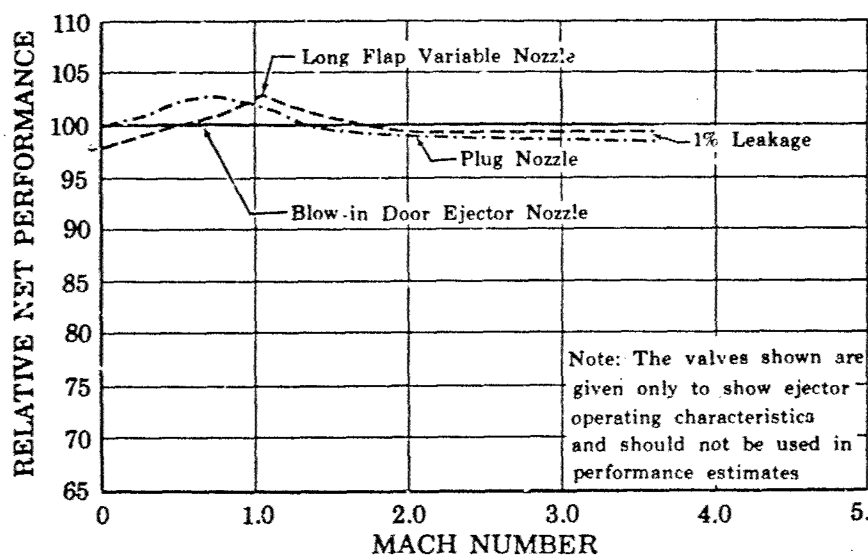


Figure 24. Nozzle Performance Comparison FD 16985 BIIF

The long-flap nozzle is competitive with the blow-in door ejector during supersonic cruise conditions assuming no flap leakage occurs. (Leakage with this nozzle is considered a particular problem because of the long flaps.) During transonic acceleration, operation of the long flaps can reduce the external pressure drags and give better performance than the blow-in door ejector; however, during subsonic acceleration and

subsonic cruise operation, the long flaps must be closed to a small exit area to avoid severe overexpansion losses. This will increase external drag losses. This combined effect will result in lower subsonic performance with the long flap nozzle than with the blow-in door ejector.

The plug nozzle, in the simplest form, possesses mechanical and weight advantages over the long flap and blow-in door nozzle, but has, however, the lowest supersonic cruise performance. To generate competitive supersonic and transonic performance, a more complex design is necessary. At supersonic conditions an external expansion surface must be provided. This surface in turn must be removed or the resultant exhaust area filled at subsonic and transonic conditions to prevent overexpansion and high base drag losses. A relatively simple plug nozzle design will be inherently poorer in performance than a comparable blow-in door ejector nozzle. The coannular blow-in door nozzle design selected for the supersonic transport actually has combined the advantages of both the plug nozzle and the blow-in door ejector to achieve maximum performance with minimum complexity and weight. The blow-in door ejector, by virtue of the tertiary air system and rotating clamshell segment, has the potential for maximum exhaust noise attenuation.

With this background Pratt & Whitney Aircraft has selected the blow-in door ejector for the JTF17 engine. For a full discussion and comparison of types of nozzles and respective performance, refer to the Phase I Proposal, Volume E-XI, Ejector-Reverser for the Supersonic Transport Engine. For a complete discussion of the performance of the blow-in door system for the JTF17 engine, refer to Report A of this Volume.

G. BEARINGS AND SEALS

1. Main Bearings

a. Description

The JTF17 engine has been designed with a simple twin rotor system with two bearings supporting each shaft. The bearings are of single row, balanced cage construction, designed to prevent skidding. Thrust bearings are mounted on a nontorque portion of the shafts. Suitability for SST application has been demonstrated by engine and rig testing.

b. Design Objectives and Requirements

The design objectives of the main bearings is to provide radial and axial support for the rotors for all engine operating conditions consistent with the following requirements:

1. 10,000 hours of operation
2. A maximum load equivalent to 10% blade loss of any stage
3. Accessibility and removal capability without major engine disassembly
4. Designs within the proven load limits of previously tested hardware.

c. Design Approach

(1) Detailed Description

The four bearing-twin rotor arrangement is made possi[illegible] because of advances in combustor technology allowing the use of sho[illegible] rners which in turn permit shortening the engine enough so that only [illegible] bearings on each shaft are required. Another important feature is the use of single row bearings instead of double row. The following table summarizes bearing use, location and function:

Bearing Number	Type	Location	Function
1	Single Row Ball	Forward, Low Rotor	Thrust and Radial
2	Single Row Ball	Forward, High Rotor	Thrust and Radial
3	Single Row Roller	Rear, High Rotor	Radial
4	Single Row Roller	Rear, Low Rotor	Radial

The bearing compartment designs are shown in figures 1, 2, and 3. The two thrust bearings are on the forward end of each rotor, to allow the more heavily loaded thrust bearings to operate in the cooler front section of the engine. The forward ball bearings furnish axial and radial restraint and the rear roller bearings furnish radial support for each rotor.

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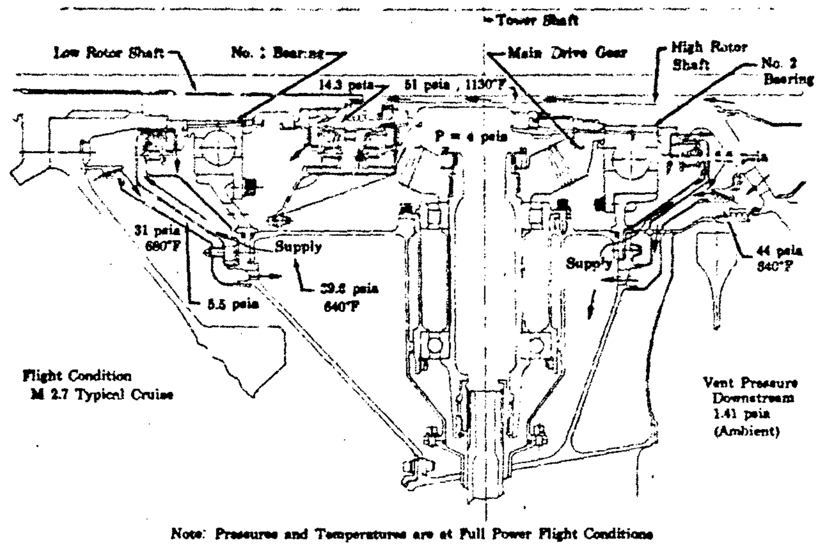

Figure 1. No. 1 and 2 Bearing Compartment Seal Pressurization

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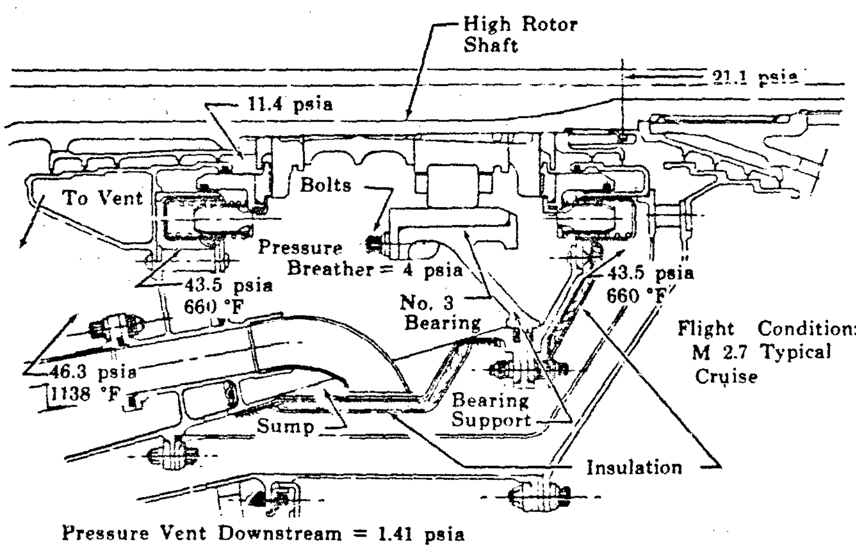

Figure 2. No. 3 Bearing Compartment

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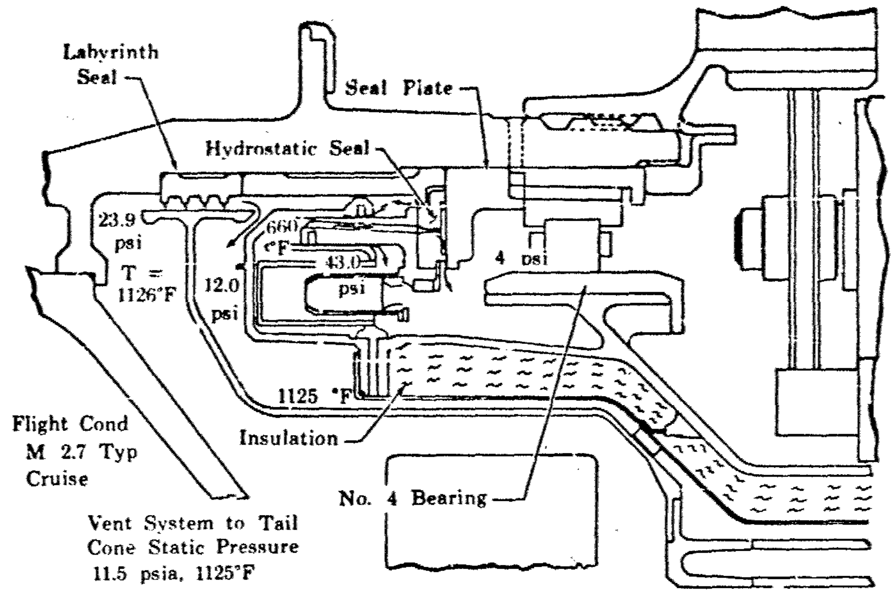

Figure 3. No. 4 Bearing Compartment FD 16414 BIIG

(a) Bearing Life

Limit life will not be applied to the main bearings. Service life will be determined based on inspection at scheduled overhaul intervals. Based on previous commercial engine experience, and the design criteria used, the JTF17 bearing service life will be in excess of 50,000 hours. The design and materials selected for the bearings, to meet this requirement (table 2), have been developed in bearing rig tests and in engine tests in the JT8D, JT3D, J58 and JTF30 engines. These bearing rig and engine tests have substantiated the fatigue life of the bearing material. The test results have shown the bearing material to have five to eight times the life predicted by the standard "Anti-Friction Bearing Manufacturers Association" (AFBMA) calculation. This data has been incorporated into the life prediction methods used by Pratt & Whitney Aircraft using the five-fold increase as a minimum.

In addition to the minimum life requirement and loading to prevent skidding the bearings are designed to be capable of taking an applied load equivalent to 10% blade loss of any stage. They will also take the ultimate maneuver loads imposed on the engine.

(b) Bearing Skid Prevention

Skidding in the ball and roller bearings must be prevented because it will cause surface damage to the races and the rolling elements. Even if skidding occurs only briefly, the surface damage that may result can significantly lower the life of the bearing. This method of preventing skidding of the SST bearings is the same as used in the J58, JT8D, and JT3D.

Skidding is prevented in roller bearings by proper preloading. This preloading is obtained by grinding the outer races to an elliptical shape so that when they are assembled into their supports, the races deform and remove the radial clearance and preload the bearings. Support spring rates are controlled in the design to control the loads imposed on the bearing. Spring rates calculated during design are substantiated by tests on the actual bearing supports.

With ball bearings, skidding is most likely to occur during a thrust reversal. The occurence of skidding can be predicted analytically by relating the friction forces and the kinetics of each element. Bearing rig and engine tests have verified this analytical procedure. The axial force acting on each rotor is maintained by the engine thrust balancing system so that they load the thrust bearings sufficiently to prevent skidding and so that the axial force is always in the same direction. Thrust balance is discussed in detail in Section H.

(c) Bearing Cage Design

All bearing cages for the JTF17 are of the fully machined type and are inner land riding. This design has proven superior in oil starvation tests in both engine and bearing rig tests. Centrifugal dirt entrapment also tends to cause higher wear in outer land riding designs. Cages are balanced separately prior to assembly into the bearing because this technique has been shown to provide long wear life in other engines.

(d) Bearing Retention

Each bearing is flanged and bolted to its housing for positive retention to prevent spinning of the outer race with respect to its bearings. In past designs and in initial JTF17 engine testing, spanner nuts have been used to clamp the bearing races. While no problems occurred during initial testing, a review of airline maintainence records showed race spinning can be a problem in engines from which long life is required.

(e) Detailed Bearing Information

(1) No. 1 Ball Thrust Bearing

The single row ball thrust bearing is a departure from commercial engine experience where duplex thrust bearings are usually used. It is made possible by improved bearing materials and advances by Pratt & Whitney Aircraft in the methods of analytically predicting internal thrust balance. (See Section H for a detailed discussion of thrust balance.)

The single row ball thrust bearing has the following advantages over the double row type previously used.

1. Does not suffer from unequal load sharing due to manufacturing tolerances or thermal growth.

2. It has fewer pieces in the assembly with consequent higher reliability.

3. Lighter in weight.

4. Less costly.

The No. 1 ball thrust bearing is mounted aft of the fan stages on the fan hub. The spline coupling the hub to the low rotor is forward of the bearing, see figure 1. This design provides the following safety features:

1. In the event of complete lubrication system failure or any other cause which might result in bearing seizure, the rotor shaft will remain intact between turbine and fan to prevent overspeeding.

2. The bearing can be easily inspected and maintained because it is accessable by removing the fan only.

The No. 1 bearing is cooled and lubricated by the engine oil system described in Section IV, Report B. The oil is fed by pressure jets into an axial scoop that feeds the bearing (figure 4). The bearing is cooled by flowing oil through the races. A portion of this oil passes through slots in the interface of the two race halves to the bearing. These slots are tapered to take advantage of centrifugal force to flow the lubricant. This type of cooling and lubrication has been proven in the J58, TF30, and JT8D engines.

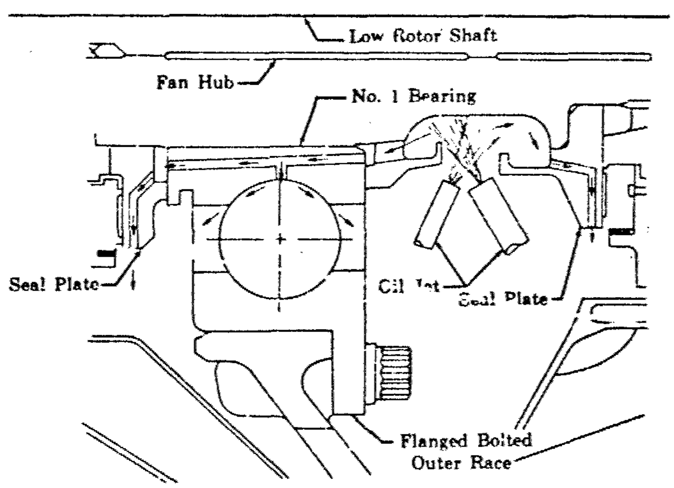

Figure 4. Typical Bearing Cooling Flow FD 16412
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(2) No. 2 Ball Thrust Bearing

The No. 2 ball thrust bearing is also a single row type. It is mounted on the front hub of the high compressor. This mounting places the bearing on a nontorque carrying member. The lubrication scheme outlined for the No. 1 bearing is also used for the No. 2 bearing. Since both the No. 1 and No. 2 bearings are in the same compartment they have similar maintainability features.

(3) No. 3 Roller Bearing

The No. 3 roller bearing is located in front of the high rotor turbine. This bearing is cooled by oil flowing under the race and lubricated by compartment mist. This cooling and lubrication method has again been proven in commercial engines as well as the J58. The No. 3 bearing and shaft are of thermally compatible materials that prevent rotation of the inner race with respect to the shaft. Assembly, disassembly, and inspection of the No. 3 bearing can be accomplished by removal of the high rotor turbine. This is a balanced assembly and can be removed and replaced without rebalancing.

(4) No. 4 Roller Bearing

The No. 4 roller bearing is located aft of the low rotor turbine; therefore, it is not on a torque-carrying shaft. The lubrication and cooling method is the same as described for the No. 3 bearing. The No. 4 bearing is accessible from the rear of the engine for maintenance and inspection without engine disassembly.

The bearing design for the JTF17 relies heavily on Pratt & Whitney Aircraft knowledge gained from commercial jet engine experience. The testing during Phase II verified the design intent of each bearing. This testing is discussed in Paragraph H of this report.

The Phase III Program will include endurance testing, heat rejection and verification of skid regions. These plans are discussed in Report E.

(2) Product Assurance Considerations

The following is a summary of the reliability and maintainability features of the JTF17 bearing design:

1. Reliability

   a. Elimination of the intershaft bearing and seals.

   b. Two bearing-supported rotors to improve engine vibration characteristics.

   c. Thrust bearings mounted on nontorque portion of shafts and at the cool section of the engine.

   d. Single row thrust bearings not subjected to unequal load sharing.

   e. Flanged outer races to prevent spinning in the bearing housings

   f. Inner race spinning prevented by mounting all bearings on high nickel alloy shaft materials for thermal compatability

   g. Symmetrical under race cooling to prevent thermal distortion

   h. Balanced cages to prevent cage wear.

2. Maintainability

a. Reduction in number of compartments.

b. Arrangement for ease of inspection and disassembly. All bearings can be disassembled without complete engine disassembly.

c. Elimination of intershaft support allows complete removable of low rotor without disturbing high rotor.

d. Flanged races eliminate large housing ner nuts.

e. Puller grooves on all bearings to faci. .ate removal.

(3) Materials Summary

Materials used, based on experience with the JT8D, JT3D, J58, and the JTF30 engines, are as follows:

Bearing Number	Race and Rolling Element Material	Cage Material
1	PWA 725 (M-50)	AMS 6415
2	PWA 725	AMS 6415
3	PWA 725/PWA 742 (Bower 315)	AMS 6415
4	PWA 725/PWA 742	AMS 6415

2. Main Compartment Seals

a. Description

The bearing compartment seals for the JTF17 engine are integrated into an advanced seal system that isolates and continually bathes the bearing compartments in cool fan discharge air. This feature, coupled with a design which eliminates rubbing seal face contact provides excellent durability along with bearing compartment temperatures equal to current commercial jet engines.

Inherent safety features gained through the use of an overboard vent prevents oil from entering the main gas generator stream thereby eliminating the danger of oil fires or cabin air bleed contamination. These advantages result in a design that exceeds the capabilities of the compartment seal systems used in current commercial engines.

b. Objectives and Requirements

The bearing compartment seal system must perform successfully throughout a wide range of environmental temperatures and pressures encountered in the flight envelope of the JTF17 engine. To ensure the safety, reliability and durability of the seal system with these stringent operating conditions, the following design objectives and requirements were established:

1. A seal system which uses the coldest available supply air source giving the required primary seal pressure differential, which ensures minimum compartment air temperatures.

2. A seal system which uses minimum supply air flow into the compartments to ensure minimum heat rejection to the lubricating oil.
3. A seal system which prevents oil vapors from entering the gas generator air stream to eliminate cabin bleed air contamination and oil auto-ignition.
4. A minimum service life of 10,000 hours.
5. Compatibility with PWA 521-B (Type II) lubricant.
6. A system which provides protection against oil contamination.

c. Design Approach

(1) Detailed Description

The seal system used in the three bearing compartments of the JTF17 engine is shown schematically in figure 5. It consists of the primary seal, the secondary seal, the overboard vent and associated air supply and vent lines.

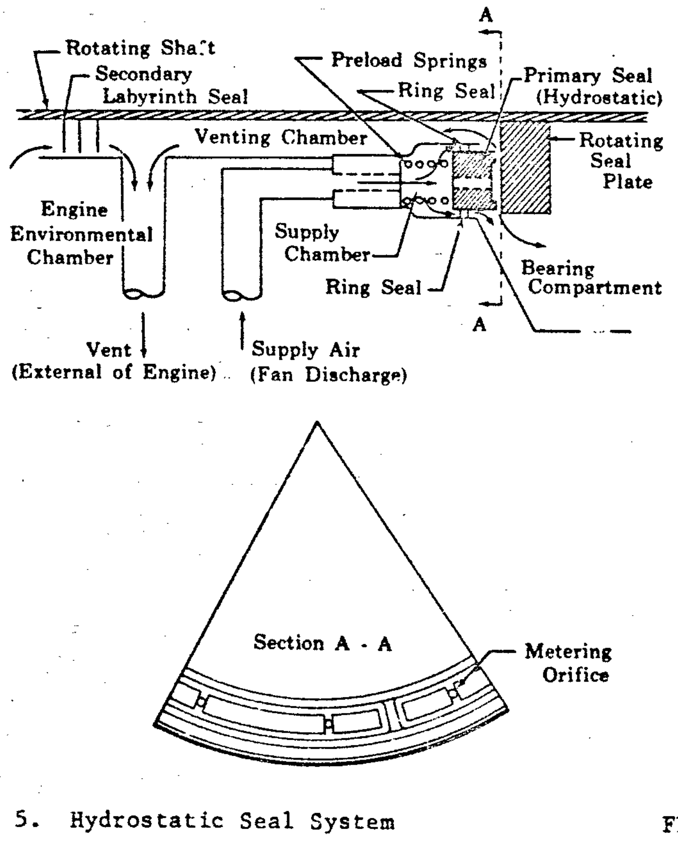

Figure 5. Hydrostatic Seal System

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In operation, cool fan discharge air is routed to the primary seal supply chamber. The airflow then passes through metering orifices at the periphery of the primary seal, which forces it to lift off from the rotating seal plate. This concept, with the seal floating on a cushion of air is called a Hydrostatic Seal. The lift off clearance is maintained by a force balance determined as shown in figure 6. Flow escaping from the seal face is routed in two paths; one path to the bearing compartment for breather system pressurization and the other path to the vent chamber where it is directed overboard through the external vent.

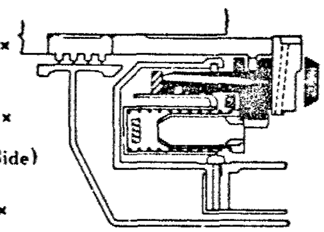
Closing Forces | **Opening Forces**

F_{HI} = Seal Supply Pressure × Area of Seal Back

F_{BRB} = Breather Pressure × Area of Carriage (Breather Side)

F_{CHB} = Chamber Pressure × Area of Carriage Back (Chamber Side)

F_{SC} = Spring Force + Spring Constant × Clearance



F_{SL} = Slot Pressure × Slot Area

F_{SLBR} = $\frac{\text{(Slot Pressure + Breather Pressure)}}{2}$ × Seal Area (Breather Side)

F_{SLCH} = $\frac{\text{(Slot Pressure + Chamber Pressure)}}{2}$ × Seal Area (Chamber Side)

$$F_{CT} = F_{HI} + F_{BRB} + F_{CHB} + F_{SC} \qquad F_{OT} = F_{SL} + F_{SLBR} + F_{SLCH}$$

$$\underline{F_{NET} = F_{OT} - F_{CT} \pm F_{RSF}}$$

F_{RSF} = Friction Factor [Ring Seal Preload + (ΔP Across Ring Seal × Seal Area)]

Figure 6. Typical Hydrostatic Seal - Force Balance FD 16384 BIIG

The secondary labyrinth seal controls air flow from the adjacent environmental chamber, into the vent chamber. This flow is also routed overboard through the external vent. The vented system prevents high temperature environmental chamber air from entering the bearing compartment and any oil vapors which might escape from the compartment are shunted overboard.

This pressurized and vented hydrostatic seal system was selected following the design, comparative analysis, and testing of three different seal systems. These three systems shown schematically in figure 7 are:

1. Radial knife edge labyrinth seals
2. Face and ring type carbon rubbing seals with backup labyrinth seals
3. Hydrostatic seals with backup labyrinth seals.

The evaluation of each system was based on its ability to achieve the design objectives and requirements in addition to the following factors relating to overall engine design:

1. Compatability with lubrication and breather system in regard to breather line size, flow rates and pressure.

2. Performance of engine de-oiler with each system
3. Size and flow rates of main oil pumps and scavenge pumps with each system.
4. Influence of seal system pressurization flow rates on engine performance.
5. Effect of seal system on engine requirement of no heat return to airframe.
6. Weight
7. Cost.

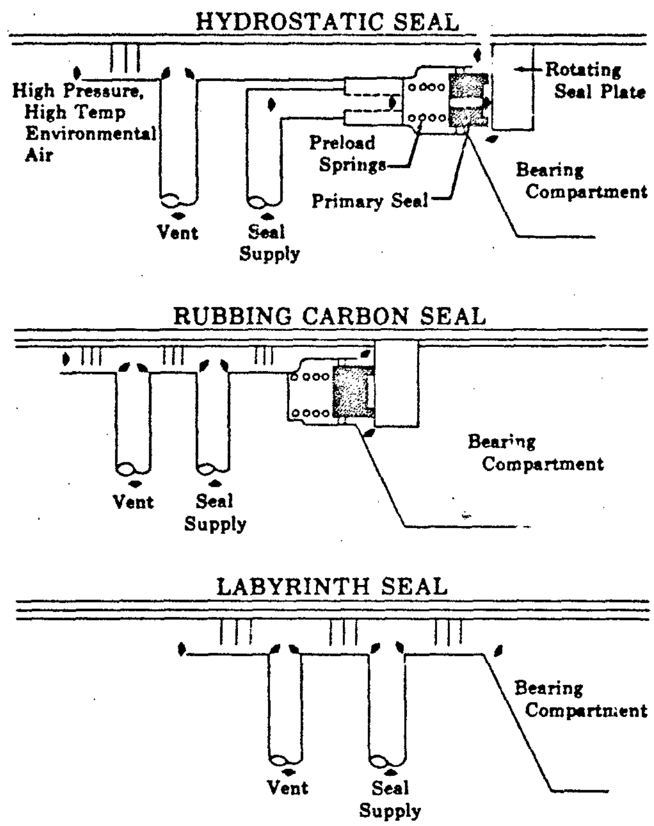


Figure 7. Comparison of Seal Arrangements FD 16398 BIIG

The system comprised entirely of radial knife edge labyrinth seals (figure 8) will provide the durability required to achieve 10,000 hours life because there are no rubbing or contacting surfaces in this design. Design and testing of this type seal determined that radial clearances necessitated by manufacturing tolerances and thermal expansion resulted in a relatively high pressurization flow rate into the bearing compartments and breather system. This in turn results in excessive heat rejection to the oil.

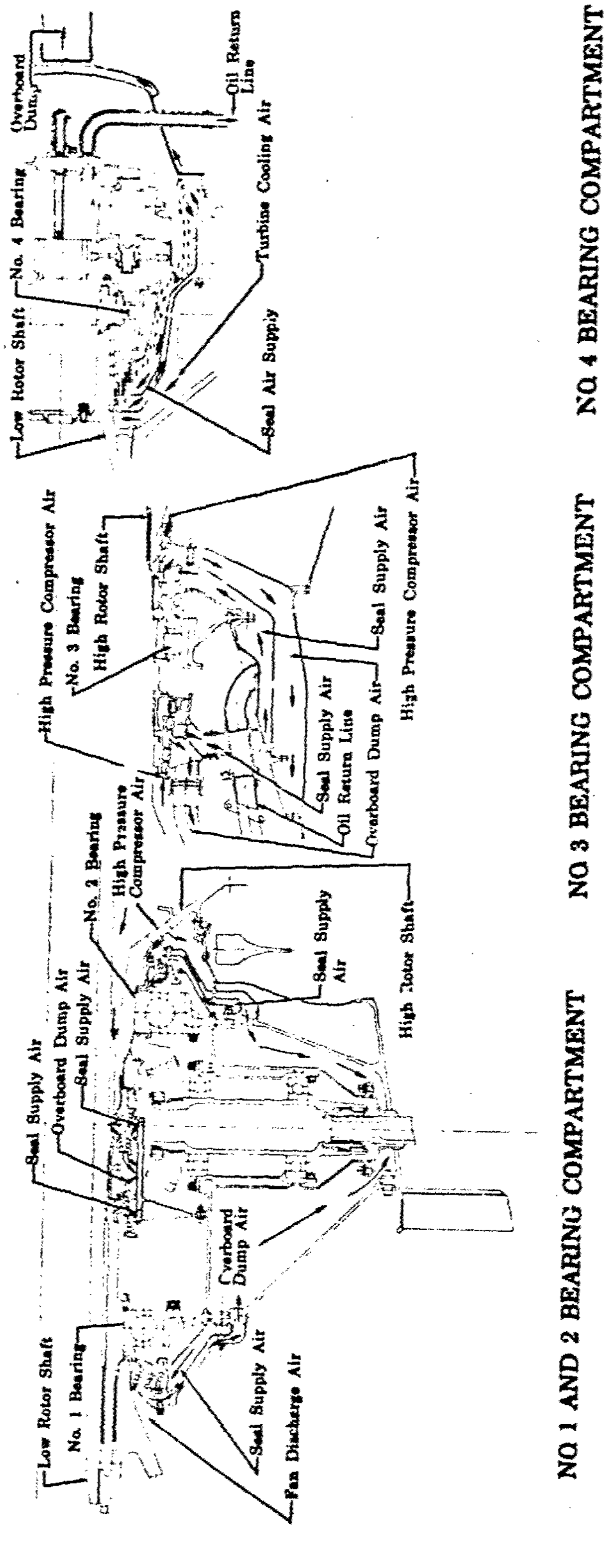


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Figure 8. Labyrinth Seal Comparison

The evaluation of this design also determined that bearing compartment breather pressure could not be maintained at the same level as the oil tank and gearboxes. This is due to the high flow rates associated with this type of system and could result in internal oil loss and possible fire danger. These deficiencies eliminated this system design. However, the durability achieved with the nonrubbing characteristics of this type seal make it an ideal selection for the secondary seal in the selected design where its flow rate through the overboard vent is not critical.

The evaluation of carbon rubbing face seals with labyrinth backups (figure 9) determined this system to be adequate in most respects for the JTF17 engine. However, frictional heat generated by the rubbing carbon seals results in higher heat rejection rates and seal wear due to seal plate rubbing as in current commercial engines.

During Phase II-C the initial experimental engines were operated with carbon rubbing seals and showed that this design would satisfy engine requirements. However, the bearing compartments have been designed to facilitate the easy interchange from carbon rubbing seals to the hydrostatic seal design because preliminary testing in the J58 program has shown that the hydrostatic seal has the advantage of less wear and lower heat rejection. The carbon rubbing seals will be retained as a backup design in the engine development program.

Preliminary testing and analysis shows that the hydrostatic seal system with labyrinth backups meets all of the requirements and exceeds the capabilities of the other systems considered, therefore, it was selected for the JTF17 engine. Bearing compartment designs with this system are shown in figures 1, 2, and 3. Included on these figures are the airflow paths, pressures and temperatures for the most adverse flight condition.

A comparison of the capabilities of the hydrostatic seal system with the other candidates systems is shown in table 1. The very low flow rate of the hydrostatic seal system in conjunction with low supply air temperatures (figure 10) produces minimum heat rejection rates, as shown in figure 11.

Table 1. Comparison of Seal Systems

	Item	Carbon Seals	Labyrinth Seals	Hydrostatic Seals
1.	Maximum lubrication system mechanical heat rejection	4550 Btu/min	6850 Btu/min	3300 Btu/min
2.	Oil flow	145 lb/min	195 lb/min	125 lb/min
3.	Maximum seal pressurization flow (% high comp air)	0.38%	0.88%	0.03%

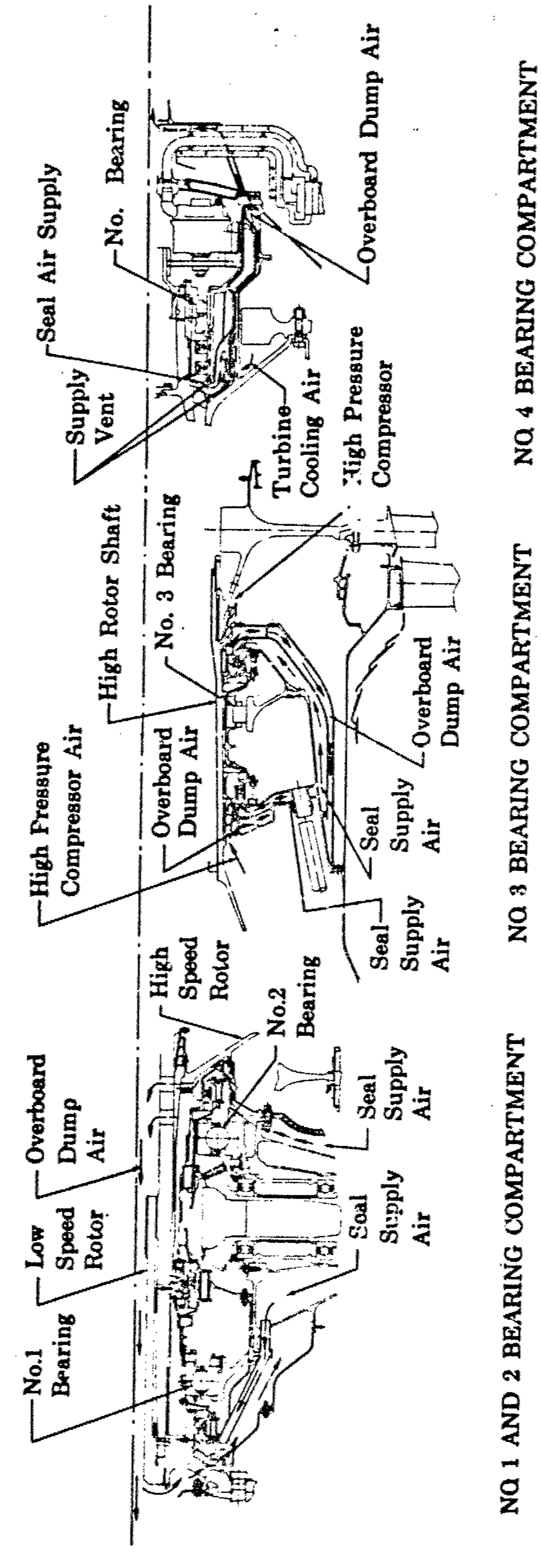

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Figure 9. Pressurized Carbon Seals

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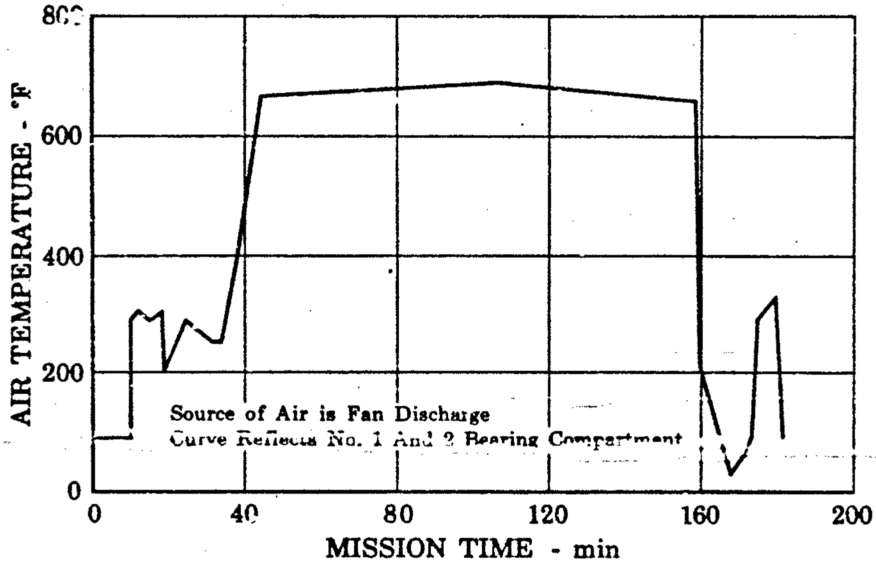

Figure 10. Seal Pressurization Supply Air Temperature for a Typical Mission FD 16385 BIIG

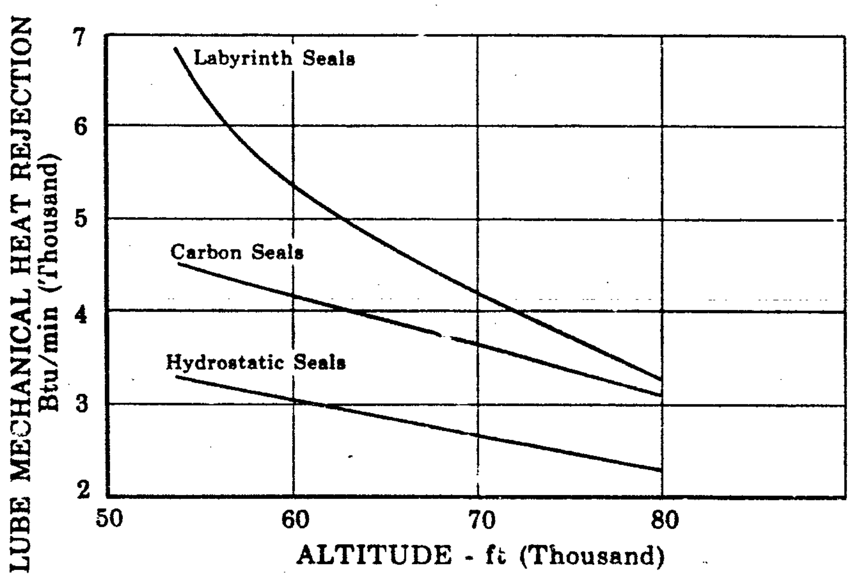

Figure 11. Lubrication Mechanical Heat Generation FD 16386 BIIG

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The hydrostatic seal system development program for the JTF17 engine will use any applicable results obtained from the J58 program and from NASA contracts NAS3-7605 and NAS3-7609 which are for advanced seal development programs using the hydrostatic seal principle.

Testing of hydrostatic seal systems at hot environmental conditions simulating J58 engine bearing compartments at flight conditions more severe than required for the JTF17 has shown that:

1. The hydrostatic seal operating characteristics can be predicted analytically.
2. Hydrostatic seals have no rubbing surfaces except on start-up.
3. Hydrostatic seals will operate successfully throughout the flight envelope.

(2) Product Assurance Considerations

Specific features of the bearing compartment seal system that enhance the JTF17 engine in the areas of Maintainability, Reliability and Safety are as follows.

(a) Maintainability

1. Maintenance is simplified because all three bearing compartments are accessable without major engine disassembly. The forward bearing compartment which houses No. 1 and No. 2 bearings is easily reached by removing the front fan. The aft compartment which houses the No. 4 bearing can be inspected and maintained by removing the tail cone. The middle compartment which houses No. 3 bearing is accessible after the turbine assembly is removed as a balanced unit. The accessibility of all three compartments is a significant design improvement over present commercial engines.
2. The material selected for the hydrostatic seal ring is a high temperature carbon used successfully in conventional carbon rubbing face seals. One of the primary goals established for the hydrostatic seal development program is to develop a material that will be more durable during normal maintenance and handling. Initial studies show that composition materials fabricated from self-wicking sintered metals and carbon can be used successfully. During Phase III this development will be accelerated in an effort to eliminate the brittle type carbon material from the seal system.

(b) Reliability

Reliability of the pressurized and vented seal system is assured by:

1. The use of a secondary labyrinth air seal and overboard vent that routes high temperature air overboard thereby preventing it from reaching the bearing compartment.

2. The hydrostatic primary seal which has the capability to act as a conventional carbon rubbing face seal in the event of loss of supply air pressure through the use of preload springs that force the seal ring against the seal plate similar to current carbon rubbing seals.

3. The seal plate which is cooled with a positive flow of oil through closely spaced holes within the plate to retard wear in the event of seal ring and seal plate contact.

(c) Safety

Safety features of the pressurized and vented hydrostatic seal system, in addition to features noted under reliability, are:

1. A positive pressure drop across the primary seals maintains a positive pressure differential in the bearing compartment throughout the flight envelope.

2. Secondary labyrinth seals prevent high temperature air from reaching the primary seal and the bearing compartment.

3. The overboard vent which discharges seal leakage air and/or possible oil vapors to an external location prevents oil or vapors from entering the main engine gas path and cabin air bleed.

4. Positive sealing of the bearing compartment at all flight conditions reduces oil oxidation and prevents dirt from getting in the oil.

5. Low supply air temperatures and flow rates give minimum heat rejection and prevent thermal degradation of the lubricating oil.

(d) Material List

Material selections for the primary and secondary seals are shown in table 2. Materials used in the supply and vent lines are listed in paragraph I of this report.

Table 2. Materials

Component	Material
1. Carbon seal ring	High temperature carbon (CDJ83)
2. Carbon seal carrier	Inconel 718 (PWA 1010)
3. Rotating seal plate	Inconel 718 (PWA 1010)
4. Seal plate coating	Chrome carbide (PWA 50)
5. Secondary seal knife edge	Waspalloy (PWA 1016)
6. Secondary seal land	Hastelloy N (PWA 1012)
7. Torque pin sleeve	L-605 (AMS 5759)
8. Primary seal piston rings	Cast iron (PWA 790)
9. Preload springs	Inconel X (AMS 5699)

3. Accessory Drives

a. Description

Accessory and external component drives are provided by three tower-shafts driven by the high rotor through a bevel gear system as illustrated in figure 12. One towershaft drives an accessory gearbox located on top of the engine, providing power takeoff drives for airframe accessories and for engine starting. A left side towershaft drives the main engine gearbox. This gearbox drives the main fuel pump, the unitized main fuel control, and the engine hydraulic pump. A right side towershaft drives the main engine oil pump, and an airframe air pump for the Lockheed installation. Orientation of these drive provisions differs slightly for the Boeing and Lockheed engines in order to secure an installation that is most compatible with airframe requirements.

The design of these drive provisions has incorporated J58 experience with high temperature gears and drives. This experience is reflected in the materials selected and the design criteria used.

b. Design Objectives and Requirements

The design objective of this system is to provide a reliable, safe means for power extraction for aircraft requirements, engine accessory requirements and for engine starting. Specific requirements are:

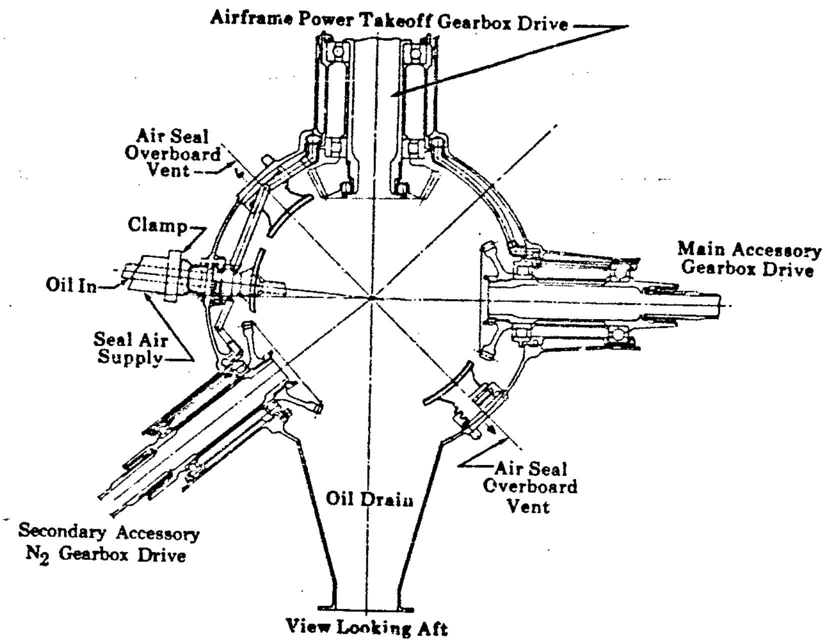
1. To provide life consistent with a 50,000 hour objective life as defined in Section I of this volume.

2. To provide an external engine configuration consistent with engine and installation requirements.

3. To provide a main engine accessory gearbox that will:
   a. Provide support and drive for the gas generator fuel pump (85 HP), hydraulic pump (120 HP), N_2 tachometer, engine lubrication system deoiler, and gearbox scavenge pump.
   b. Provide drive for the unitized fuel control.
   c. Provide a stubshaft drive to allow manual rotation of the high compressor.

4. To provide a secondary N_2 gearbox that will:
   a. House and drive the engine oil pressure pump and a three element oil scavenge pump.
   b. House the lubrication system filter and oil pressure regulator valve.
   c. For the Lockheed installation only, drive and partially support an airframe air-conditioner compressor (approx. 300 HP, 9500 rpm)

Boeing



Lockheed

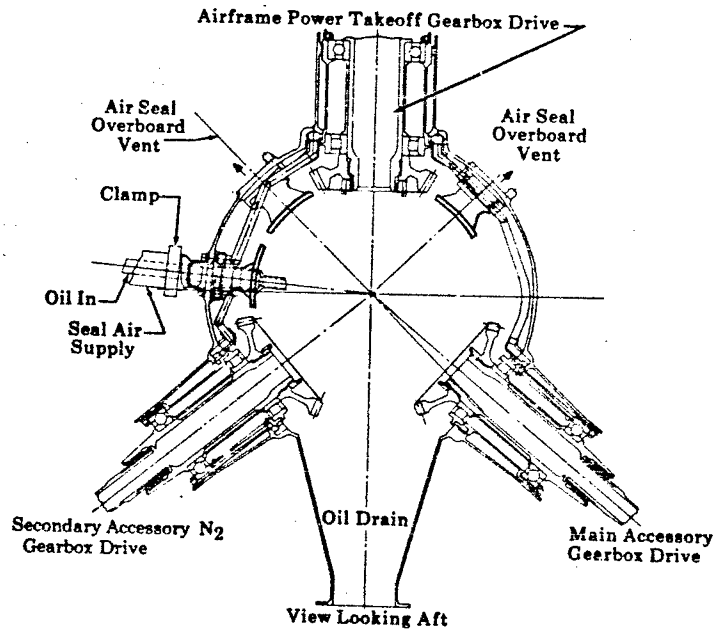


Figure 12. Bearing Compartment

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5. To provide a PTO gearbox that will:

   a. Provide a drive system capable of transferring power for starting the engine from an airframe mounted starter to the engine.

   b. Meet the airframe requirements for accessory drive given below: The Boeing Company requires a fixed gearbox that supplies the support and drive for two airframe hydraulic pumps and an airframe accessory drive. For the hydraulic pump the requirement is to supply two pads according to specification AS 519 that will drive at approximately 4500 rpm and transmit 20 horsepower each. For the power takeoff the requirement is to drive at approximately 8000 rpm and transmit 550 HP continuously. This pad must also conform to specification AS 519.

   The Lockheed California Company requires a power takeoff gearbox which includes (1) provision for coupling to an airframe-furnished flexible shaft, (2) a remotely actuated decoupler, and (3) provision for rotation of gearbox housing for right and left installation. The PTO drive must rotate at approximately 9250 rpm and deliver 550 HP continuously.

c. Design Approach

(1) Detailed Description

The engine main accessory gearbox is shown in figure 13, and is common to both installations. This gearbox provides drive capability for the gas generator fuel pump, the engine hydraulic pump, the engine unitized fuel control, the N_2 tachometer, and the lubrication system deoiler.

The engine secondary accessory N_2 gearbox for each installation is shown in figures 14 and 15. In the Boeing installation the purpose of this gearbox is to contain the oil pressure pump, oil filter, oil pressure regulating valve, No. 1 and No. 2 bearing compartment scavenge pump, and the No. 3 bearing compartment scavenge pump. In the Lockheed installation the purpose of the gearbox is the same as for the Boeing installation, with the added requirement for supporting and driving an air compressor that is part of the airplane air-conditioning system.

The main and secondary N_2 gearboxes are mounted directly to the engine intermediate case by means of three attachment points. (See figure 16.) Hinged joints are located fore and aft of the towershaft pad on the towershaft centerline. A drag link is located from the forward corner of the gearbox housing to the engine case. In addition to the three attachment points, a vertical alignment lug is located directly in line with the towershaft pad. This mounting system provides complete gearbox restraint without inducing stresses in the gearbox or accessories because of mechanical or thermal deflections of the engine.

The PTO gearbox provides drive capabilities for airframe accessories and the engine starting. The Boeing Company and Lockheed California Company have different requirements, and gearboxes for the two installations are shown in figure 17. PTO gearboxes are mounted directly on the engine intermediate case by a bolted flange connection around the towershaft drive.

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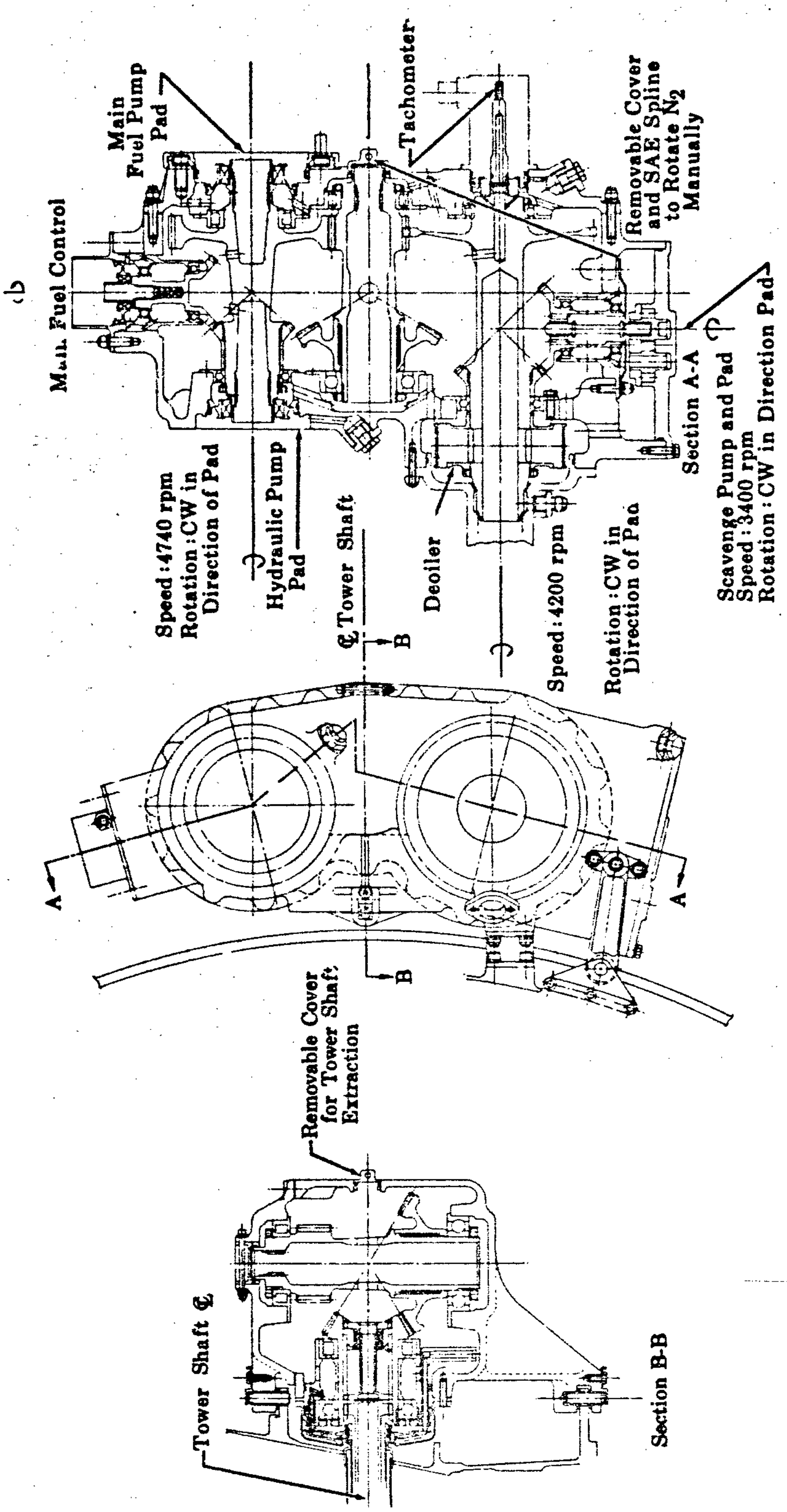

Figure 13. Main Accessory Gearbox and Lockheed

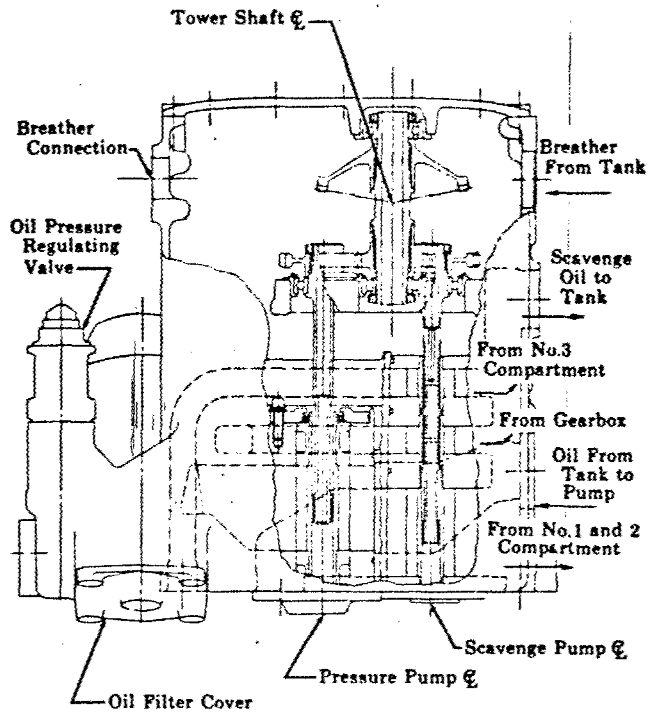

Figure 14. Boeing Secondary Gearbox FD 16855
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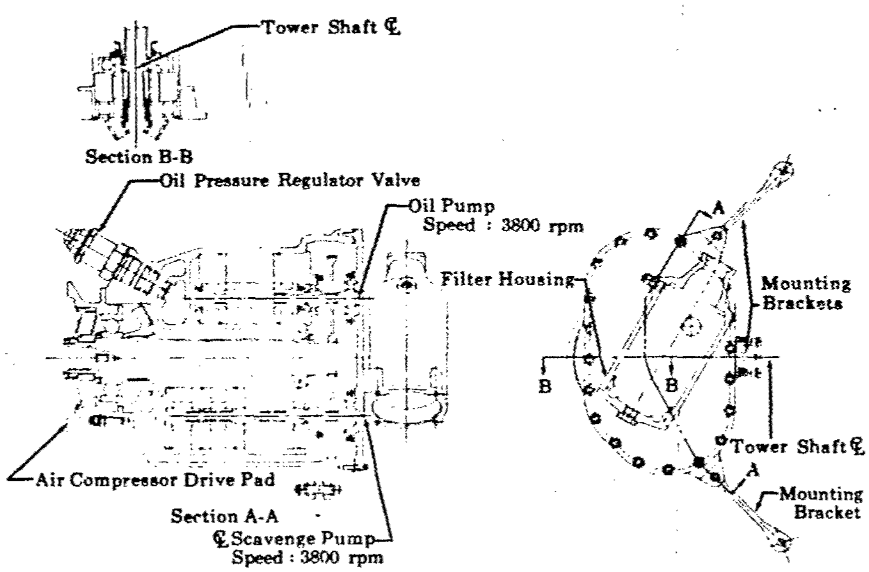

Figure 15. Lockheed Secondary Gearbox FD 16681
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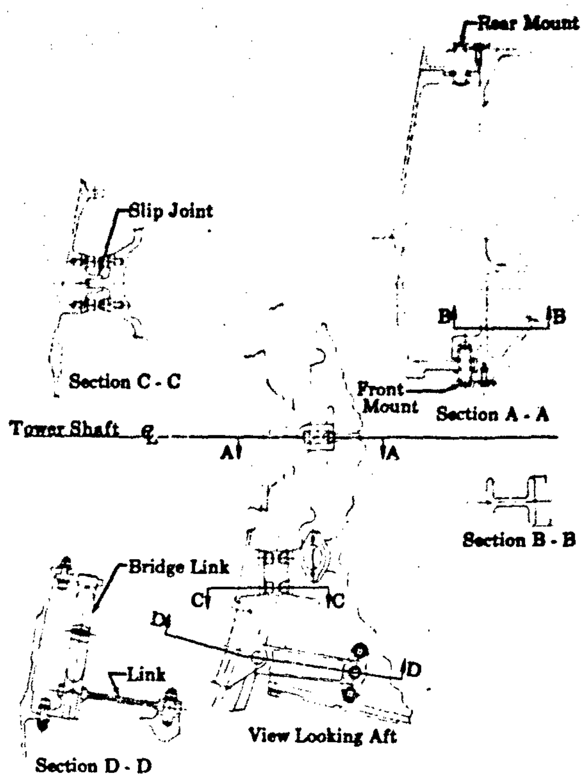

Figure 16. Main Accessory Gearbox Mounting FD 16424 BIIG

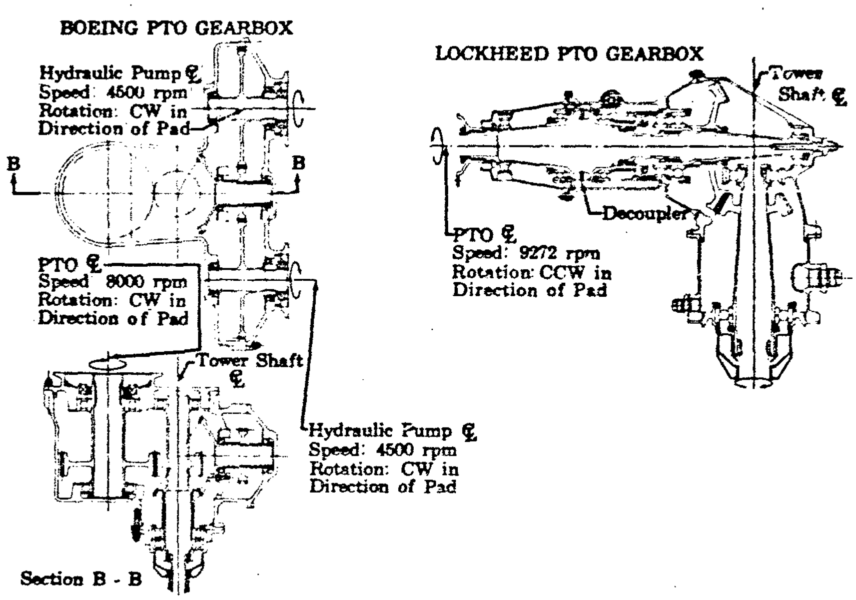

Figure 17. Power Takeoff Gearbox - Boeing and Lockheed FD 16676 BIIG

All production gearbox housings are cast titanium. This material provides the best strength-to-weight ratio at the temperatures encountered, and utilizes the stiff supporting characteristics of the thick walled cast construction. During Phase II-C an investigation was initiated to improve casting techniques used for titanium gearboxes. Present casting technology is illustrated by figure 18. Present casting of gearbox size units would result in (1) thickwall castings that require machining if minimum weight is to be maintained, (2) the limiting of possible core configurations for internal oil passages and (3) high unit cost. This investigation will be continued during Phase III. Until these casting techniques are thoroughly proven, early prototype engines will be fitted with fabricated titanium housings. Fabricated titanium gearbox housings have been used successfully throughout the J58 development and production program.

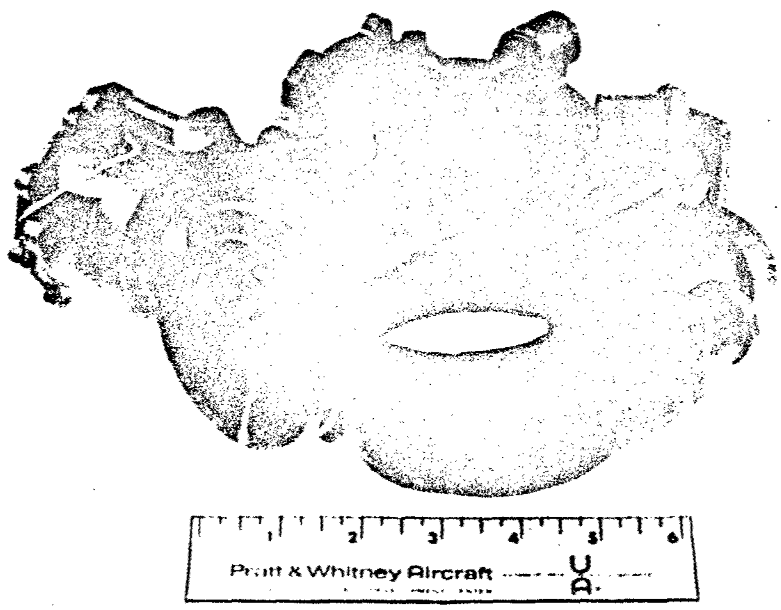

Figure 18. Titanium Casting FE 61376
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Gearbox lubrication is supplied directly from the main engine oil system. Oil is introduced through jets located at strategic points in the gearbox and is distributed through the various shafts to the gears, bearings, and seal faces for lubrication and cooling. All accessory drive splines are provided with positive lubrication to reduce wear and binding. In addition to the oil that is delivered directly to the bearings and seals, sufficient oil is dispersed throughout the gearbox to cool the housing walls to a maximum housing wall temperature of 450°F.

The main and secondary N_2 gearboxes are drained by a gear-type scavenge pump at the bottom of each gearbox housing. The scavenge pump is accessible for inspection or replacement without removal of the gearbox or accessories. Maximum discharge oil temperature is 380°F.

All shafts and gears are straddle-mounted on antifriction bearings that are designed to support the required radial and axial loads for a

minimum B-10 life expectancy of 10,000 hours as defined by the AFBMA (Anti-Friction Bearing Manufacturers Association). Bearings are heat-treated at 1000°F to stabilize them for minimum distortion at operating temperatures. All bearing cages are steel and are silver plated. These bearing materials are used in all J58 accessory drives and have demonstrated reliability and long life in development and service. In J58 use, accessory bearings operate at temperatures of 600 to 650°F. In the JTF17 the operational temperature will be approximately 430°F (170 to 220°F cooler).

All rotating bearings are designed so that the inner race rotates. The inner races have tight fits and positive retention on their associated shafts. All outer races are either flanged and bolted, or are set in replaceable bolted steel liners with closely controlled fits and mechanical locking to prevent rotation. Thrust bearings are provided on all shafts, and provision for bi-directional thrust is provided where necessary. All bearing liners are mechanically retained in the housing casting to minimize gear and bearing misalignment due to load and thermal deflections. Bearings are set as far apart as possible to minimize misalignment due to tolerance and thermal deflection. Maximum allowable misalignment between meshing gears is limited to 0° 07' under most adverse conditions to avoid undesirable mesh conditions that result in excessive gear wear and high bearing loads.

The balanced tooth concept is used for all gear design. With this concept, the endurance beam strength of the pinion and gear in a mesh is equalized to give maximum possible life. Tooth form layouts are made to establish the best tooth properties for the design application with respect to wear and strength, contact ratios, and the ability to run without interference under adverse mounting conditions. All gear tolerances are governed by specification PWA 350 to ensure accuracy of the gear teeth with respect to tooth form and tooth errors that may affect dynamic tooth loading. All gear teeth and gear webs are shot peened to increase fatigue life.

All gears and shafts are designed of material with the following properties:

	psi
Ultimate stress	150,000
0.2% yield stress	130,000
Endurance strength	65,000
Shear stress	78,000

Hertz compressive stress criteria for gear teeth are:

	Bevel Gears, psi	Spur Gears, psi
Continuous	90,000	105,000
Overload	120,000	135,000
Starting	140,000	155,000

Gear and shaft stress values are comparable to those used in the J58 engine and have been verified by over 32,000 hours of development testing

on engines and rigs. Of this, more than 12,000 hours have been at high Mach number environmental and oil temperatures. In addition, substantial service operation is being accumulated at high temperature. The J58 gears and shafts are designed to operate at temperatures up to 725°F. The JTF17 gears and shafting will operate approximately 300°F cooler (430°F maximum).

All splines are standard specification AS 84B half-depth involute splines, except where misalignment is the primary concern. Shafts subject to misalignment, such as the towershaft, have special full-depth splines to give maximum working depth for a given face width. Non-working splines are unhardened, double-piloted, and secured. Working splines are case hardened, ground where possible, and lubricated.

Bearing stress limits for splines are:

a. Case hardened loose splines subject to up to 0.008 in/in misalignment - 3800 psi.

b. Soft spline (RL30 min) with a hand press fit or sliding fit secured by bolt or nut and double piloted - 12,500 psi.

All accessory pads are sealed with replaceable cartridge type seals similar to the J58. The seal carbon ring material was chosen for its low frictional coefficient and its oxidation resistance at elevated temperatures. Seal plates are integral with the accessory drive shaft, where possible, to reduce weight and to simplify configuration.

Seals and seal plates follow the design of those developed for, and used in, J58 engine accessory drives. During J58 engine development various carbon grades and types and various seal plate materials and coatings were evaluated. In addition to development testing as gearbox components, individual cartridge seal rigs were operated in excess of 2600 hours. In J58 engine operation the seals are subjected to temperatures of 650 to 750°F. In JTF17 engine service the seals will be subjected to maximum temperatures that are approximately 200°F cooler (450°F).

Carbon seal faces are finished flat to within 3 helium light bands with a surface finish equivalent to 4 AA according to PWA specification 351. Seal plates are finished flat to within 2 helium light bands with a surface finish equivalent to 5 AA according to PWA specification 351 and are held square to the shaft bearings within 0.0005 in. FIR.

All accessory drives and their associated power train have been designed to conform with current aeronautical standards governing the capabilities of the various accessories as follows:

1. Individual accessory drives are capable of delivering the rated power continuously from ground idle to maximum speed, and can withstand a static torque equivalent to five times the rated power at ground idle speed without failure or permanent deformation.

2. The power train is capable of delivering the rated power to all associated accessory pads continuously from ground idle to maximum speed, and can withstand a torque equivalent to five times the continuous rated power of any one accessory in addition to the continuous rated power of all other associated accessories at ground idle speed without failure or permanent deformation.

Components within the starting train are sized to reach yield stress at a minimum factor of 2.65 times maximum starting torque. Shear sections are sized to reach ultimate stress at a factor of 2 times maximum starting torque. The factor of 2 times the starting torque for the shear section is selected to handle impact loading that may occur during starting. The factor of 2.65 times the starting torque is to ensure that the shear section will fail first, thereby protecting components within the starting train. These factors, proven to be acceptable and reliable margins, are used on current P&WA commercial and military engines. Starting torque requirements for the Lockheed California Company and The Boeing Company engines are shown in figures 19 and 20.

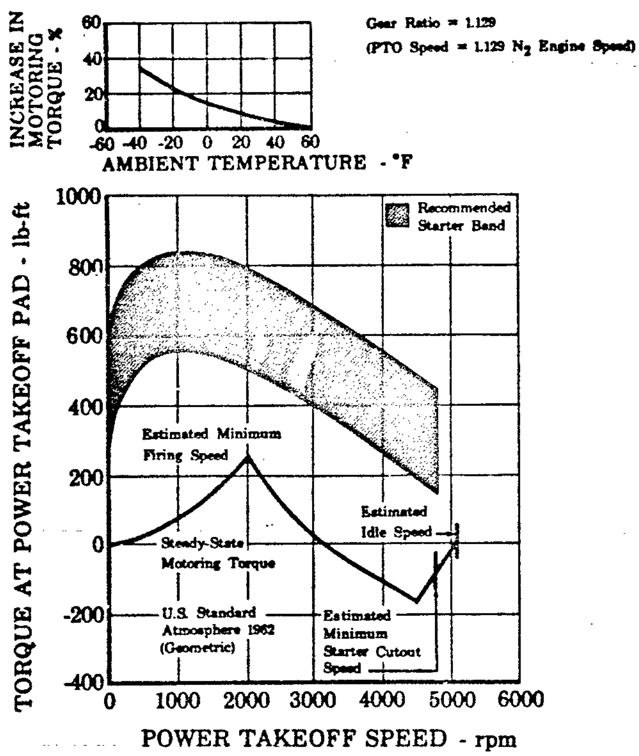

Figure 19. Starting Requirements - Lockheed FD 16399 BIIG

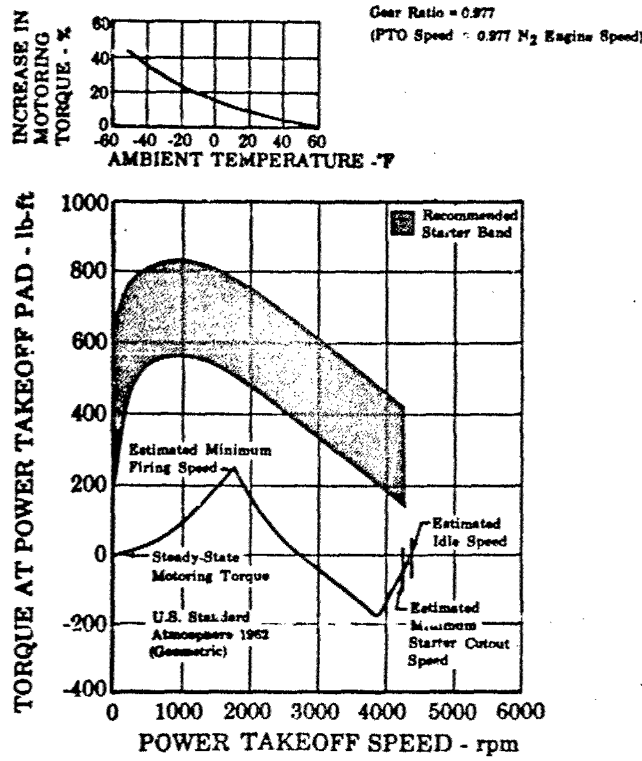

Figure 20. Starting Requirements - Boeing

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Critical speed of shafts and gears has been designed to be a minimum of 1.2 times the maximum operating speed. All shafts and gears are designed to minimize resonant frequencies within the operating range. Prior to final design, each gear and shaft design has been individually tested to determine the natural vibratory frequencies and modes. Resonance of the 1st and 2nd nodal frequencies are avoided entirely, and frequencies of the higher nodal orders have been shifted to points of transient operation by recontouring, remounting or damping.

The lubrication system centrifugal deoiler (figure 21) is of the "pin wheel" type. It is based on development testing conducted for the J58 engine program. Measured oil loss and pressure drop of a production J58 deoiler (which is completely adequate) compared to the equivalent pin wheel design are shown in figures 22 and 23. The data indicate that this design is significantly more efficient in oil separation and has a reduced pressure loss. This efficient design will minimize oil loss through a broad range of breather airflow and pressure.

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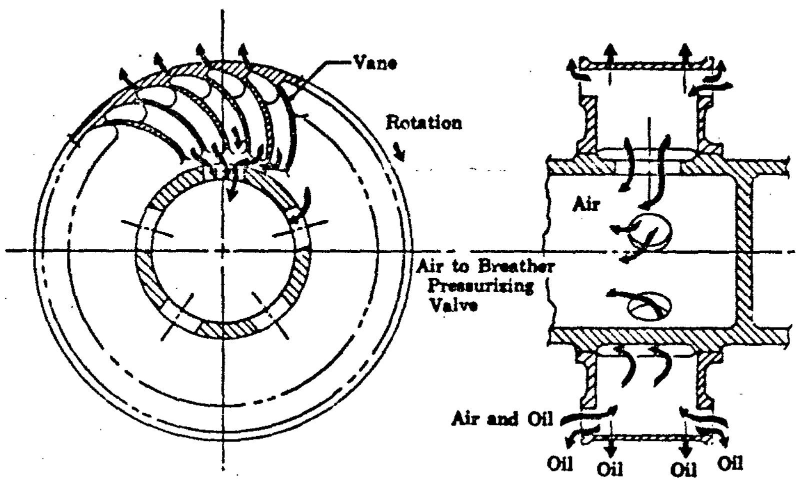

Figure 21. Centrifugal Deoiler FD 16423
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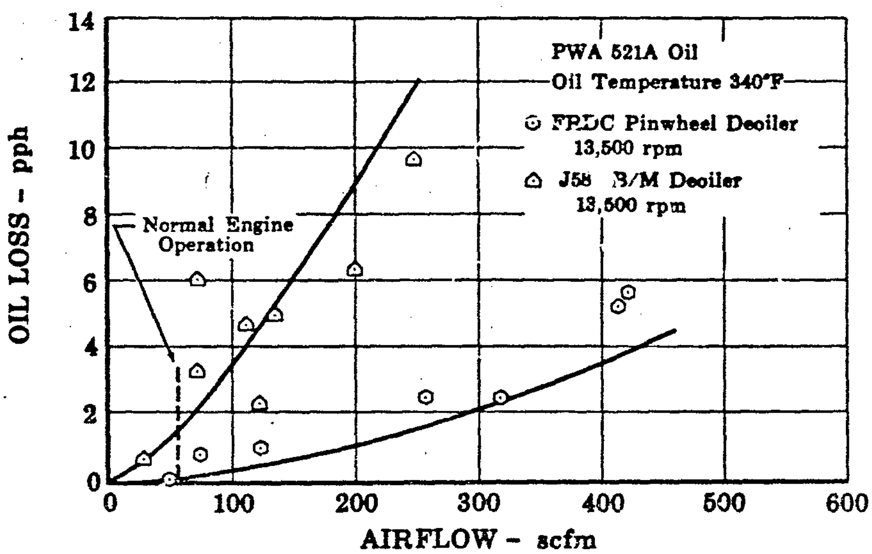

Figure 22. Deoiler Oil Loss Comparison FD 16401
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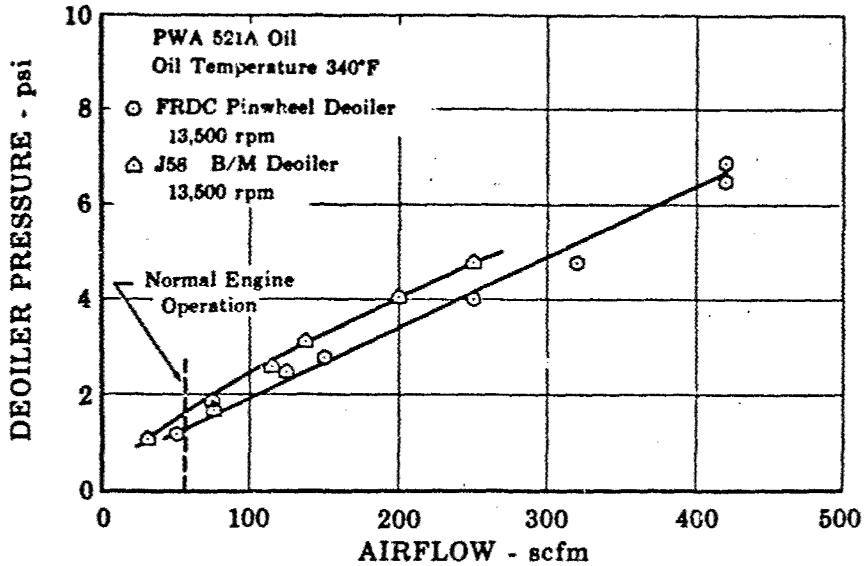


Figure 23. Deoiler Pressure Drop Comparison FD 16402 BIIG

(2) Product Assurance Considerations

Maintainability and reliability considerations of the design include the following:

1. In the main accessory gearbox the provision for manual rotation of the rotor is made to allow effective borescope inspection of both compressor and 1st-stage turbine blading without removing the gearbox or accessories.

2. In the main and secondary N_2 accessory gearboxes the provision to disengage the transfer shaft is made to allow access to, and replacement of, the No. 2 bearing without the removal of the gearbox and attached accessories.

3. Accessory pad cartridge-type carbon face seals are accessible for inspection or replacement without gearbox disassembly.

4. All tapped holes in the housings are provided with replaceable steel inserts, and all brackets, attachments, or other areas subject to wear are separable from the housing or are provided with removable bushings.

5. A magnetic-type chip detector is located in the sump area of each gearbox housing. The detector is accessible for inspection without removal of any other engine components.

6. The scavenge pumps, oil pressure pump, and oil filter are individual component assemblies that fit into cavities within the gearbox housings. They are readily removable for inspection, cleaning, or replacement without further engine disassembly.

7. Bearings are either directly bolted into the housing or are retained in bolted in, replaceable liners.

8. Accessory drive splines are lubricated.

9. Gear teeth and gear webs are shot peened to increase fatigue life.

The following safety considerations were used in the design of the accessory drives:

1. Proved and standard safety margins are incorporated into the gear and drive train designs.

2. Critical and resonant frequencies are not in normal operating speed ranges.

3. Temperature resistant materials are used for gearbox housings.

4. Bearings inner and outer races are retained and restrained against uncontrolled race spinning or unseating.

5. Shear sections are provided to protect internal engine components in the event of loads exceeding design requirements and margins.

6. Overboard drain provisions are incorporated between each gearbox drive pad and driven accessory to prevent contamination of oil in the event of accessory shaft seal failure.

(3) Material Summary

1.	Gearbox housings	Titanium (see Design Approach)	
2.	Gears and Shafts	Alloy Steel	PWA 724 or PWA 742*
3.	Bearings		
	Races	Alloy Steel	PWA 724, PWA 742* or PWA 725
	Balls/rollers	Alloy Steel	PWA 724, PWA 742* or PWA 725
	Cages	Alloy Steel	AMS 6415
4.	Cartridge Seal		
	Carbon Ring	High Temperature Grade	CDJ83
	Seal Face	Alloy Steel	PWA 724 or PWA 742*
5.	Gearbox Mounting Links	AISI 410 Stainless Steel	AMS 5613
6.	Centrifugal Deoiler	Cast AISI 410 Stainless Steel	AMS 5350

*PWA 742 (new specification) is same as PWA 724 except vacuum melted.

H. STRUCTURAL/MECHANICAL

1. General

The preceding paragraphs of Section II of this Report contain a detailed description of the individual major components of the JTF17 engine. Those items affecting overall engine configuration that have not been previously described are included in this paragraph. The subjects discussed are bearing and structural tests, engine mounts and thrust bearing design criteria, engine thrust balance system, windmilling thrust balance load analysis, vibration characteristics, field inspection and repair capability, and features to extend the service life of the engine structure. A brief description of the maintainability features has been included because of their direct influence on the engine configuration, even though they are discussed in detail in Volume IV, Report F, Section I (Maintainability). Unless otherwise indicated, all descriptive material applied to both prototype and production engines.

2. Bearing and Structural Tests

a. Description

Structural tests conducted during Phase II consisted primarily of rig evaluation of bearings and test stand operation of the engine. Past experience has shown that any inadequacies in the design are usually discovered during this initial evaluation. The early detection of possible problem areas has, in the past, reduced the number of changes required as a result of the full-scale engine testing. Because laboratory evaluation to verify the structural adequacy of individual engine components is planned for Phase III, information gained in other engine development programs, such as the J58 and JT3D, was relied upon for the current JTF17 design. Furthermore, the successful high-temperature turbine used in the J58 engine laid the groundwork for the structural design of the JTF17 turbine. The laboratory and engine test data obtained in these programs were applied directly to the SST engine.

b. Test Objective

The general objective of the bearing test program was the initial evaluation of the main rotor bearings in a standard P&WA acceptance test. The specific objective consisted of operating each bearing under load at design speed.

c. Test Results

A summary of the test conditions and results for each of the main rotor bearings tested is given in table 1. Ball bearings are used in positions No. 1 and No. 2 and roller bearings are used in positions No. 3 and No. 4. A range of thrust loads was applied to the ball bearings during the test to simulate engine conditions; roller bearings were subjected to bearing preload only.

Table 1. Main Bearing Test Results

Position	Bearing Function	Maximum Test Speed (rpm)	Test Load Type	Test Load Range (lb)	Test Duration (hr)	Results
1	Low compressor thrust bearing	5960	Thrust	2800	37.5	Successful
2	High compressor thrust bearing	8800	Thrust	2000 to 14,000	6.5	Cage Failure (See Explanation below)
2	High compressor thrust bearing	8800	Thrust	2000 to 14,000	72.8	Successful
3	High turbine shaft radial support	8400	Preload only	2000	21.4	Successful
4	Low turbine shaft radial support	6200	Preload only	500	30.9	Successful

The only difficulty that occurred during these tests was severe wear of the cage-race riding surfaces of the No. 2 bearing (high compressor thrust bearing). A bearing from another vendor, made to the same PWA Specifications, passed the test with no difficulty and was used in the initial engine builds. The successful hardware had a larger clearance in the failure area as compared to the first test part. All subsequent parts have been modified to ensure adequate clearances.

Although the bearings on the towershaft drive were not rig tested during Phase II, engine operation was very successful, and no problems have occurred in this area.

3. Engine Mounts and Thrust Bearing Design Criteria

a. General

The design criteria that we use for engine mounts and thrust bearings have resulted from the experience P&WA has had on many successful engine programs. For example, engine mounts designed to our standards have accumulated over 39 million hours of commercial flight time without a single failure.

b. Engine Mounts

(1) Description

The engine mounting system (figure 1) is composed of front and rear mount rings with airframe attachment points designed to resist all the engine generated loads, as well as the maximum anticipated maneuver and externally applied aerodynamic loads. The mount rings are located in

approximately the same axial position relative to the engine centerline for both the Lockheed and Boeing designs. However, the attachment configurations between the engine and airframe are completely different and are discussed separately below.

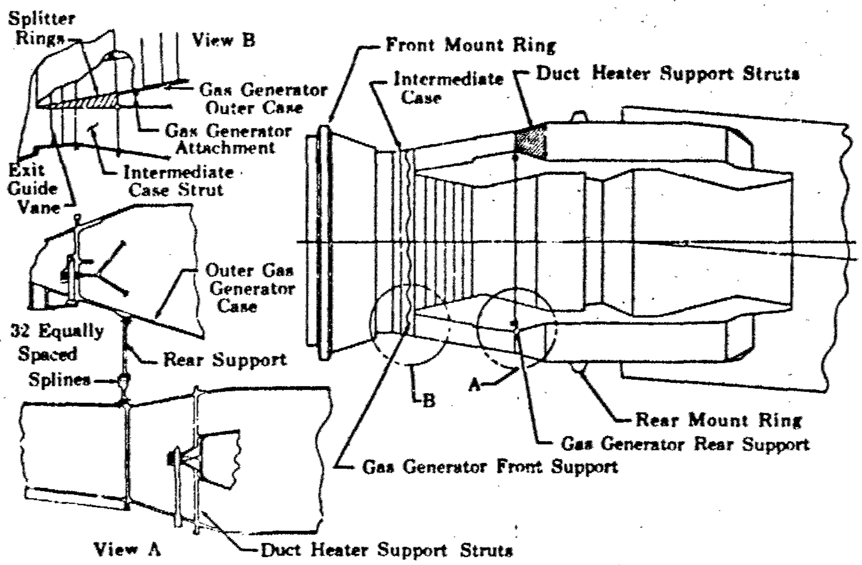


Figure 1. Gas Generator Support Structure FD 16277 BIIH

The gas generator is supported radially and axially by the intermediate case struts and radially by the duct heater support struts, as shown in figure 1. Loads from the gas generator are transmitted through the eight intermediate case and duct heater struts to the outer duct case, which then distributes the load to the front and rear mount rings.

Ground-handling attachment points are provided on both front and rear rings for Boeing and Lockheed as shown on Installation Drawing No. 2128091 and 2128090 respectively.

(2) Design Objectives

The design objectives of the engine mount system are as follows:

1. Provide attachment between the airframe structure and the engine
2. Provide optimum accessibility for maintainability (engine removal and installation)
3. Ensure optimum load distribution in airframe and engine structure.

The gas generator support is intended to:

1. Provide support and alignment for the gas generator
2. Reduce the vibratory excitation of the gas generator (amplitude)

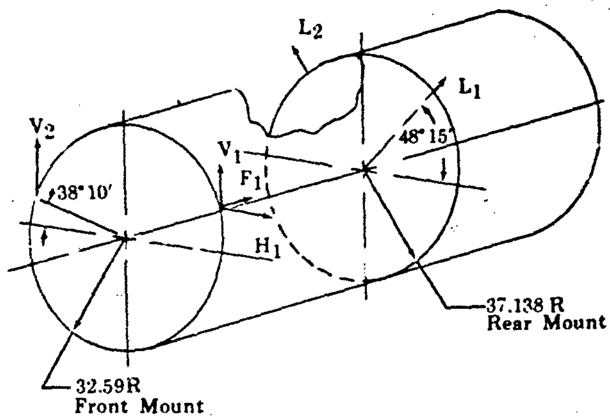
3. Reduce the moment applied to the intermediate case struts under maximum maneuver conditions to minimize gas generator deflection.

(3) Design Requirement

The engine mount configuration is designed to match the mating wing structure and provide maintainability features required by the respective airframe installations. The design requirements are discussed separately for the Lockheed and Boeing installations in the following paragraphs. The primary gas generator mounting is the same for both airframe configurations.

(a) Lockheed Configuration

On the Lockheed installation (shown in figure 2), the front mount ring is located just forward of the fan section with attachment points at 38 degrees above the horizontal centerline on each side of the engine. Axial and side loads can be resisted at either attachment point to facilitate right- or left-hand engine installation while vertical loads are resisted at both points. The rear mount ring is located aft of the duct heater support struts with attachment points located approximately 45 degrees above the horizontal centerline on each side of the engine. To provide ease of installation and maintenance, radial link attachments are used as shown in figure 3.



Distance Between Mount Planes	75.823 in.
Engine Weight	10,132 lb
Engine CG from Front Mount	71.0 in. Aft

The Allowable Limit Loads for Each Mount Attachment Point Must Satisfy All Equations Below:

$\lvert V_1 \rvert$ = 28,000 lb	$\lvert V_2 \rvert$ = 8,000 lb
$\lvert H_1 \rvert$ = 35,000 lb	$\lvert L_1 \rvert$ = 72,000 lb
$\lvert F_1 \rvert$ = 75,000 lb	$\lvert L_2 \rvert$ = 53,000 lb

Figure 2. Lockheed Mount Loads

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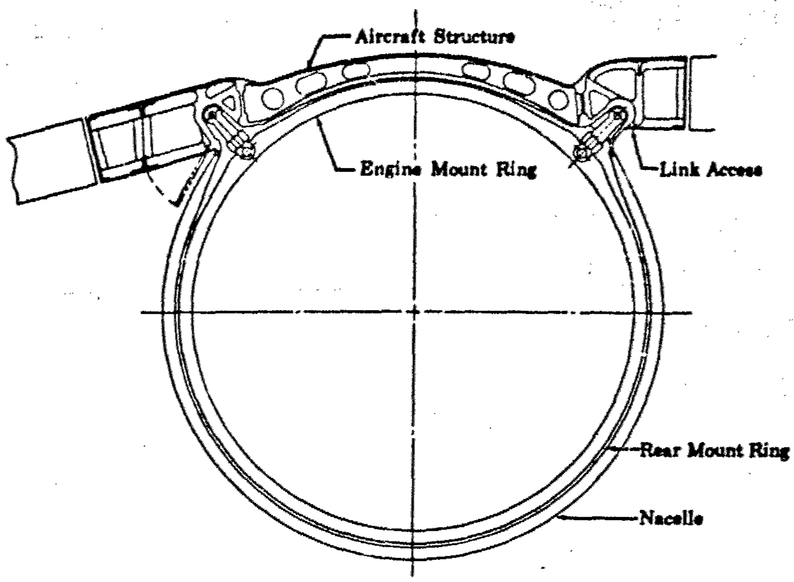


Figure 3. Radial Link Attachment - Lockheed FD 16387 BIIH

A schematic of the mount system with maximum loads is shown in figure 2. The thrust load is transmitted through a ball-and-socket front mount and a thrust link to the airframe. Side loads and yaw moments are resisted by side supports on the front and rear rings. Vertical support points on the front and rear rings carry roll and pitch moments and vertical loads. Bending moment curves for the front and rear rings are shown in figures 4 and 5.

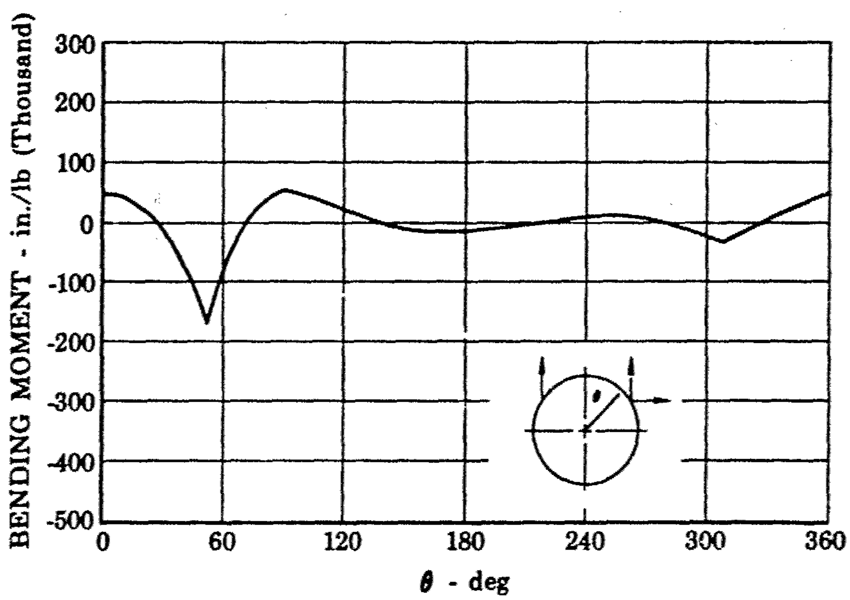


Figure 4. Bending Moment vs Ring Circumferential Position - Lockheed Front Mount Ring FD 16380 BIIH

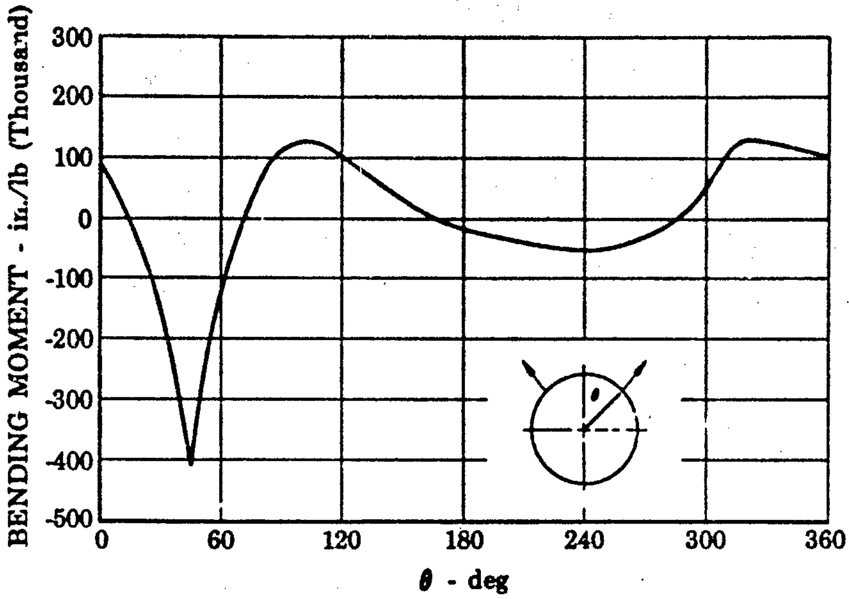
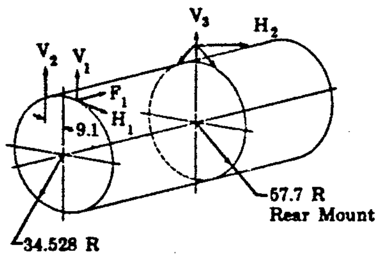


Figure 5. Bending Moment vs Ring Circumferential Position - Lockheed Left Rear Mount Ring — FD 16377 BIIH

(b) Boeing Configuration

For the Boeing installation (shown in figure 6), the front ring is located forward of the fan section with attachment points 9.1 inches either side of the vertical centerline. Horizontal and thrust loads can be carried by either attachment point to facilitate right- or left-hand engine installation, and vertical loads are carried by both points. The rear mount ring is located aft of the duct heater support struts. Attachment points are located 15° on either side of the top vertical centerline with loads applied tangentially.



Distance Between Mount Planes	75.823 in.
Engine Weight	10,950 lb
Inlet Weight	2,600 lb
Engine C G from Front Mount	71.2 in. Aft

The allowable limit loads for each mount attachment point must satisfy all equations below :

$|V_1| \leq$ 91,000 lb

$|V_2| \leq$ 114,000 lb

$|H_1| \leq$ 41,000 lb

$|F_1| \leq$ 122,000 lb

$|H_2| \leq$ 22,500 lb

$|V_3| \leq$ 66,500 lb

Figure 6. Boeing Mount Loads — FD 16276 BIIH

A schematic of the mount system showing the maximum loads is shown in figure 6. Maximum mount reactions occur as a result of aerodynamic loading combined with maximum thrust. Bending moment curves are shown in figures 7 and 8.

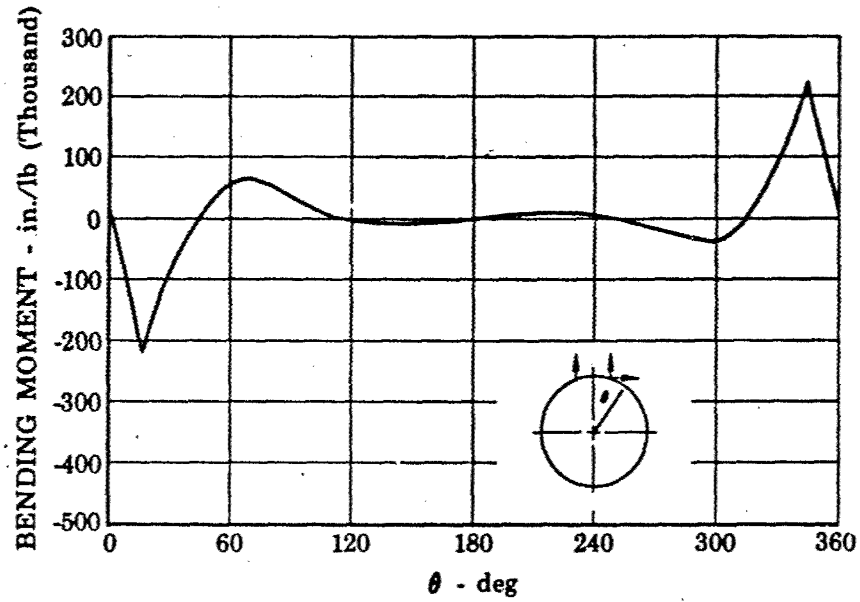


Figure 7. Bending Moment vs Ring Circumferential Position - Boeing Front Mount Ring FD 16378 BIIH

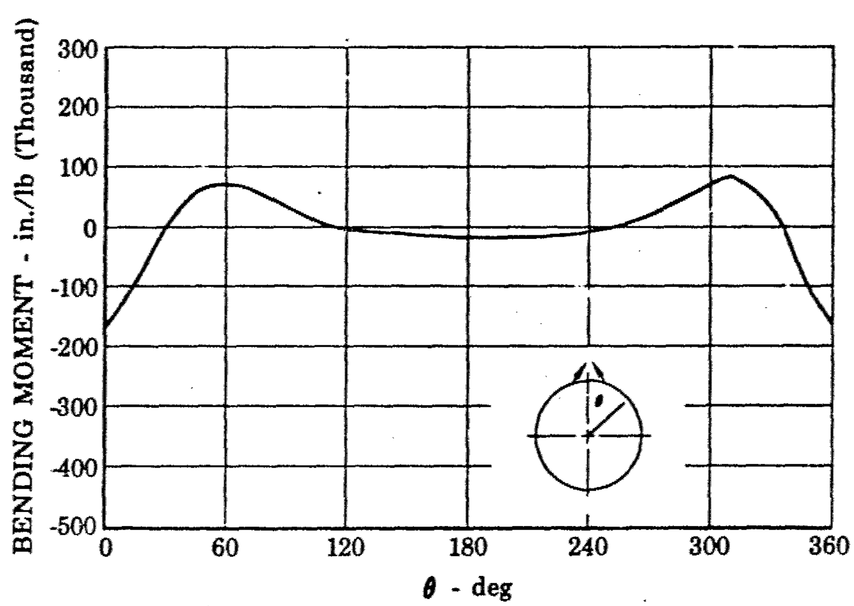


Figure 8. Boeing Rear Mount Ring FD 16379 BIIH

(c) Primary Gas Generator Mounting

The primary gas generator support, which is the same for both the Lockheed and Boeing designs, is shown in figure 1. It is designed to satisfy the following requirements:

1. Allow removal of the outer and inner fan duct and the duct heater as a single unit without a need for access to this area for disconnecting the rear gas generator support

2. Allow adequate room for routing of No. 4 bearing oil plumbing, duct fuel plumbing, and instrumentation leads

3. To provide 360 square inches of airflow area which is required to ventilate the region between the duct heater and the gas generator, and for cooling the ID liners of the duct heater.

(4) Design Criteria

Design loads for the engine mount system, as shown in figures 2 and 6, were based on maneuver conditions specified by the respective airframe manufacturers. (Refer to Report B, Section II, paragraph A, of this Volume.)

Combined stresses from engine, maneuver, and gust loads were limited to 0.2% of the yield strength or 67% of the ultimate strength, whichever is lower. In weld areas, the maximum stress will be equal to or less than 80% of the 0.2% yield strength. This criteria, based on our commercial engine experience, ensures no permanent set at limit load conditions and no failures at ultimate load conditions.

The limiting criteria in the front mount ring was a stress limit of 80% of the yield strength of the material in bending. For the rear mount ring, a deflection limit was established at 0.50 inch maximum to prevent undue flexure of the engine. Local stresses in mount lugs were allowed to reach the 0.2% yield stress.

(5) Material Selection

In both installations, the front mount is fabricated from AMS 4966 (A-110) titanium and the rear mount from AMS 1009 (Inconel 718). These material selections provided a minimum engine weight. Titanium in the front mount ring was dictated by blade containment in the fan outer case. (Refer to Report B, Section II, paragraph A, of this Volume.) Inconel 718 was used in the rear ring for three reasons:

1. Strength at the required operating temperature
2. High modulus of elasticity
3. Ease of fabrication.

The materials used in the rear gas generator support are Inconel 718 for the inner element and AMS 5616 for the outer element. The Inconel 718 was selected because it meets the strength and temperature requirements. The AMS 5616 was used because of its thermal growth compatibility with the mating titanium structure.

(6) Design Approach

The engine mount ring cross section in both installations is trapezoidal in shape (figure 1). This configuration allows for thermal expansion while providing maximum structural strength. The trapezoidal design was chosen because the conventional "U" cross section ring structure used on the JT3D and JT4 was unsatisfactory in the J58 application, while the selected shape performed satisfactorily. These failures were attributed to the larger cyclic temperature gradient imposed on the straight sides of the ring during each engine environmental operating cycle. This larger gradient is due to the higher Mach number operation of the J58 as compared to previous engines.

The primary gas generator is supported at a front and rear location. The front support utilizes the ring structure required in the intermediate case for aerodynamic considerations. This ring, located at the strut mid-span, separates the fan flow from the high compressor flow and also provides the attachment for the gas generator cases (figure 1). The rear support is located axially at the duct support struts. This support allows axial, unrestrained thermal growth of the gas generator, while providing radial support through 32 splines, as shown in figure 1. A detailed description of this support is given in paragraph D of this section (Duct Heater). This feature provides for:

1. Assembly and disassembly of the duct heater without the need for access to the inner cavity
2. Axial and radial thermal expansion between the "hot" gas generator and the "cold" fan duct.

To satisfy the airflow, plumbing and instrumentation requirements — and still maintain adequate stiffness — the rear support structure is a truss configuration.

c. Thrust Bearing

(1) Description

The thrust bearings selected for the JTF17 engine are ball bearings similar to those used in all P&WA engines. One such bearing is used on the high compressor-turbine rotor shaft and one on the low compressor-turbine shaft. Although the thrust balancing system described in paragraph 4, below, is designed to minimize the unbalanced axial thrust by the use of pressure forces, some unbalance always remains and makes the use of mechanical thrust bearings necessary.

Since the target life for the engine is 10,000 hours without major repair, the thrust bearings must have at least that degree of durability. Our approach to achieving this durability is presented below.

(2) Design Criteria

The minimum design criterion life of both thrust bearings is 10,000 hours

(3) Design Approach and Material

A minimum design criterion life of 10,000 hours is far in excess of that used in current turbofan engine design. For example, the JT3D turbofan engine main thrust bearing criterion life is 2000 hours. We expect the JTF17 bearings to be better than those currently used in the JT3D because of the following:

1. The use of improved materials, such as PWA 725 (M-50) and PWA 742 (Bower 315), that have produced good results on the J58, JT3D, and TF30 engines
2. An improved cage design and individually balanced cages similar to the TF30 engine thrust bearing
3. Special attention will be paid to bearing compartment design to provide sealing and packaging that will protect the bearing from corrosive atmosphere and dirt
4. Careful selection of the lubricant to ensure that it is compatible with the SST requirements, coupled with operation at the lowest practicable temperature
5. Housing design flexible enough to minimize the effects of misalignment.

Experience has shown the service life of bearings generally exceeds the minimum criterion design life by a large factor. For example, the design criterion life used for the JT3D engine design was 2000 hours, whereas the TBO for the engine is 8000 hours and the in-flight shutdown (IFS) rate attributable to the high rotor main thrust bearing is 1,340,000 hours.

4. Engine Thrust Balance System

a. Description

The thrust balancing system used on the JTF17 makes use of the pressure differential forces acting on the internal components of the engine (hubs, shafts, and disks), and the aerodynamic forces on the blades and vanes in the compressors and turbines, to minimize axially unbalanced forces. Since these forces cannot be perfectly balanced for all modes of engine operation, rolling element thrust bearings are used to resist any residual unbalance.

The air used for cooling the internal engine components such as turbine disks and blades also supplies the pressure differentials which produce the forces required to balance the rotors. Thus the cooling airflow and thrust balance system must be designed together.

b. Design Objectives

The major design objectives are:

1. To design a system that will produce bearing loads at all flight conditions and power settings that are within required bearing life

2. To prevent the axial thrust load on either rotor from reversing direction at any operating condition to prevent bearing skid
3. To supply cooling airflow to the fan, compressor, turbine disks and blades, and all other engine parts that require a specific thermal environment
4. To limit radial outflow of air into the main gas path, consistent with fan, compressor, and turbine performance requirements.

c. Design Requirements

The major design requirements are:

1. Thrust bearing loads must be commensurate with a 10,000-hour engine use without major repair
2. The thrust balance system must always apply a minimum thrust load to the bearings to prevent skidding.

d. Design Criteria

Because the thrust balance system must be compatible with component cooling and performance requirements, the basic balancing method was formulated early in the engine design program. The following design criteria were established in the initial design stage:

1. Cooling airflow rates that are high enough to ensure that the disk metal temperatures respond at a rate commensurate with the disk low cycle fatigue (LCF) requirements. (These flows must satisfy start-up and in-flight transient and steady-state conditions.)
2. Minimum cooling air outflow to the main gas path
3. Insensitivity of the system to possible variations in seal clearances in terms of effects on bearing loads or cooling flows
4. Minimum thrust load of 1100 pounds on the low rotor and 2400 pounds on the high (The values have been determined analytically. Although they are consistent with past engine experience, substantiation is planned in rig and JTF17 environmental operation.)
5. A system flexible enough to allow for future design changes and engine growth
6. A system easily varied for the purpose of adjusting bearing loads during the engine development program
7. A system in which the pressures in important locations in the engine are known by a correlation with a conveniently measurable location, such as P_{t3}.

e. Design Approach

(1) General

The aerodynamic forces on the compressor and turbine blades are closely related to engine performance and normally cannot be changed solely to accomplish a more desirable thrust balance. Therefore, the static pressure balance of the internal rotating parts (disks, shafts, and seals) is the only means available to control the axial force acting on the rotor thrust bearings. This is accomplished by dividing the engine into pressure "chambers," and controlling the pressure in these "chambers." Interstage labyrinth seals are used extensively to provide both functions (division and pressure control).

Cooling flow paths must be studied simultaneously with the thrust balance design because the internal engine air circulation has to be commensurate with the overall engine cooling requirement. Therefore, a mathematical model in which the complete engine geometry and flow parameters can be dynamically simulated on a digital computer program was established. With this model, which has been verified by extensive testing in the J58 program, a variety of flight conditions, such as Mach number and altitude, can be analyzed in a minimum amount of time. The final thrust balance and cooling method are shown in figure 9. This system will meet all objectives and requirements of the JTF17. The system offers the following advantages:

1. Versatility from the standpoint of future design changes and engine growth
2. A superior labyrinth sealing system that can operate at low clearances, react favorably to rapid thermal transients, and result in loads being insensitive to any reasonable clearance variation
3. Minimum design risk because thrust load forces rely principally on engine station pressures.

An off-design study to determine the effects of changes in the labyrinth seal clearances on bearing load and cooling flows was conducted. This study showed that a 200% variation of flow area in any one seal may occur without seriously affecting the thrust balance. Since this change in seal flow area would not be expected to occur in normal operation of the engine, it can be concluded that the present design is very insensitive to seal leakage. Figure 10 shows a typical seal clearance sensitivity plot for the turbine high rotor forward seal. It indicates a 10% change in thrust balance force for a 200% increase in seal flow area.

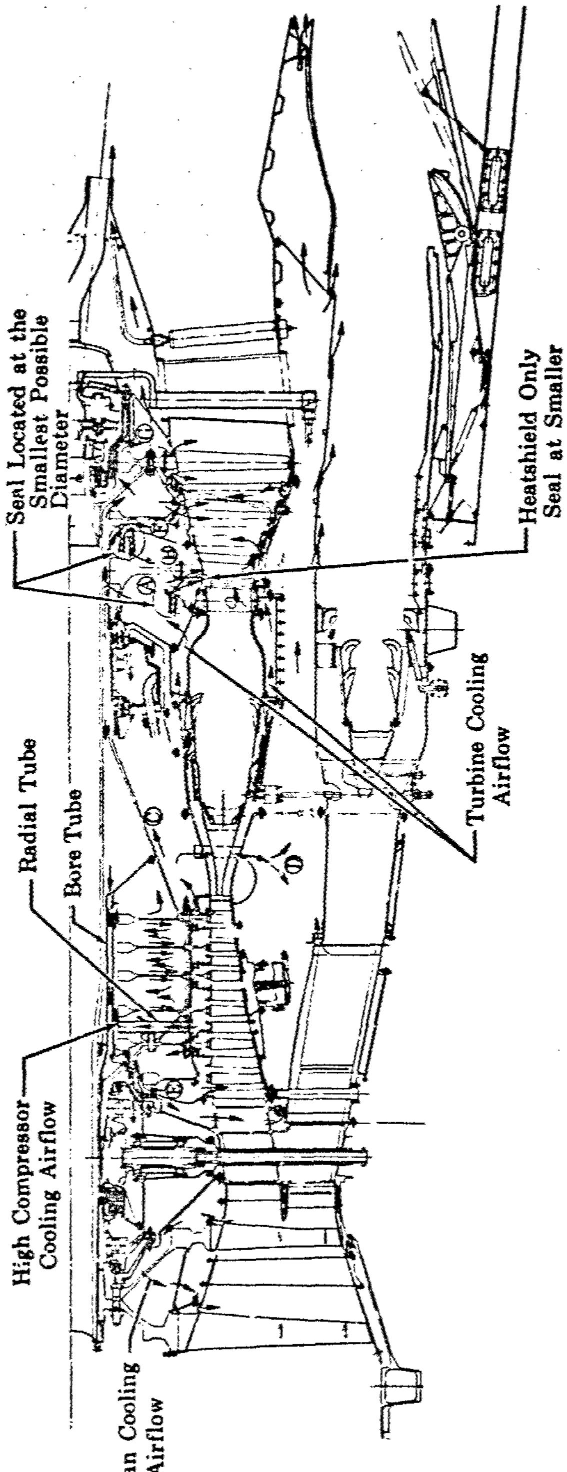


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Figure 9. Thrust Balance Air Flow Schematic

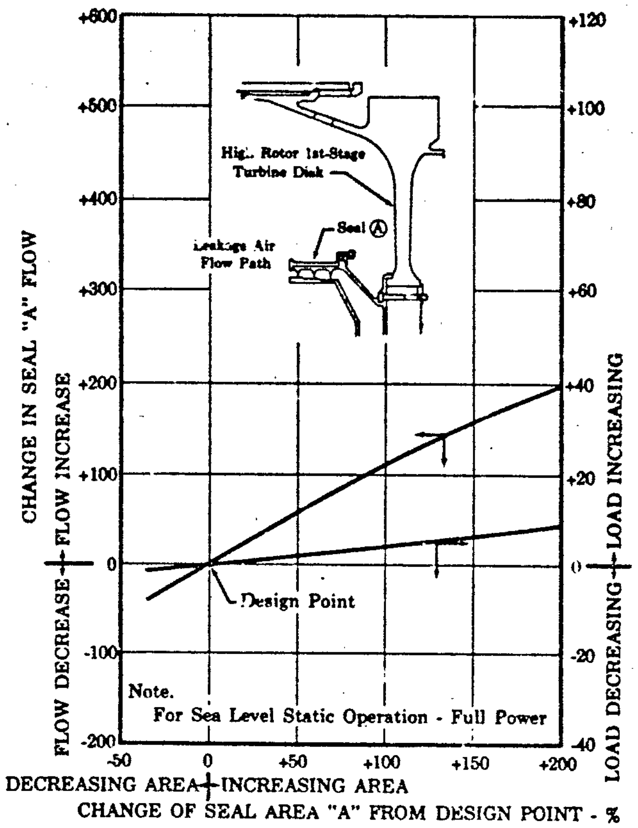

Figure 10. Results of Typical 1st-Stage Turbine Labyrinth Seal Flow and High Rotor Bearing Load Sensitivity Study

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(2) Thrust Balance

The computer program mentioned above was used to design a system so that the predominant direction of force on the thrust bearings for the expected engine operating envelope would be forward on the low rotor and rearward on the high rotor.

The forward loading on the low rotor results from the fan blade forward aerodynamic load. This load can be reduced to an acceptable level, but it is too large to be reversed by the available pressure balance forces.

The rearward-loaded high rotor is primarily due to the 1st-stage turbine design requirements. The pressure on the forward side of the 1st-stage disk (cavity Ⓐ, figure 9) must be higher than the main stream turbine inlet pressure to (1) prevent "hot" turbine gases from mixing with the cooling air system, and (2) supply the cooling airflow required in the 1st-stage turbine blade. Furthermore, the high turbine performance requirements necessitated a low leakage seal between the first and second turbine stages. To satisfy this requirement, the seal was located at the minimum possible diameter to obtain a minimum flow area at the labyrinth (figure 9). This configuration exposes the rearward side (cavity Ⓑ) of the 1st-stage turbine disk to a

low pressure (as compared to the front side). This condition results in a predominant rearward net force on the rotor. To offset the forward thrust of the high rotor, cavity C (aft of the 8th-stage disk) is vented through the high compressor diffuser struts to cavity D (between the gas generator and the inner fan duct liners). At all flight conditions, the pressure in cavity C is essentially equal to the pressure in cavity E in front of the high compressor. Thus, this configuration gives a balanced-piston effect on the high compressor rotor.

The final thrust bearing load map (bearing thrust vs Mach No. and altitude) for the high and low rotor is given in figures 11 and 12, respectively. No reversal in the load direction is expected with the present bearing system.

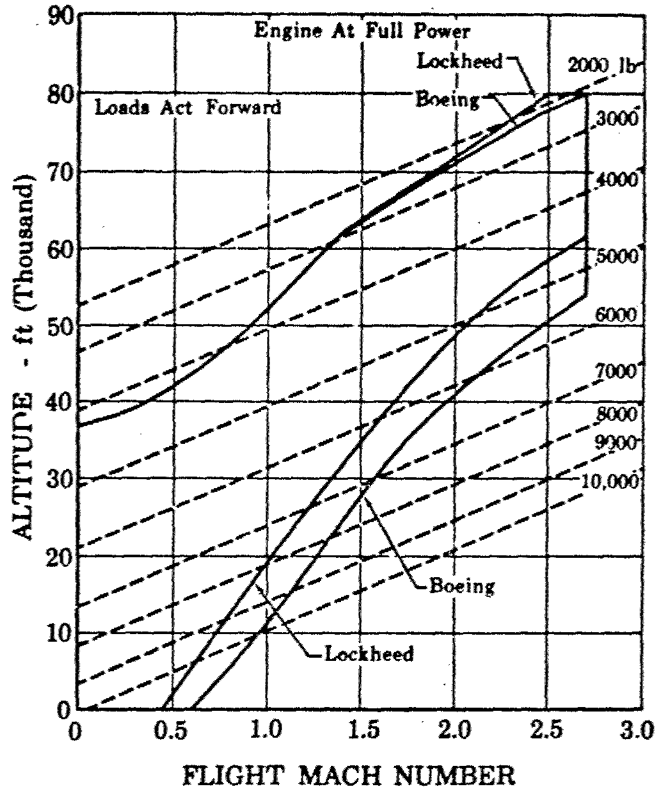

Figure 11. No. 1 Thrust Bearing Load Map FD 16375 BIIH

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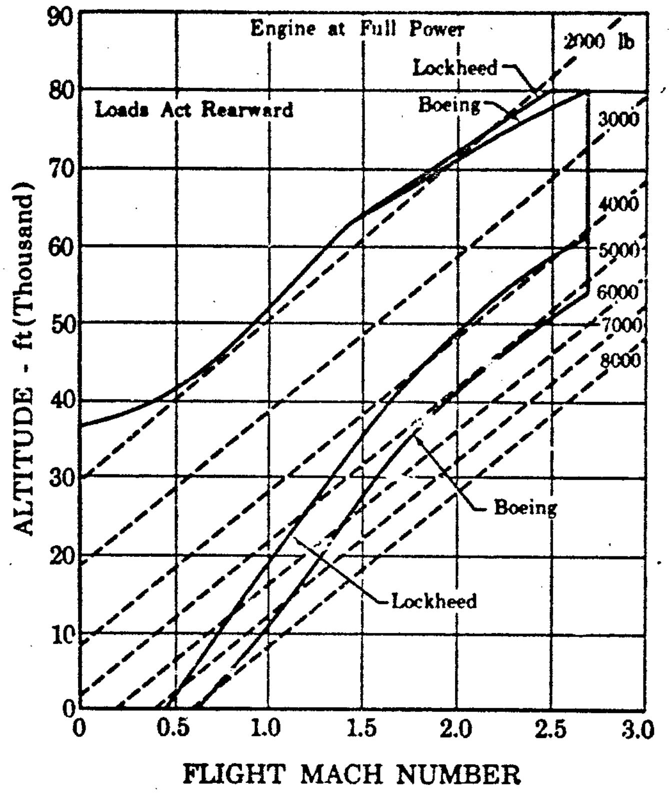

Figure 12. No. 2 Thrust Bearing Load Map FD 16376
BIIH

Testing of the J58 has verified the bearing load calculation procedure described above. Two methods of verifying rotor thrust were used in the J58. In the first method, measured chamber pressures and temperatures were used to verify calculated values. The second method uses a calibrated thrust ring built into the bearing stackup. This ring has a direct readout of axial bearing loads. In addition to development testing, the J58 engine was also equipped to verify the bearing load on each production engine by recording critical engine pressures. This same system has been incorporated in the JTF17.

(3) Cooling Flow System

The cooling system (figure 9) has three major sources of cooling air:

1. Air from 4th-stage compressor stator is used for cooling the high compressor bores

2. High compressor discharge air is used for cooling the turbine

3. Fan discharge air is used for cooling the fan disks.

These three cooling flow sources act independently, and provide the pressure-temperature levels needed to meet internal engine cooling requirements. These flow sources for the respective engine component, fan, high compressor, and turbine, are as follows.

(a) Fan Cooling

The cooling air is supplied from the fan discharge. This flow is required to prevent large temperature gradients in the fan disks.

(b) Compressor Cooling

Compressor disk cooling flow is supplied from the 4th-stage compressor stator root. The flow enters a series of radial flow tubes located between the 4th- and 5th-stage disks. These tubes create a "reverse pump" action (forced vortex) that is required to establish a compressor bore pressure high enough to satisfy the disk cooling requirements. If a free vortex were permitted to exit at this point, a lower static pressure would result, requiring tap-off from a higher stage, with resulting higher temperature air and poorer cooling.

After leaving the radial tubes, cooling flow is forced over the 5th- through 8th-stage disks by an annular bore tube. The flow then enters the inner region of the bore tube and flows forward through the high rotor hub into an annular passage through the intermediate case, and through struts to the fan cavity region. This flow path is required to assure minimum outflow forward of the 3rd-stage compressor. The bore tube leakage flow into cavity E (figure 9) is controlled by a four-lip labyrinth seal located at a small radius.

(c) Turbine Cooling

Proper turbine cooling requires air at the maximum available engine pressure. This is required for optimum cooling of the 1st-stage turbine blade as explained in paragraph C of this section (Turbine).

The turbine-cooling air is supplied through orifices in the inner burner case to a chamber forward of the 1st-stage turbine disk. From this chamber, air is directed to the disk rim for cooling the 1st-stage turbine blades, and through the high rotor hub orifices for cooling the 2nd-stage blades and low rotor disks. A small amount of the air from this chamber leaks across a three-lip labyrinth seal. This leakage air flows into the main gas stream forward of the 1st-stage blades. Minimum leakage through these seals is achieved by maintaining small seal flow areas. These areas are held to a minimum through the design of "thermally responsive" seals and placing the seal at the minimum diameter possible. A seal is said to be "thermally responsive" when the knife edges and lands are located far enough away from their supporting members (disks) so that the radial thermal growth is only a function of local air temperature and is not influenced by the attachment growth.

The bore tube arrangement for the high compressor provides a unique benefit in that the compressor cooling system is completely independent of any air source other than the 4th-stage stator air, and can be increased or decreased, if required, to obtain lower or higher airflows without affecting the engine thrust balance system or the cooling flows of other engine components.

The stationary parts of the turbine (the 1st-, 2nd-, and 3rd-stage vanes) are cooled by ducting high compressor discharge air past the outside of the burner section. This air is metered to each vane at the turbine case. The turbine section of this report explains this system in detail.

5. Windmilling Thrust Balance Load Analysis

a. Description

A windmill brake located in front of the high compressor is incorporated into the engine design. A detailed description of this brake and its operation is presented in paragraph A of this section. Briefly stated, however, its function is to reduce the engine rpm during windmilling.

A thorough analysis was made of the thrust balance loads both with and without the windmilling brake and these results are presented below.

b. Results of Load Analysis

The engine thrust balance analysis for windmill operation, both with and without the brake, indicates that the thrust load on the bearings will approach zero. This means that bearing skidding can be expected to occur. However, based on our experience with the J58, this skidding at 2000 rpm (the maximum speed expected for either rotor with the brake applied) will not be detrimental to the bearings. Damaged bearings would be expected if skidding occurred at nonbraked windmill conditions

6. Vibration Characteristics

a. Description

Careful attention has been given to engine vibration resulting from the interaction of the rotating and stationary structure since this is one of the primary causes of premature engine removal. Problems that result from excessive engine vibration include: (1) metal fatigue, (2) wear of mating surfaces, (3) contact between rotating and stationary engine parts, and (4) loosening or separation of mechanical joints. Aircraft structural vibration may also be excited by excessively rough engine operation.

b. Design Objective

The design objective of the engine vibration analysis is to ensure low shaft bending excitation and a minimum amplitude for engine case vibration. Furthermore, particular attention is given to individual part design to minimize expensive and time consuming rework due to excessive vibration at initial new engine assembly, and at initial assembly after overhaul.

c. Design Requirements

The attainment of optimum engine vibration characteristics requires careful consideration of the desired engine performance, assembly methods, and overall engine size. Major factors desired are an adequate rotor

critical speed margin and an engine having minimum weight. This must be achieved with various restrictions on the design such as those listed below:

1. Low rotor shaft must have a diameter which will pass through the No. 3 bearing compartment to permit removal of the low rotor turbine assembly with minimum engine teardown
2. Hole diameter through which the low rotor shaft must pass is limited by No. 3 bearing design requirements (refer to paragraph G of this section, Seals and Bearings)
3. Shafts must be capable of withstanding 10% blade loss from any single compressor or turbine stage.

d. Design Criteria

Operational vibration limits set for the JTF17 when airframe mounted are:

1. Single amplitude excitation should not exceed 1.5 mils on the intermediate and duct heater case
2. Single amplitude excitation should not exceed 2 mils on the fan inlet
3. Each shaft rigid bearing critical speed should be at least 20% above the maximum design speed.

e. Design Approach

(1) General

The design analysis of engine vibration is an evaluation of the severity of vibration that will be encountered at various engine operating conditions accounting for the natural modes of vibration and the excitation that will result from residual unbalance. Engine vibration amplitude is usually maximum at a critical speed where an engine rotor turns at a speed that is equal to any of the family of engine natural frequencies. If the rotor is imperfectly balanced, it is bowed by a load that rotates with the rotor unbalance. Engine vibration is excited by the centrifugal load acting on the bowed rotor. The nonrotating engine structure and nonresonant rotor in a twin-spool engine are subject to vibratory loads, while the resonant rotor is subjected to simple bending related to initial unbalance. The amplitude of engine vibration and rotor bow are limited by rotor stiffness and the damping provided by the friction forces acting to resist vibratory motion of mechanically joined parts of the engine static structure.

The 20% margin required on rigid bearing critical speed provides shaft stiffness that is insensitive to residual rotor unbalance because amplification of shaft bending is kept to a minimum. Further advantages of adhering to a 20% margin are that rotors are stable under unsymmetric pressure loading. Circumferential variations in radial pressure on labyrinth seals or on rotor stages, caused by slight out-of-roundness of lands and

shrouds, cannot then cause unstable rotor whirl. Axial pressure fluctuations over part of the flow annulus, in the event of surge, produce negligible shaft deflection.

(2) Rotors

In the JTF17, the high-speed and low-speed rotors are the exciting mechanisms that cause rotary loads on the bearings. Each rotor is supported by two bearings, which eliminates the inherent alignment difficulties associated with three-bearing supported rotors. This alignment has been a primary factor in excessive unbalance and premature engine removal in past engines. The two bearing system eliminates rotating forces produced by a misaligned three bearing system.

The short engine design, coupled with the gyroscopic stiffening inherent in the large diameter fan, results in a lightweight and stiff low-speed rotor. Similarly, the high-speed rotor shaft is light and stiff because of the short distance between bearings and the gyroscopic stiffening resulting from the overhung turbine.

The advantage of a stiff rotor is that it is relatively insensitive to residual rotor unbalance. Minimal rotor excitation results because of small rotor bending. Therefore, the total unbalance is primarily a function of initial unbalance, which can be controlled by close machining tolerances and balancing of the rotor disks, shaft, and rotor assembly.

The stiff rotor concepts utilized in the JTF17 engine design have been demonstrated on advanced development engines, such as the JTF14C, JTF14F, and the JTF17 initial test engines. In addition, the stiff rotor concept is employed in the compressors for P&WA JT3, JT3D and JT4 commercial engines. These engines have proved this concept through millions of hours of service operation. All parts that comprise the primary rotor structure are provided with positive radial pilots and torsional restraints, and are held together by a number of uniformly loaded tie rods that ensure adequate clamping action. Critical joints in the rotor assemblies are seated (with the multi-tie rod construction) prior to balancing. Proper seating and the clamping action provided by the tie rods ensures that there is no shift in the relative position of adjacent parts during engine operation. Even with removal and replacement of the fan section, balance is maintained, because the fan section is dynamically balanced as a separate package.

(3) Bearings and Supports

The engine rotors are supported on three separate bearing support structures. The intermediate case supports both thrust bearings. The high-rotor rear bearing structure is supported from the primary gas generator diffuser case, and the low-rotor bearing is supported by struts in the turbine exhaust case. The bearing support structures transmit shaft loads to the primary gas generator cases.

The intermediate case support struts and primary gas generator aft support transmit loads to the outer case. The aft support provides additional stiffness to the primary gas generator case with the advantage of frictionally damping vibratory loads.

The bearing support structures are sized to adjust critical speeds away from speed ranges at which long time operation occurs. No critical speeds occur in the cruise ranges of the rotors. Critical speeds do exist within the operating range, but are of short duration and have low shaft bending and high damping from the cases. The No. 4 bearing support is an example of design that considers vibration. Figure 13 shows the No. 4 bearing support that has been designed with a slotted cylinder spring integral with the structure, that is relatively flexible, in order to reduce both shaft bending and the transmission of vibration.

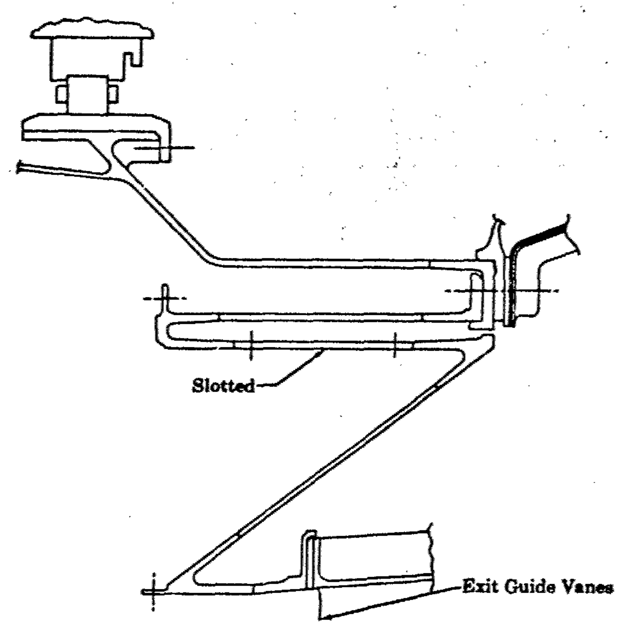

Figure 13. No. 4 Bearing Support FD 16388 BIIH

(4) Overall Engine Vibratory Mode Shapes

In the calculation of the critical speeds, a lumped parameter analysis is used that incorporates the Timoshenko differential equations for a rotating shaft together with the Prohl method of integration to derive sets of difference equations. The complete engine and mount structure, as shown schematically in figure 14 is accurately represented. Solution of the difference equations yields the natural frequencies (critical speeds) that are used to compute the normalized deflection curves (mode shapes).

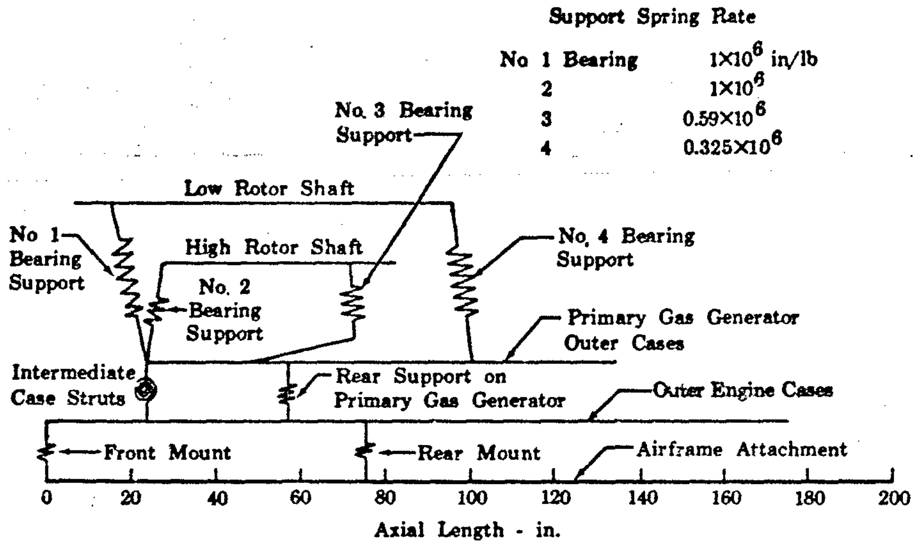

Figure 14. JTF17 Schematic Diagram for Critical Speed Analysis FD 16372 BIIH

The various modes of vibration for the engine are illustrated in figures 15 through 25. These modes consist of the rigid body modes (figures 15 through 17), the low-rotor excited modes (figures 18 through 21), and the high-rotor excited modes (figures 22 through 24). The rigid body modes can be excited by either the high-speed or low-speed rotor.

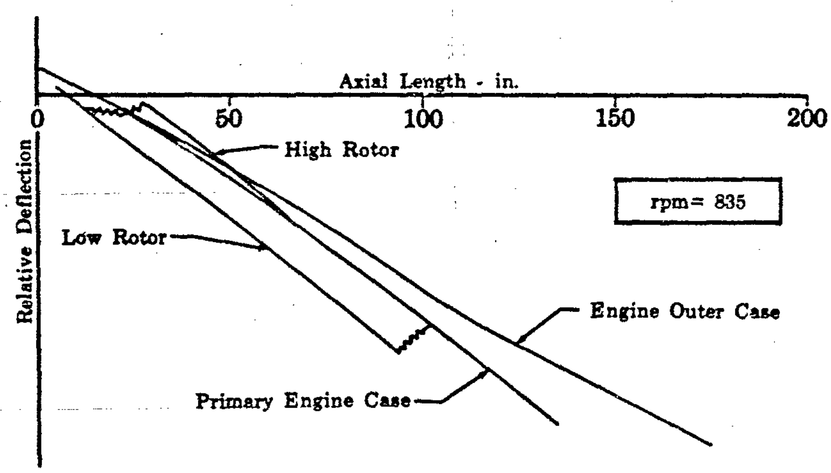

Figure 15. JTF17 Critical Speeds for Bouncing and Pitching of Engine on Stand Mount FD 16974 BIIH

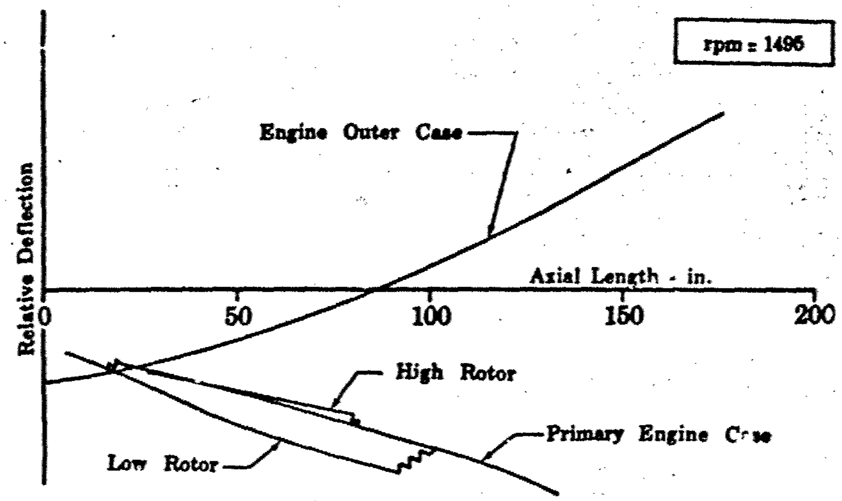

Figure 16. JTF17 Critical Speeds for Primary Gas Generator Pitching Out of Phase with Engine Outer Case FD 16975 BIIH

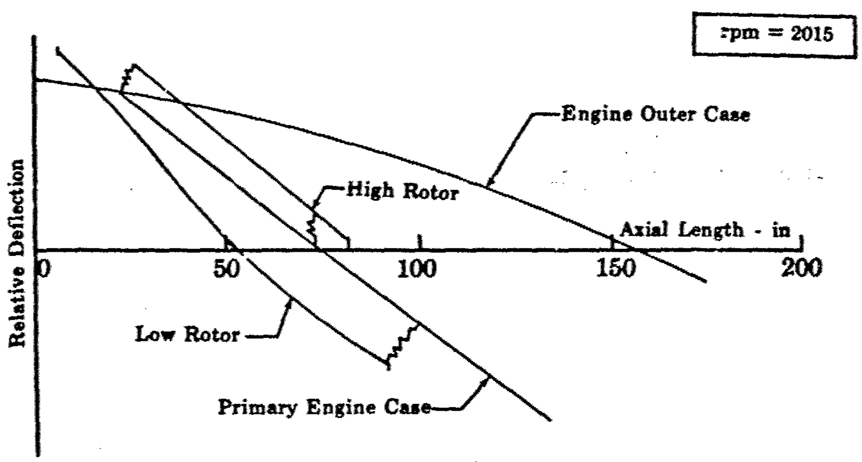

Figure 17. JTF17 Critical Speeds for Pitching Engine on Stand Mount FD 16976 BIIH

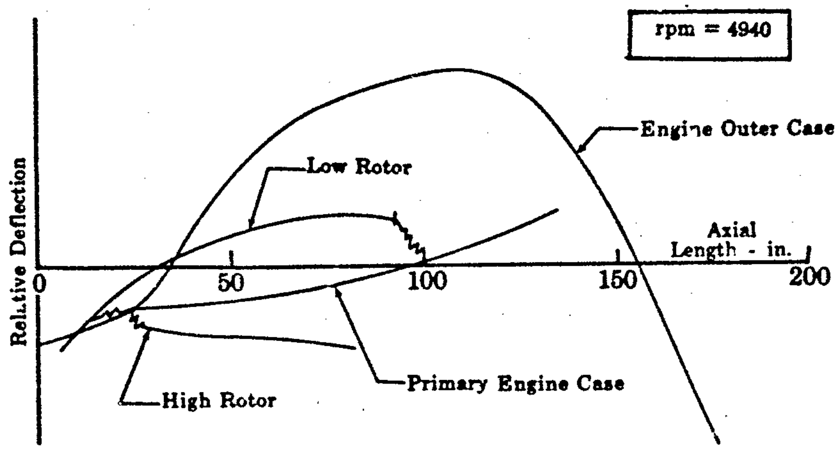

Figure 18. JTF17 Critical Speeds for First Bending of Engine Outer Case FD 16977 BIIH

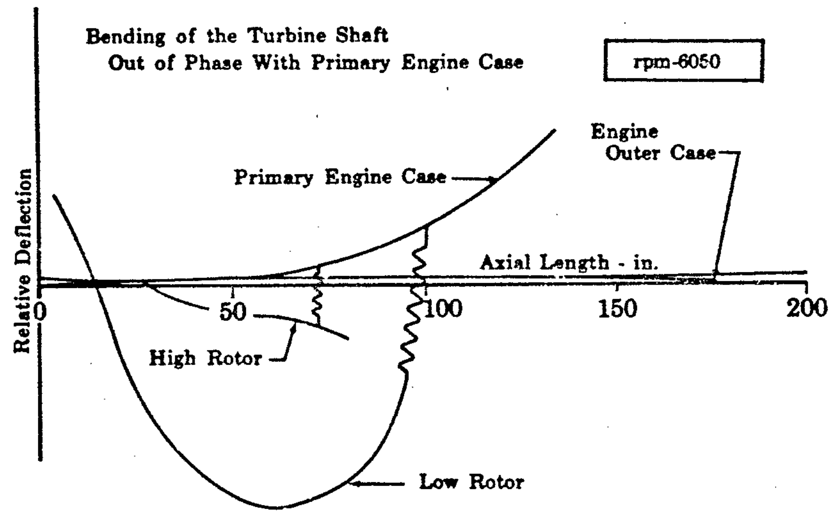

Figure 19. JTF17 Critical Speeds for First Bending of Low Rotor FD 16978 BIIH

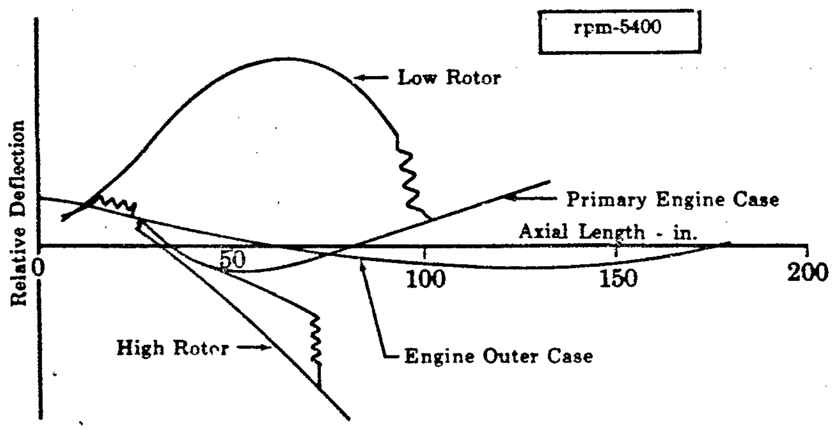

Figure 20. JTF17 Critical Speeds for Low Turbine Mode FD 16979 BIIH

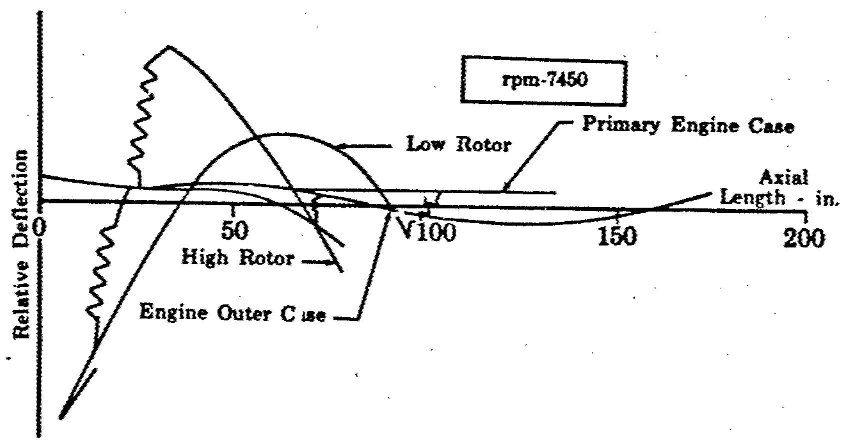

Figure 21. JTF17 Critical Speeds for Fan Mode FD 16980 BIIH

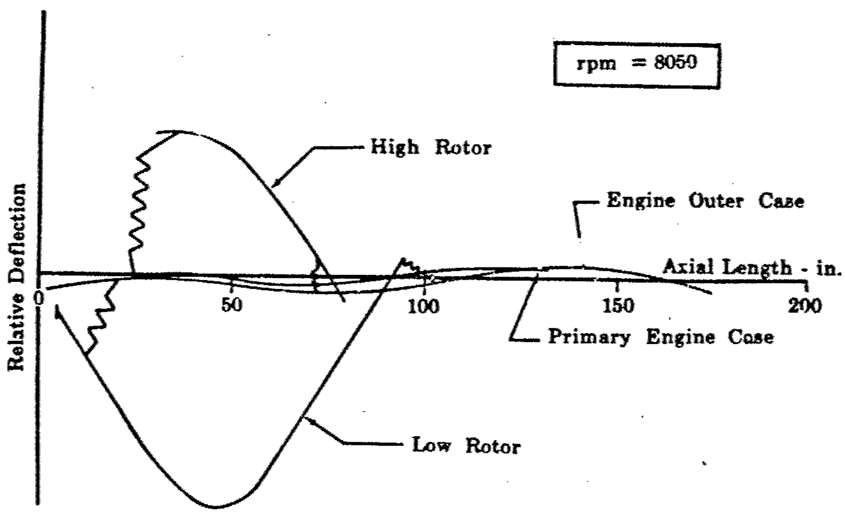

Figure 22. JTF17 Critical Speeds for High Compressor Mode FD 16981 BIIH

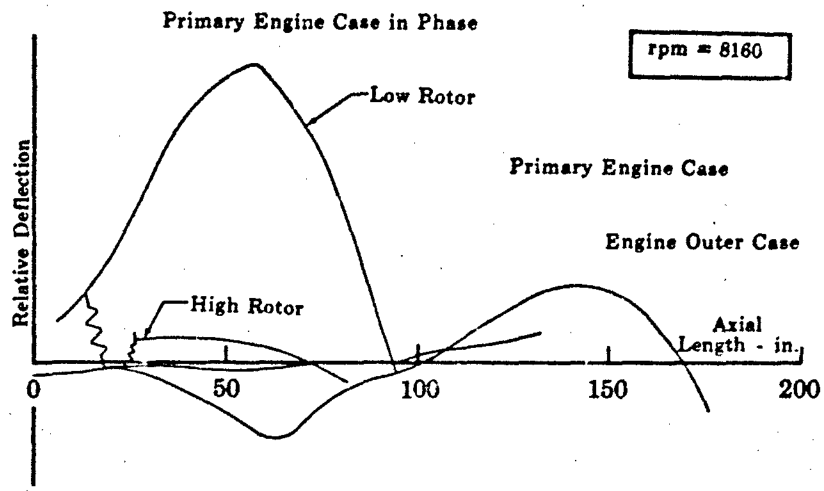

Figure 23. JTF17 Critical Speeds for Second Bending of Engine Outer Case FD 16982 BIIH

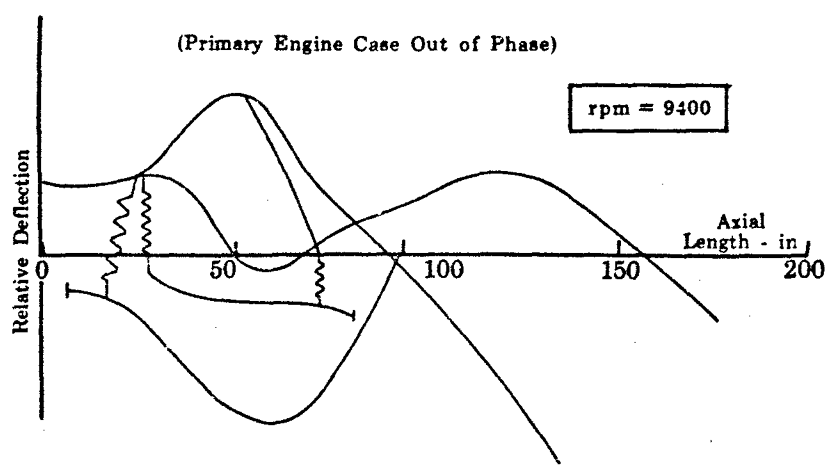

Figure 24. JTF17 Critical Speeds for Second Bending of Engine Outer Case FD 16983 BIIH

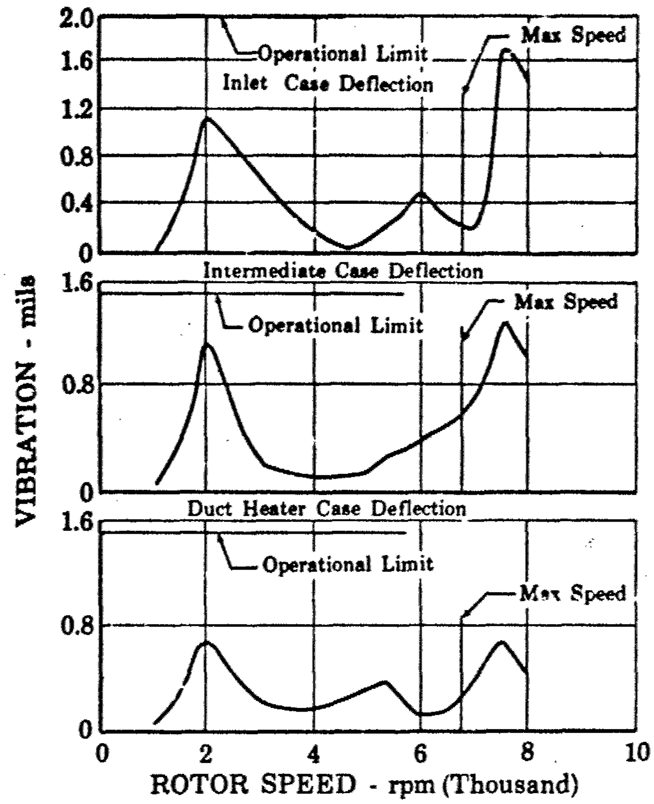

Figure 25. Engine Response to Maximum Fan Unbalance FD 16984 BIIH

The rigid body modes are characterized by little or no bending of any part of the engine rotor and frame. The first mode is pitching and bouncing of the engine on its support structure. The second mode consists of the primary gas generator moving out-of-phase with the main engine case. The third mode is pitching of the engine on the engine mount system. It is doubtful that these modes could be excited strongly enough to be identifiable unless very large unbalances are added to the rotors. These mount modes are never troublesome because they are encountered at low engine operating speeds. The fourth mode (4940 rpm) consists of first bending of the entire engine case and, therefore, is not affected by rotor changes. The large amount of case damping in this mode effectively suppresses any response in this mode.

The fifth mode (5400 rpm) consists of low rotor bending and turbine bounce, and is placed below the cruise range of the low rotor. The sixth mode is placed above cruise at 6050 rpm. In this mode the turbine is moving out-of-phase with the primary gas generator case. The No. 4 bearing support stiffness is the prime factor in determining the speeds and amount of rotor bending in the fifth and sixth modes. The support stiffness of the No. 4 bearing support was established at 325,000 pounds per inch, which keeps rotor bending well within acceptable levels.

The seventh mode (fan mode) occurs at 7450 rpm. This mode will not be excited because it is above the low-rotor range. The high compressor

mode (eight) at 8050 rpm is placed below the cruise range of the high rotor (8200 to 8700 rpm) and above loiter rpm of 7300. It is characterized by bending of the high compressor and pitching about the No. 3 bearing.

The ninth and tenth modes are second engine case bending. They differ in that the ninth has the engine external structure and the gas generator cases in phase, whereas in the tenth, the inner and outer engine cases are out-of-phase. Negligible excitation is expected in these modes because the high rotor has small bending content relative to the cases. Also, the case damping is large for these modes.

Vibratory response of the engine cases to the maximum possible fan unbalance, calculated using the worst assembly tolerance stackup, is shown in figure 25. These vibration amplitudes are calculated by means of a modal analysis. Contribution of each natural mode of vibration is calculated using damping coefficients set to equal amplifications at resonance, ranging from 5 for well-damped modes to 2? for poorly damped modes. The resultant amplitudes of the engine cases are much lower than the allowable limits which are given in the design criteria of this section. It can be concluded that the JTF17 has very low sensitivity to unbalance.

7. Field Inspection and Repair Capability

The maintainability plan, Volume IV, Report F, Section I, of this proposal, discusses in detail the inspection and repair features of the JTF17 engine. Adequate external access to frequently inspected items and modular construction for unit disassembly and replacement are provided. The size of each module was determined by handling requirements (weight and size) and the most convenient way of exposing those parts that are most likely to need repair.

a. Inspection Provisions

Features are incorporated into the design to allow detection or evaluation of damaged parts without engine disassembly, and also to allow access for borescope and radioisotope inspection. Components such as filters, screens, and magnetic plugs are given special attention, since they need to be readily accessible. These features are available to maintenance personnel with the engine installed in the aircraft, and are located as follows.

(1) Filters

1. Main fuel pump inlet line
2. Fuel control inlet line
3. Hydraulic discharge line
4. Oil tank.

(2) Screens

Located in the fuel pump interstage bypass.

(3) Magnetic Plugs

Provisions have been made for installation by the airline companies, if they desire, of magnetic chip detector plugs in the following locations:

1. Main oil filter
2. No. 1 and 2 bearing compartment sumps
3. Main gearbox
4. Oil pump gearbox
5. Accessory drive overboard drain
6. Oil tank.

(4) Borescope Inspection

1. Access is provided at each stator stage of the high compressor at the bottom centerline providing for inspection of leading and trailing edges of all high compressor blades

2. Each igniter hole of the primary burner is readily accessible to provide for inspection of the burner, fuel nozzles, and leading edges of 1st-stage turbine vanes

3. Borescope inspection is provided in front of the 1st- and 2nd-stage turbine vanes — at the bottom centerline — providing for inspection of the leading edges of these blades.

No special provisions were made for inspection of the blades and vanes of the fan stages or the last turbine stage, since these components are accessible through the engine inlet and exhaust openings. Binoculars, or physical entry into the inlet and exhaust ducting, will be required for detailed inspection of these parts.

The rotor can be turned to allow inspection of each blade in a stage from a single access point by a device in the main gearbox.

Mockups of both flexible and rigid borescopes were built to aid in evaluating viewing capability and removal and replacement of borescope access plugs in the high compressor and turbine cases. These studies resulted in a special tool that keeps the plugs captive during inspection and precludes the possibility of losing them inside the engine. A complete description of these studies is given in Volume IV, Report F, Section I (Maintainability), of this proposal.

(5) Radioisotope Inspection

The low rotor shaft provides access for the radiographic equipment. This feature allows the inspection of any suspected problem areas in the internal components of the engine.

Radioisotope inspection of the engine is expected to be a primary factor, along with borescope and in-flight recording systems, permitting the elimination of the current TBO system of engine overhaul. Toward this end, P&WA has purchased radioisotope equipment and is currently

developing an inspection standard which can be employed by the airlines. To accelerate this task, P&WA personnel have studied the radiographic facilities that are currently in operation at Wright Air Development Center, Eastern Airlines, and the United Aircraft Corporation Research Laboratories. The UAC Research Laboratories have been active in developing the improved techniques currently employed on several P&WA commercial engines.

Techniques developed to date have been employed during Phase II-C to evaluate the condition of the compressor, turbine, and burner after each engine operation. On one occasion, the reason for an engine malfunction was completely diagnosed by this inspection. Normally, a complete engine teardown would have been required to define the exact problem. Furthermore, these inspections were conducted without the removal of the engine from the test stand.

b. Repair Capabilities

The engine design has been tailored to provide access, with a minimum of teardown, to those parts that are most likely to need maintenance. Airline records on all existing P&WA engines were reviewed to determine the areas that needed the most maintenance time in the past. These studies were coordinated with several airlines in an effort to ensure that all repairs were consistent with their facilities and anticipated turnaround times. Wherever practical, repairs are completed with the engine installed on the airframe. Unitized component construction facilitates repairs. A brief description of the SST engine repair features is given below. They are separated into two groups; repairs on an airframe-installed engine and repairs requiring engine removal from the airframe. Some of the repairs that are performed on the engine while it is installed in the airframe may require disassembly of airframe components.

(1) Engine Installed in Airframe

Fan Blades - Any number of blades in either the first or second stage can be removed and replaced. Moment-weighted blades are used to preserve rotor balance. Mechanical fixture attachment points are provided on 1st-stage blade shrouds to facilitate rotor and stator removal as a unit.

Fan Rotor Bolts - In the event of thread damage during assembly of the 1st- and 2nd-stage fan disks, any of these bolts can be replaced by removal of the fan stages only. This repair is accomplished without disturbing the No. 1 bearing compartment.

Fan Assembly - A special technique for assembling the fan outer case to the intermediate case permits removal of the entire fan without further engine disassembly.

Fan Stator Vanes - Separate mechanical attachment of vanes allows replacement of these parts.

Turbine Exhaust Probes, Harness, and Junction Boxes - The probes are designed for removal through the exhaust duct. The harness and the junction box used for the exhaust temperature measuring equipment are mounted on the exhaust duct nozzle and can be replaced after removal of the inner duct liner.

Primary Burner and Turbine - Past experience indicates that the burner, transition ducts, and 1st-stage turbine nozzles require easy access for inspection and repair. Access to these parts requires removal of the reverser-suppressor and the primary burner case. This case is segmented and can be removed without disassembling the turbine section. The transition ducts are also segmented to facilitate removal. The vanes are then exposed for inspection and replacement if required.

Start Bleed Valves, Primary Fuel Nozzles, and Manifold - These assemblies are accessible through eight access ports on the fan diffuser case. These ports expose the attach points on the gas generator. The fuel manifold is assembled in sections to permit replacement of any one section without removing the entire manifold.

Duct Heater Secondary Fuel Injectors, Duct Heater Secondary Combustion Turbulators, and Outer Duct Heater Liner - Replacement or repair of each of these units requires the removal of the reverser-suppressor and the portion of the duct heater aft of the rear engine mount. Each component is in segments and is individually replaceable with a minimum of maintenance time and cost.

Duct Heater - Removal of the reverser-suppressor and the portion of the duct heater aft of the rear engine mount is required for access to this assembly. Repairs can be accomplished by removing the heater from the engine. Replacement of the heater is facilitated by removing the two igniters and the eight burner retaining pins. This design eliminates the necessity of removing the Zone I fuel nozzles to permit burner replacement.

Duct Heater Fuel Nozzles - These nozzles and manifolds are accessible from the outside of the engine. Individual nozzle replacement is possible.

Primary Burner and Duct Heaters Igniters - These items will require a minimum of inspection and replacement time. The four spark igniters, two in the duct heater and two in the primary burner, are located on the outside of the bottom half of the engine. This feature provides the most convenient access for maintenance personnel.

Bearing Compartment Scavenge Pumps - The pumps for compartments 1, 2, and 3 are mounted on the outside of the engine and are completely accessible. The pump for compartment No. 4 is inside the engine, but is accessible through the engine tail pipe.

Engine Controls - All controls are mounted to permit removal of any one individual control without disturbing the remaining components. Quick-disconnect features are designed into control attachment points to reduce maintenance and replacement time. The plumbing to each control can be disconnected from the unit without removal of the plumbing from the engine. Controls weighing 45 pounds or more have handling provisions to assist in removal.

(2) Engine Removed From Airframe

Splitter Nose In Fan Section - This part is of riveted construction for easy replacement. The fan removal feature, previously discussed, makes this part accessible for repair.

Low Turbine Assembly - This assembly consisting of the 3rd-stage turbine disk, blades, and shaft, is balanced as a complete unit and can be removed and replaced as a complete unit. Present commercial engines, with the exception of the JT9D, require a stage-by-stage reassembly in the engine after prior balance of the rotating assembly as a separate unit. This latter procedure requires precise reassembly of the prebalanced unit to obtain an acceptable final balance.

All low turbine blades are moment-weighted to facilitate replacement without rebalancing the assembly.

Rotor Bearings No. 1, 2, and 4: Carbon Seals; Seal Plates; and N_2 Drive Gear - Several features were included in the design of these compartments to allow replacement of all critical parts. The bearing races are provided with puller grooves and bolted flanges. These features reduce the time required for replacement. The flanged races are retained with standard bolts and lock wire. This system eliminates the spanner nut retention previously used, and its attendant torquing requirements. The need for spanner nut wrenches is also eliminated. The design of the intermediate case now allows removal of the N_2 tower shaft drive gear train and the No. 2 bearing and seals. With the current configuration and the fan rotor removal feature, the No. 1 bearing and seal, the N_2 drive gear, and the No. 2 bearing and seal can be removed through the forward end of the intermediate case without further engine disassembly.

Modular assembly aids access to the No. 3 bearing. This access is provided by the removal of the duct heater assembly, inner duct heater liners, and the low and high turbine assemblies.

The No. 4 bearing and seal can be removed rearward through the exhaust end of the engine. The design incorporates features for removing the center body cone, oil inlet, and scavenge drain lines through the exhaust. This procedure minimizes the engine teardown required to provide access to this area.

Fixtures have been designed to support the rotor during bearing removal to prevent damaging the blade tips. If desired by the airlines, the repairs described above can be performed with the engine installed in the aircraft.

8. Features To Extend the Service Life of the Engine Structure

Wherever possible, those features that have been instrumental in extending TBO times in the current P&WA commerical, military, and industrial engines, have been incorporated in the JTF17 design.

These features include design concepts that utilize the advantages of the fan configuration, minimize damage to surfaces due to vibration, use replaceable sleeves and liners, employ surface peening and coatings, and use metallurgical improvements. A brief description of the use of these items in the JTF17 is given below.

a. Design Concepts

(1) Duct Heating Fan Configuration

Utilization of a heater in the fan duct for augmentation results in a lower operating temperature of the augmentation system parts as compared to an afterburning turbojet. Lower temperature reduces maximum metal temperature, thereby increasing engine life.

(2) Case-Supported Segmented Liner Construction

This configuration permits the use of a tension-membrane structure as illustrated in paragraph D of this Section (Duct Heater). It is used in the primary combustor and in the duct heater liners. The practical application of this type of structure was proven through several years of experimental liner investigations during the J58 engine development program. This new concept, which resists the pressure load by membrane forces, provided a lightweight design that has no known life limit.

(3) Minimizing Vibration Excitation

In general, engine components are designed to eliminate vibration excitations in the natural frequency range. Fan and turbine blade damping methods are based on configurations proven to be reliable in the J58, JT8D, and JT3D engines. Part-span shrouds, tip shrouds, and airfoil platform damper weights (friction type) are used where required to minimize vibration.

(4) Material Surface Conditioning

Past engine development programs have shown that surface conditioning is required in certain types of jet engine hardware. Compressor and turbine blade attachments, airfoils of the turbine blades and vanes, and friction damping surfaces are those generally requiring special treatment. Typical applications in the JTF17 are as follows:

1. Shot Peening - Compressor and turbine blade roots
2. Glass - Bead Peening

   a. Complete fan blades (airfoil and attachment)

   b. High compressor airfoils

3. Coatings

a. Turbine vanes and blades are coated for erosion protection

b. Compressor and turbine blade dampers are hard faced

c. High compressor blade attachments are lubricated (silver plate)

d. Fan blade attachments are lubricated (graphite varnish)

e. Bolt threads are usually lubricated by silver plating or applied lubricants to improve reusability

(5) Metallurgical Improvements

The metallurgical advances anticipated in the near future are being considered in predicting the growth potential and projected life for the JTF17. The recent development by P&WA of the single-crystal turbine (MONOCRYSTALOY$_{TM}$) blade will definitely contribute toward extending the service life of this engine. In testing to date, this material has shown excellent thermal shock properties. It has withstood 2400 thermal shock cycles, where the metal temperature is increased from 70°F to 2200°F at a rate exceeding that expected in normal operation, with no failures. This material is undergoing test evaluation in both the J58 and TF30 engines.

I. LINES AND FITTINGS

1. Description

The designs of the lines, fittings and associated hardware used in the JTF17 engine fuel, oil, hydraulic and air systems are based directly on J58 experience. The J58 engine is currently in service operation and has accumulated many hours of sustained flight operation at speeds above Mach 3. The operational environment is more severe than that which will be encountered in the SST aircraft. Maintainable fluid lines and hardware concepts were developed to operate satisfactorily in this environment during the J58 program. These concepts, which have been proven satisfactory by service experience, are used in the JTF17 design.

The JTF17 engine piping systems consist of fuel, hydraulic, lubrication, air systems and signal lines. These lines conduct fluids and pressures from one section of the engine to another. The arrangement of the lines and fittings is shown in figures 3, 4, 6, and 7 in Section I, Introduction. Stresses, clamping locations and routings are established by extensive calculations made by high speed digital computer programs which were developed for the J58 engine. Such programs are used for all fluid systems, and provide a quick and accurate means of routing tubes for the best compromise of short lengths, while maintaining sufficient flexibility to keep stresses low and uniform throughout the tube. The lines are designed with support brackets located to keep tube resonant frequencies well above any engine frequencies and with routes designed for easy access to inspection ports, ease of engine section removal and easy replacement of external components.

Integral tube connectors are used extensively to ensure maximum tubing system reliability. These connectors were developed during the J58 program and are machined as an integral part of the tube to provide maximum fatigue strength and minimum cost. Experience has shown this design to be superior to tubes with brazed fittings, and therefore brazed connections or ferrules are not used on the JTF17 engine. Problems associated with stress concentration and low fatigue characteristics of the braze fillet, and the inherent inability to assure complete and positive braze by any developed inspection methods make the brazed ferrule approach undesirable for this application.

Two types of connectors are used in the JTF17 engine. These are integral ferrule AN type threaded joints using a standard nut-tube coupling arrangement as shown in figure 1, and bolted flange joints using metallic face seals as shown in figure 2. These joints have been developed to a very high level of reliability and low incidence of leakage on the J58 engine. This reliability is due to the wide use of the integral connector design and to the excellent performance of the seals used. The threaded connector joint utilizes a thin nickel gasket between the ferrule and connector and the flange connectors incorporate Inconel X "K" seals, both of which have excellent thermal shock capabilities.

A major development program is in progress at FRDC to develop titanium tubing for possible future use on the JTF17 engine. The low values of modulus of elasticity and density will effect a substantial weight and space savings. The program includes the manufacture of integral upset end fittings, and fatigue testing at elevated temperatures and pressures to determine design values that will ensure reliability equal to currently used materials.

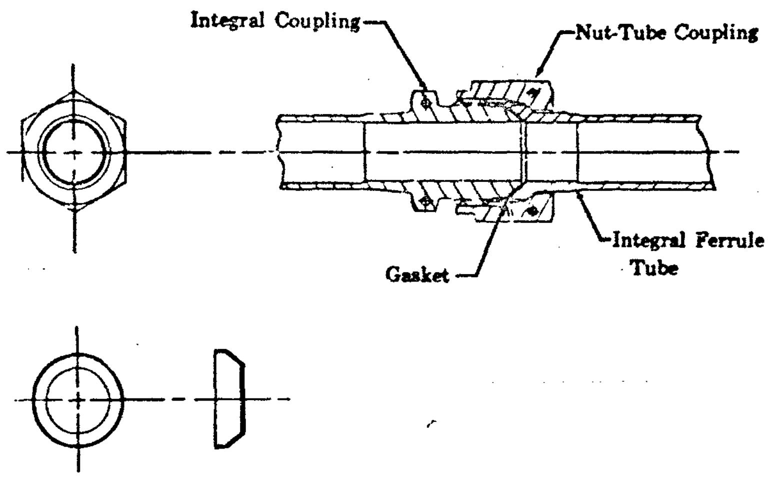

Figure 1. Integral Ferrule Joint With Sealing Gasket — FD 12336 BII

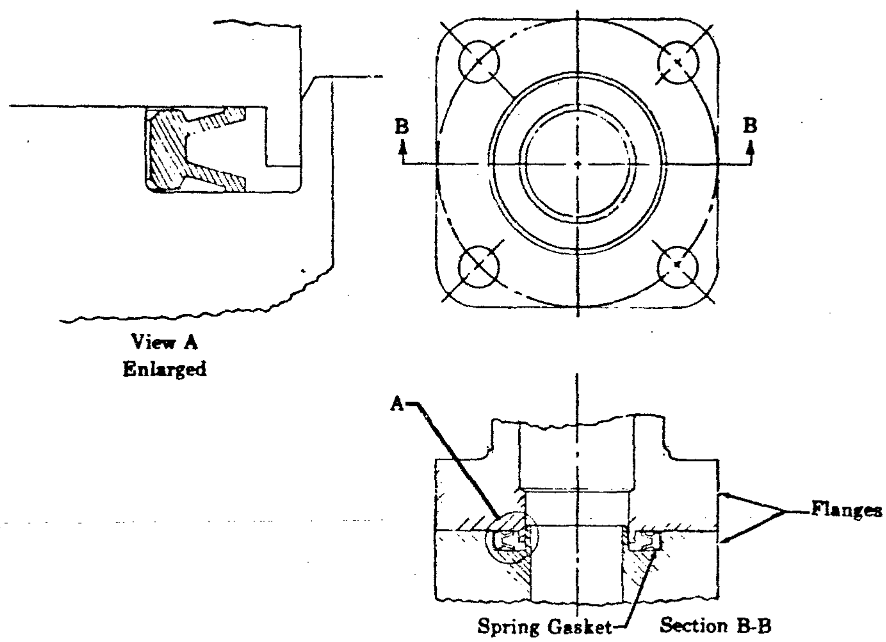

Figure 2. Metallic Static Face Seal For Joints Subjected to High Thermal Shock — FD 12338 BII

2. Objectives and Requirements

The design objectives for engine lines and fittings are to provide durable, low-stressed tubing to conduct fluids and pressures from one section of the engine to another without leakage. The tubes are routed to provide complete accessibility for removal of engine components and access covers, with zero

leakage connect points that permit optimum engine maintenance and foolproof assembly. The tubes are designed for easy maintenance and inexpensive fabrication and with adequate support to keep resonant frequencies well above engine operating frequencies.

Specific objectives and requirements are:

1. Each tube is designed for its specific application, i.e., function and environment, with a usable life that is consistent with 50,000 hours aircraft life and with 10,000 hours engine TBO without major repair.

2. Stresses associated with differential expansion and contraction of various parts of the engine during steady-state and transient operation must be kept within acceptable limits through proper routing and support arrangements.

3. Tubes are routed to ensure a nominal clearance envelope of 0.500 inch between engine components and other tubes. A clearance of 1.000 inch nominal is maintained adjacent to airframe structure.

4. Fuel and oil supply lines must function satisfactorily in ambient temperatures from -65°F to 700°F and fluid temperature to 400°F maximum.

5. Air lines, including breather and signal lines, and fluid lines that are drained during part of the flight envelope must withstand ambient temperatures of 1050°F maximum.

3. Design Approach

a. Detailed Description

(1) Tubing

Experience gained from the design of the high Mach number and high performance J58 engine is being directly applied to the JTF17 engine. Development engine time in excess of 22,000 hours has been accumulated on tubing of the type used on the JTF17. Tubing systems for the JTF17 engine are designed using refined tube computer programs which were developed in conjunction with the J58 engine project. Two major tube computer programs are utilized. These are: (1) the tube stress program, and (2) the tube clearance program.

The procedure outlined in figure 3 demonstrates the approach used to design engine tubing systems. Each tubing system is analyzed to determine optimum tube size, wall thickness and tentative routes.

Pump inlet line sizes are designed with a fluid velocity of 6 to 10 ft/sec to prevent cavitation or erosion. Tube sizes for the duct heater fuel system are derived from a maximum allowable fill time requirement with resulting velocities and pressure drops in the system compatible with a pressure requirement at the nozzles. Other systems are

de..g mit minimum tube size within allowable pressure loss requi... and minimum expansion loop length.

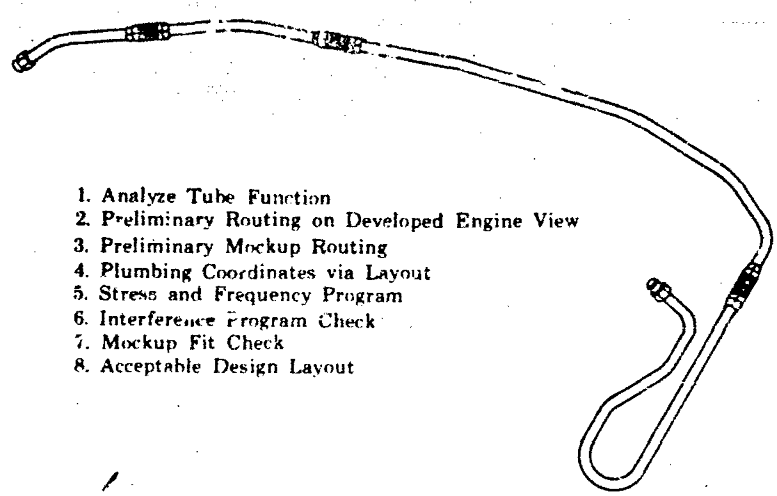

Figure 3. P&WA Tube Design Process

FD 17001
BII

Tube wall thicknesses that have been established for the JTF17 engine are as follows:

Tube Size (Outside Dia, in.)	Maximum Operating Pressure, psi	Basic Wall Thickness, in.
Up to 2.500	Under 1000	0.035
Up to 0.438	Over 1000	0.035
0.500 to 0.688	Over 1000	0.049
0.750 to 1.500	Over 1000	0.065

Tubing routes are checked on engine mockups to provide a visual 3-dimensional aid for final route selection. Removal of components, ease of engine maintenance with respect to inspection parts, access panels and ports are considered during this phase of tubing design.

Tube OD wall thicknesses, coordinates, end points, bracket locations and final thermal calculations are input into a tubing computer program which computes stresses along the tube length, relative thermal movements at sliding bracket locations, and resonant frequencies between support points. The tube is redesigned and recomputed, in an iterative process, until the design requirements of stresses and frequencies are met The maximum combined hoop, tension and bending design stress is limited to 22,000 psi, providing ample stress margin for tolerances and fatigue stresses for the material used. Tubing resonant frequencies are kept above 180 cycles per second, which is 20% above high rotor frequency.

Tube coordinates are input into the tube clearance program. This computer program verifies final tube clearances with respect to engine cases, components and other tubes. The tube clearance program was used extensively for checking compatibility of advance system designs with existing tubing and components during the J58 engine development.

PWA 770 (type 347) material is used for all lines operating at temperatures less than 800°F. AMS 2666 silver braze is used at stand-off locations to attach standoff collars to tube. Tubes routed in environments over 800°F, necessitating high temperature PWA 19 gold nickel braze at standoff locations, are fabricated from PWA 1060 (Inconel) material. These tubing materials were selected because of their "forgiveness" factor, having a fatigue strength considerably greater than yield strengths. Both PWA 770 (type 347) and PWA 1060 (Inconel) tubing have a 0.2% allowable yield stress of 30,000 psi and fatigue stress allowable of 36,000 psi at room temperature.

Fittings, adapters and tube nuts are made from AMS 5646 (type 347) material. Tubing nuts used in environments under 800°F are silver plated, those above 800°F are coated with molybdenum disulphide to prevent seizing.

(2) Air Ducts

Air duct lines include the duct heater fuel turbopump air supply, cabin bleed air ducts, and labyrinth seal vents. These designs are based on J58 practice, and utilize a vibration damping laminated bellows expansion joint to compensate for engine thermal growths, and have bolted flanges. Ducts are fabricated from AMS 5536 (Hastelloy X) material.

(3) Integral Connectors

Initial J58 development engine tubing utilized completely brazed "on engine" connections. Installation accessibility, braze quality, and maintenance procedures, inherent with the design, proved to be completely unacceptable and this design was discontinued. Mechanical connectors, brazed to tube ends replaced the brazed joint to permit quick replacement and servicing of components. Variations in the fatigue strength of brazed ferrules attributed to braze fit, fillet size, braze coverage, and brazing temperature were encountered with this configuration. Because of this the integral connector design shown in figure 1 was developed to replace the brazed ferrule, and brazed joints are not used on the JTF17 engine. The evolution of the integral connector is shown pictorially in figure 4. The integral connector is produced by [illegible] upsetting the parent tube material into a head from which the connector configuration is machined. This design has proven to be very satisfactory on the J58 engine, and is used extensively in the JTF17 design.

The approximate fatigue strengths of comparable size integral ferrules and brazed ferrules at test conditions of 450°F and 4500 psi internal pressures are as follows:

Sizes, (in.) Tube and Ferrule	Approximate Fatigue Strength (psi) Silver-Brazed Ferrule	Integral Ferrule
0.375	22,500	32,500
0.500	20,000 to 22,500	30,000
0.750	20,000 to 22,500	25,000

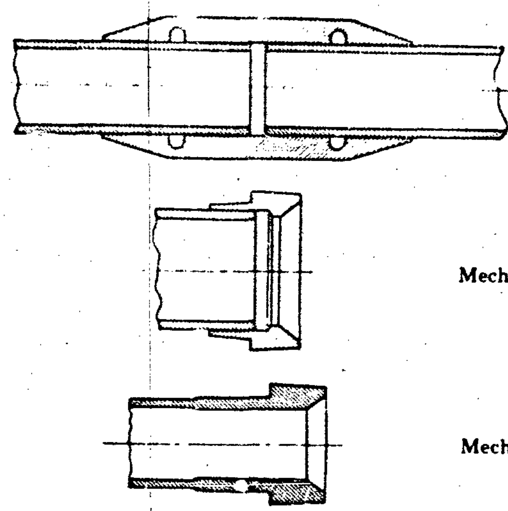

Figure 4. Evolution of Integral Connector FD 16618

The fatigue strength of the integral ferrule is affected only by the physical configuration of the machined ferrule and the parent tube material. Although the fatigue strength of the integral ferrule is only slightly higher than that of the best brazed joints, the integral ferrule has high reliability due to the strict quality control that can be exercised over the only variables, upsetting and machining. The engine experience with mechanical connectors is as follows:

Engine Experience	Hours
Total time	22,476
Total time, integral end fittings	5,208
Total time above 400°F, T_{t2}	5,870
Total time above 400°F, T_{t2}, integral	2,147
Total hot fuel time above 200°F	3,297
Total hot fuel time above 200°F, integral	1,353

Nickel gaskets, compressed between the ferrule and male connector, are used at all threaded connector joints. These gaskets, as shown in figure 1, were developed in the J58 program and have proven highly successful. These are used extensively in the JTF17 engine.

(4) Bolted Flanges

Bolted flanges with face seals are used on large lines (above 1.25 inches OD) where the size and weight of the AN type threaded connectors would become excessive. Spring type "K" seals are used with these flanges. Spring type "K" seals and bolted flanges are also used on lines which are subjected to

abrupt thermal changes such as the duct Heater Zone II fuel manifolds which are subjected to fluid flow after being operated dry in a hot environment when duct heater augmentation is initiated. This configuration was developed in the J58 program.

Engine components such as controls, actuators, etc., utilize threaded bosses and adapters at tubing connect points. Metal "wye" gaskets are used between the boss and adapter (see figure 5). The tube connection is made at the adapter, minimizing replacement cost if external threads are damaged during tube makeup or component handling. Initial component costs are also reduced by the elimination of male threads on the housings.

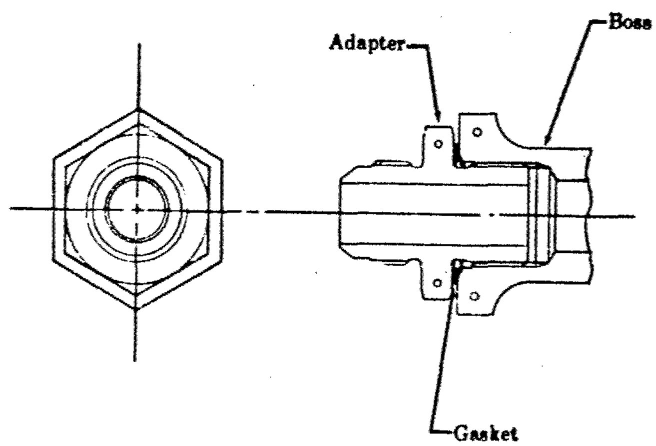

Figure 5. Boss-to-Adapter Joint

FD 12337
BII

"Last chance" filters will be used in a number of locations in both the fuel and oil systems in the JTF17 engine to protect local metering jets, and servos. The threaded adapters provide an excellent location for these filters and makes their installation simpler and lighter.

(5) Brackets

Sliding support brackets are used where movement must be allowed to account for thermal expansion. These brackets are designed to allow the tube to move in the direction that reduces the tube stress as the engine thermals change. Brackets are positioned in the most accessible locations with the minimum stress levels. The bushings for the bracket are assembled in the bracket slots with the retainers and are flared as shown in figure 6. This provides foolproof bracket assemblies for convenient installation and maintenance.

Static (fixed) brackets are used at support points that require no movement due to thermal growth. These brackets support the tubes and eliminate vibration. A combination of sliding and static brackets are used to shift the load from a critical area to a non-critical area of a given tube. Careful consideration of strength and natural vibration frequency is taken in designing both static and sliding brackets.

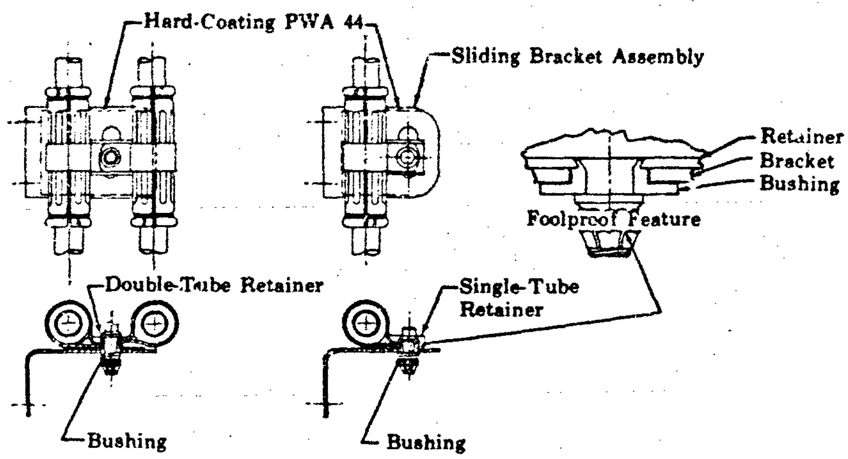

Figure 6. Sliding Bracket Assembly FD 17710 BII

The tubing stress program is used to calculate the tube stress at any predetermined number of intervals for a particular set of bracket points. The amount and direction of movement is given for each sliding bracket point. The amount of available movement at any bracket is input and the resultant tube stress calculated. The combination of sliding and static brackets is used to shift the stress from a critical area, i.e., connectors and component fittings and maintain low uniform stresses throughout the tube length.

Tube brackets are made of Inconel X (AMS 5542). When fully heat treated, these brackets have high strength up to 1500°F and good oxidation and corrosion resistance up to 1800°F. The retainers and bushings shown in figure 6 are made from cobalt base alloys and have good oxidation, thermal shock, and corrosion resistance up to 1800°F. The retainers are cast from Stellite 31 (AMS 5382) to reduce manufacturing cost and the bushings are machined from L-605 (AMS 5759). The areas of the brackets that are in contact with the sliding retainers and bushings are coated with diffused aluminum (PWA 44). Laboratory tests and extensive J58 experience have proved this combination of metals and coatings to have excellent wear life and strength properties at temperatures equal to or above those on the JTF17.

(b) Clamp [illegible]

Another feature used on the JTF17 engine and developed in conjunction with the J58 program is the tubing clamp standoff. (See figure 7.) This standoff eliminates clamp wear on the tube at the point where the clamp attaches the tube to a static or sliding bracket. The standoff consists of a three-piece, loose cover that is supported by small tee-shaped collars attached to the tube. This design eliminates highly stressed sections at tubing support points by permitting the tube to flex through the standoff instead of forming a rigid local tube section and dampens the tube vibration. The sleeve sections are lockwired for easy installation and disassembly. Clamps used at standoff locations are modified MS type clamps providing thicker cross section and close tolerance inside diameters.

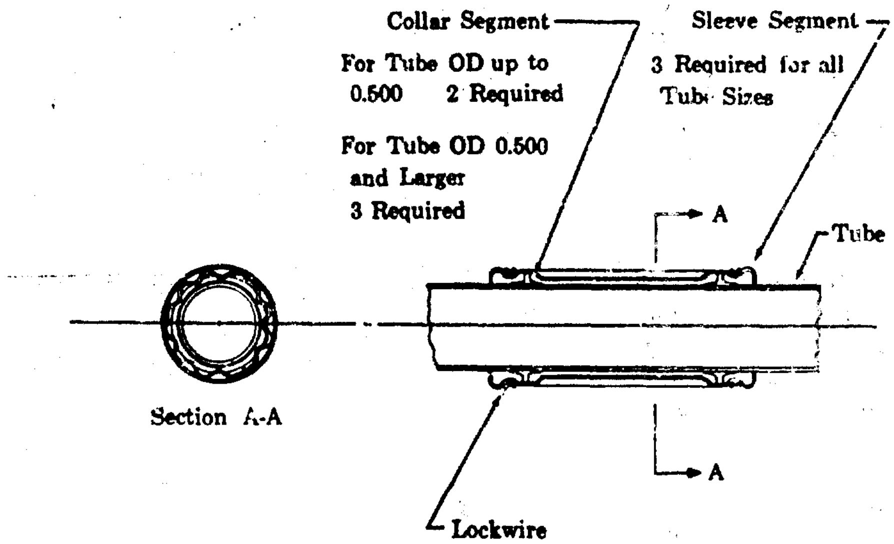

Figure 7. Tube Support Clamp Standoff FD 12341 BII

The sleeves and collars are made of L-605 (AMS 5537 and AMS 5759) to obtain the necessary strength and wear properties at elevated temperatures. The corrugated sleeve sections are manufactured by stamping, and are light, inexpensive, and very durable. The collars are brazed to stainless steel (PWA 770) tubes per AMS 2666 (silver braze), and high temperature Inconel (PWA 1060) tubes per PWA Specification 19 (gold-nickel braze).

(7) Rapid Replacement

Development testing of the J58 engine emphasized the need for simplified component removal. This feature has been incorporated into the JTF17 engine design and refined to the extent that the unitized control, with its 29 external ports, requires that only three tubing connections and one electrical connection be made to assemble the control on the engine. All other external connections are accomplished in the pedestal mount plate as described in the Control and Accessories Section of this proposal. (See Volume III, Report B, Section III.) Similarly, the hydraulic pump requires two tube connections and the main fuel pump requires but one tube connection plus the drive pad connections. Replacement time for these three components has been reduced to an estimated one-half hour.

Tubing connections are provided at engine flanges to facilitate engine section removal. Engine duct nozzle removal, for example, can be accomplished with eight-tube connectors located at the removal flange.

Systems containing two or more of the same type components such as nozzle actuators, fuel nozzles, etc., manifolded together, are designed with "flow-thru" fittings to minimize the number of joints required. The component then becomes the "tee" in the manifold and eliminates a jumper tube and two mechanical connections at each manifold on each component. (See figure 8.) The reduced number of connections and tubes increases engine reliability and constitutes a weight and cost savings.

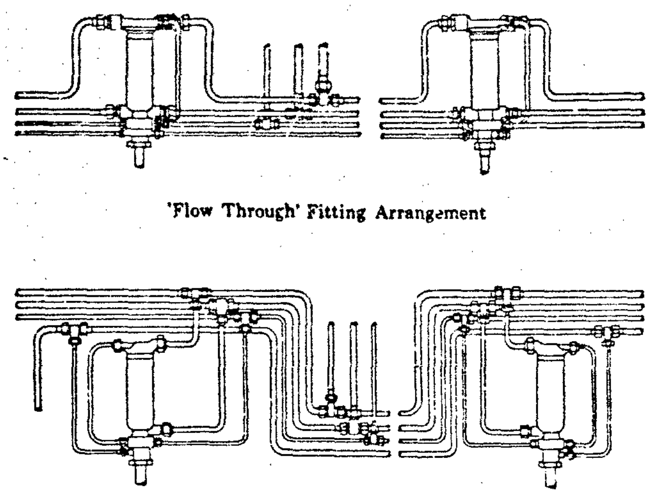

Figure 8. Comparison of "Flowthrough" vs Jumper Tube Piping FD 17711 BII

Main and Zone I duct heater fuel nozzle manifolds are designed with "flow-thru" fittings on the nozzle covers in clusters of five, six or seven covers. The design incorporates tubes with upset ends, similar to an integral ferrule, providing a thick cross section at a butt-welded joint. (See figure 9.)

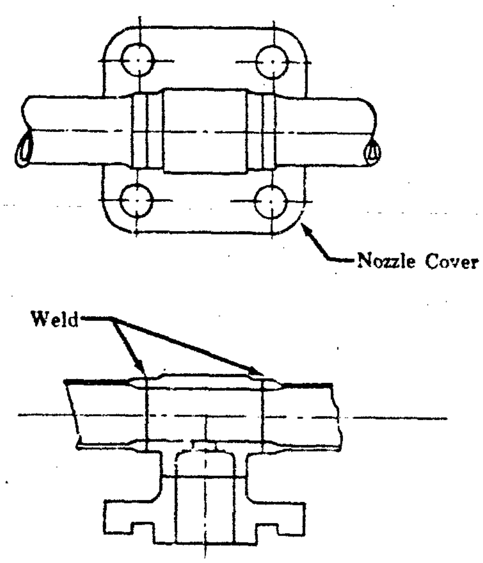

Figure 9. Welded "Flowthrough" Nozzle Covers FD 17712 BII

Fuel nozzle clusters, manifolded in this manner, are flow checked for uniform fuel distribution required for efficient burner operation. Line length, volume and number of mechanical connectors are kept to a minimum.

(8) Growth

Development programs in progress include titanium tubing and integral tube flanged connectors of both steel and titanium. Tests on titanium are being run to determine tubing stress levels. The low values of modulus of elasticity and density of titanium would effect a substantial weight and space savings without loss of tube integrity.

Development of the integral tube bolted flange configuration for larger size tubes and thermal shock applications will provide greater over-all engine reliability.

The design technology used on the JTF17 engine piping systems is based on more than 22,000 hours of J58 engine stand and flight testing at environmental temperatures and hydraulic pressures above those required for the JTF17 engine. As the design technology is advanced by continuing development and flight experience programs, these advances will be incorporated into the JTF17 tubing systems design.

b. Materials Summary

Part Name	Material	Spec.	Substantiation of Material Selection
Tubing - Wet Tubing - Dry	347 Inconel	PWA 770 PWA 1060	Fatigue Strength above yield strength properties and corrosion resistance
Tubing - Nut	347	5646	Corrosion resistance
Couplings & Tees	347	5646	Corrosion resistance
Brackets	Inconel X	AMS 5542	Strength
Bracket Retainers, Cast	Stellite 31	AMS 5382	Wear resistance and cost
Bracket Bushings	L-605	AMS 5759	Wear resistance
Standoff Collar Segment	L-605	AMS 5759	Wear resistance
Standoff Sleeve Segment	L-605	AMS 5537	Wear resistance
Connector Gasket	Nickel		Ductility
"K" Type Seal	Inconel X	AMS 5542	Strength
Wye Gasket	Inconel X	AMS 5542	Strength

c. Design Assurance Considerations

(1) Maintainability

1. Manifolded component tubing to reduce disconnect points for component replacement (i.e., quick disconnect components)
2. Tubing connectors located at engine flanges to facilitate engine section removal
3. Boss to tube adapters permit easy replacement of damaged external threads
4. Access and inspection port accessibility.

(2) Reliability

1. "Integral ferrule" tube connectors
2. "K" seals at thermal shock locations
3. Tubing computer program for low stress levels
4. Tube clamp standoff design lowers stress concentration.
5. Nickel gaskets at all tube connectors
6. Wear resistant sliding bracket parts
7. Wear resistant clamp standoff collars and covers.

(3) Safety

1. All threaded connectors are safety wired or lock nuts are used
2. See reliability items also.

(4) Value Engineering

1. Cast bracket retainers
2. Stamped clamp standoff sleeve sections
3. "Integral ferrule" tube connector
4. Boss to tube adapters to minimize thread replacement costs and initial component costs by eliminating male threads
5. Flow through fittings on components and bosses.

(5) Human Engineering

1. Non-interchangeability of tubes
2. Sliding brackets designed to prevent misassembly.

J. FLIGHT INSTRUMENTATION

1. Introduction and Objective

The JTF17 engine will be equipped with a well-developed, flight instrumentation system capable of providing accurate and reliable indications of pertinent engine parameters for airframe readout, ground checkout, and use in the Airborne Integrated Data System. The accuracy, response rate, and signal level characteristics for this instrumentation will be coordinated with the airframe manufacturer and the airline customers during Phase III, to assure compatibility with the airframe systems and with the engine functions requiring monitoring. Currently, the JTF17 engine design includes engine instrumentation and provisions for airframe/airline instrumentation as shown in tables 1 and 2.

Table 1. JTF17 Engine Instrumentation

Parameter	Indication	Component
Turbine Exhaust Pressure (P_{T7})	0 - 50 psi	Four probes and averaging manifold
Turbine Exhaust Gas Temperature (T_{T7})	600 - 2000°F	Nine probes wired for average and individual circuits with separate electrical harnesses
Duct Heater Exhaust Nozzle Position (ENPI)	3 - 12 sq feet (0-7 in stroke)	Linear variable reluctance transducer
Reverser-Suppressor Position	Cruise	Electrical Switch
Aerodynamic Brake Position (Inlet Guide Vanes)	Cruise (Start)	Electrical Switch
Secondary Air Valve Position (Boeing only)	Closed	Electrical Switch
Low Rotor Speed (N_1)	0 - 7500 rpm	Inductive Pickup

Table 2. JTF17 Airframe/Airline Instrumentation

Parameter	Provision
High rotor speed tachometer	Pad for tach generator at rear of main gearbox, left side of engine. Pad speed is 0.513 x high rotor speed (N_2)
Lubrication oil temperature	Boss in fuel-oil cooler discharge line, right side of engine
Lubrication oil pressure	Boss in fuel-oil-cooler discharge line, right side of engine
Primary gas generator fuel flow meter	Flow meter location is in the fuel line downstream of the fuel control discharge, left side of engine.
Duct heater fuel flow meter	Flow meter location is in the fuel line between the duct heater fuel pump and the fuel control inlet, right side of engine.
Pressure drop across oil strainer	Bosses in oil pump housing, right side of engine.
Pressure drop across fuel filter	Bosses in fuel pump housing, left side of engine.
Vibration pickup mounting brackets	Space provided for a bracket on two external flanges of the engine, left side of engine.
Fuel pump inlet pressure	Connector on fuel pump housing, left side of engine.
Fuel pump inlet temperature	Connector in fuel pump inlet line, left side of engine.
Lubrication oil quantity	Pad located at top of oil tank, right side of engine.
Chip detectors	Boss in the No. 1 and No. 2 bearing compartment oil sump, bottom of engine
	Boss in gearbox, left side of engine
	Boss in gearbox, right side of engine

The airframe/airline instrumentation components that are selected shall be developed in conjunction with the engine and the engine/airframe development programs. The selection, development and engine testing of these units as components and as part of the engine system shall be conducted in cooperation with the airframe contractor and using airlines during Phase III. This coordinated effort will provide reliable instrumentation for safe operation of the engine and for AIDS monitoring of the engine condition to assure the most economical operation of the SST in airline service.

The flight instrumentation system provides connect points for the airframe as indicated on the Installation Drawing.

2. Description

The JTF17 engine flight instrumentation components will be of the same basic design and employ the same operating principles as similar devices used successfully on other P&WA engines including the TF30 and J58.

The types of sensors and the current state-of-the-art relative to this type of instrumentation is described in the following paragraphs.

a. Turbine Exhaust Pressure

The turbine exhaust pressure will be obtained by employing four pressure probes that are manifolded in the gas generator cavity to provide an average pressure at the external connector on the engine. The probe is a gas averaging unit with total pressure sampling at eight locations across the radius of the engine.

Similar probes have been used successfully on the Phase II-C JTF17 engine for the individual location pressure measurement.

b. Turbine Exhaust Gas Temperature

The turbine exhaust gas temperature measurement will be obtained by employing immersion thermocouples. The gas temperature probe is a gas aspirating unit which consists of five dual chromel-alumel thermocouples contained within a streamlined fairing. The five gas sampling stations, spaced radially on the probe fairing, are designed to measure the average gas temperature in five equal annular areas in the engine. The engine design utilizes nine turbine exhaust temperature probes. The five dual element thermocouples are wired into two separate parallel circuits, thus providing two output signals for flight deck monitoring and ground check of the engine flight deck system. The temperature probe utilizes gas aspiration for rapid response to changes in turbine exhaust gas temperature. This geometry also allows the thermocouple junction to be shielded inside the fairing and out of the "line of sight" of the turbine, thereby reducing radiation effects. The turbine exhaust gas that is aspirated through the probe is ported into the engine tail-cone to achieve the required gas flow across the thermocouple elements. The probes feature a "simple-supported-beam" structural design which provides a considerable reduction of gas bending stresses and improves the vibratory characteristics of the probe.

The JTF17 exhaust gas thermocouple transmission harness is a flexible two-piece assembly covered with a braided nickel wire shielding. A junction box joins the two sections of the harness at the rear of the engine exhaust case facilitating installation and removal of the harness.

The JTF17 exhaust gas instrumentation system employs the design refinements developed in the J58 high Mach number program and in the Phase II-C program. Special efforts have been made to design the system to eliminate problems experienced by airline service on current engines. The features incorporated into the design to meet these goals are:

(1) The installation and removal of the probe through the turbine exhaust nozzle -- eliminates necessity of removing any engine part or engine cowling to replace a defective probe.

(2) Simple-beam-support probe configuration -- provides reduction in gas bending stresses and improves vibratory damping.

(3) Machined-heavy-duty harness eyelets -- provides improved durability in eyelets and eyelet/leadwire joint.

(4) Improved bracket attachments -- provides tight fit between bracket and harness during thermals and during normal installation to avoid chafing.

c. Duct Heater Exhaust Nozzle Position

The exhaust nozzle area position indicator will use a linear variable reluctance transducer connected to the duct exhaust nozzle mechanism to produce an output signal proportional to the duct heater exhaust nozzle area. This electromagnetic transducer has windings encased in a stainless steel housing, and a movable magnetic stainless steel rod, which does not require sliding contacts or moving leads. A fuel cooled unit of this type has been successfully employed on the Mach 3+ J58 engines. This experience will be utilized to develop the JTF17 transducer for accuracy, reliability and durability.

d. Position Indicators

The inlet guide vane position, reverser-suppressor position, and secondary air valve position (Boeing only) will be obtained by the use of mechanically actuated switches designed for precision operation at elevated temperatures. The design utilizes a snap action contact that provides increased reliability by reducing the effect of contact arcing. Switches of this type have been successfully used on the J58 engine at temperatures that are several hundred degrees higher than will be experienced on the JTF17. This high temperature experience will be employed to develop the JTF17 switch to reliability and durability levels consistent with the requirements of commercial service.

e. Low Rotor Speed (N_1)

Low rotor (fan) speed will be obtained by the use of an inductive pickup that is located approximately 0.100 inch above the maximum tip thickness of the second-stage fan blades. This pickup employs the concept known as "stray magnetic field effect"; therefore, no "return" magnetic circuits or paths are necessary. The dynamic discontinuity generated by the fan blades in the field of the pickup pole piece produces an electrical output. This output voltage and wave shape requires amplification and shaping by an amplifier located within 200 feet of the pickup. A differential operational amplifier and an analog or digital readout for this rotor speed are recommended.

The inductive pickup output voltage and wave shape are dependent upon the following characteristics of the interrupting material:

(1) Probe-passing velocity of the blade
(2) Mass or size of the blade
(3) Magnetic properties of the blade
(4) Distance from the pickup

The second-stage fan blades of the JTF17 design will produce the dynamic discontinuity required by the inductive speed transducer. Laboratory tests and J58 development experience have indicated that a specially designed low inductive pickup will produce a useful and reliable speed signal when positioned above a rotating rotor. A similar low inductive pickup has been utilized on the J58 engine as a means of detecting blade flutter. During a laboratory test conducted with the J58 pickup, the output signals were routed through 200 feet of standard cable to a Differential Operational Amplifier which produced a shaped signal of 0 to 20V between speeds of 0 to 30,000 rpm.

The installation provisions for these components are further described in Report B, Section II, Fan and Turbine Sections, and Report D, Section I.

A. INTRODUCTION AND BACKGROUND

The Pratt & Whitney Aircraft JTF17 engine for the supersonic transport is a twin-spool, turbofan engine with thrust augmentation provided by an advanced design duct burner with a variable exhaust nozzle in the fan discharge duct. The gas generator has a fixed area exhaust nozzle.

The P&WA JTF17 engine control system described in this section is a hydromechanical system using the computing and metering devices and main fuel pump common to P&WA commercial engines. The duct augmentation fuel and exhaust nozzle area control mechanisms and the duct and hydraulic fuel pumps have a background of successful supersonic military application. The gas generator fuel control, the duct fuel control, and the duct exhaust nozzle area control are packaged in a common housing (unitized control) and use a common hydromechanical computer to provide additional simplicity. A rapid replacement design is provided to minimize removal and installation time.

1. Requirements for Prototype & Production Engine Control System

A simplified representation of the P&WA JTF17 engine installation for the supersonic transport showing the controlled variables is shown in figure 1. The engine control system controls the gas generator fuel flow, the duct heater fuel flow, the duct heater exhaust nozzle area to obtain total engine airflow control, compressor bleed position, the compressor inlet guide vane angle, the engine reverser-suppressor position, and the duct heater fuel pump speed. The engine and the engine control system are designed to be fully compatible with the airframe inlet and the inlet control system.

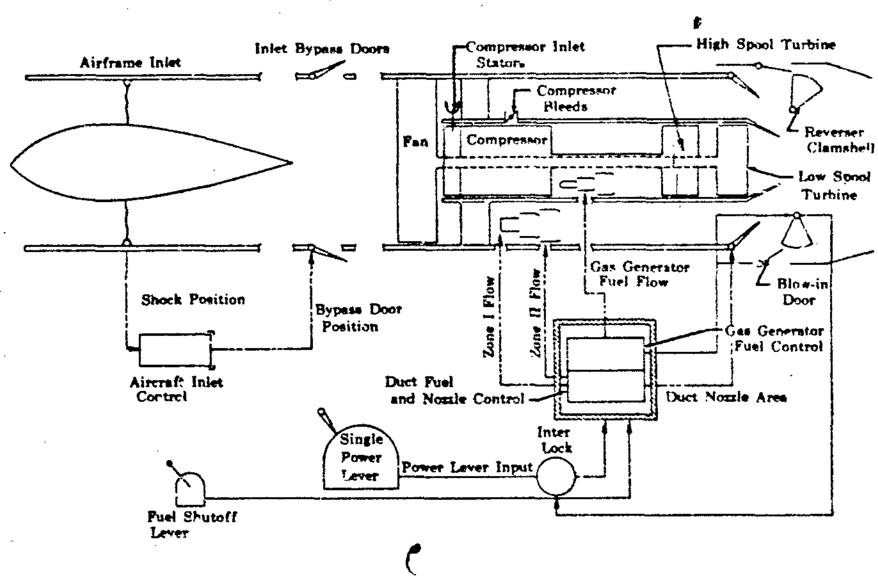

Figure 1. Simplified Representation of JTF17 Engine Installation Showing the Controlled Variables

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Components have been selected to provide a high degree of reliability to achieve a high mean time between failures (MTBF) and a high degree of accuracy in order to permit achievement of high time between overhaul (TBO) on the engine hot section parts as well as achieve minimum thrust specific fuel consumption and maximum aircraft range. Past experience has been used as a guide to obtain a high degree of durability in order to achieve high TBO's on the control system components. The system will function satisfactorily, without the use of external cooling, over the operating envelope of the supersonic transport engine, with its wide ranges of control inputs and outputs and environmental conditions, as indicated in the table below.

Mach number	0 to 2.7 continuous
	0 to 2.9 transient
Altitude	0 to 80,000 feet
Compressor inlet temperature	-65 to 560°F continuous
	-65 to 650°F transient
Fuel inlet temperature	-40 to 310°F
Control component ambient temperature	-65 to 600°F
Control component ambient air velocity	up to 400 fps
Gas generator fuel flow	1200 to 30,000 pph
Duct heater fuel flow	800 to 90,000 pph

The control component engine installation has been designed to provide serviceability and maintainability; the design of the components permits replacement of an installed control systems component on a service aircraft in 30 minutes.

The proposed JTF17 control system has been reviewed with Boeing, Lockheed, the Federal Aviation Agency, the SST Airlines Propulsion Committee, and the Aero Propulsion Laboratory at Wright-Patterson Air Force Base, and incorporates suggestions from these reviews.

2. Background

The development experience, personnel, and facilities utilized to develop the control system for the P&WA J58 engine, which is currently operational above Mn 3, will be used for developing the control system proposed for the JTF17 engine for the Supersonic Transport. The J58 engine installation in supersonic military aircraft has required an engine control that will withstand a more severe environment than that of the SST, as follows:

Mach number	0 to 3 plus
Control component ambient air temp	-65 to 800°F plus
Control component ambient air velocities	up to 400 fps
Fuel inlet temperature	-30 to 300°F
Compressor inlet temperature	-65 to 800°F plus

During the J58 development program, computer techniques for dynamic analysis of the complete powerplant system, including the supersonic engine air inlet, the air inlet control, the engine, and the engine control, to provide a completely compatible engine-inlet system were developed and were used to establish the basic control parameters. As engine

and flight test results became available, the analytic programs have been updated, and have served as a valuable diagnostic tool during the flight test program. A dynamic analysis computer program, based on this J58 experience, has been and will continue to be applied to the JTF17 engine installation as a part of the control development program.

Control component hardware experience acquired on the J58 development program, and which will be applied to the SST program, showed that the high Mn environment imposed special problems that required special attention. For example, this experience showed that the fuel control production acceptance bench tests required a test with fuel at a temperature of 450°F to demonstrate accuracy and performance of each control over the fuel temperature range. Another example is that control schedule shifts were encountered in flight due to thermal expansion because a common ground was not provided on certain control linkages. This type of experience will be applied to the JTF17 to minimize the problems associated with high Mach number operation.

The component test facilities, test techniques, and test personnel at FRDC, as established to develop the J58 engine control components, will be used to develop the SST control components. There are 44 separate component test benches available at FRDC; 15 of these benches include capability for simulating high Mach number fuel, oil, and/or ambient temperature. It is planned to use 25 benches for the JTF17 component development program. These benches will be used to test JTF17 engine components at simulated SST environmental conditions early in the program, thus permitting early detection and correction of problems associated with operation at SST flight conditions. (Refer to paragraph M for a more complete discussion of component bench test facilities.)

The engineering personnel who developed the operational J58 engine control system will be utilized to develop the JTF17 control system. The personnel assigned to the JTF17 control program at FRDC have detailed experience on the J58 engine control program including development, qualification, production, flight testing, and service.

Commercial experience on turbine engine control systems has been and will continue to be thoroughly reviewed and applied, as applicable, to the JTF17 control program. Commercial P&WA turbojet and turbofan engine control systems have accumulated over 39 million hours of operation to date and are continuing to accumulate time at approximately 1,000,000 hours a month. Time between overhaul (TBO) on the fuel control is 8000 hours for American Airlines, 7000 hours for United Airlines, 6600 hours for Pan American Airlines, and in the same range for other airlines. American Airlines demonstrated a reliability of 35,000 hours per premature control removal. The same basic type of control components will be used for the JTF17 engine with additional design features aimed at correcting the problems currently being experienced.

Pratt & Whitney Aircraft maintains a Service Record Department which keeps complete and comprehensive records of all reported discrepancies which occur in service or at overhaul for both military and commercial engines. These data are analyzed on a continuing basis to show trends, significant problem areas and areas in which corrective

action is required. Through this means, the accumulated experience of many millions of hours of operation will be made available to the cognizant JTF17 design and development groups, both at P&WA and at the vendors. (Refer to Volume IV, Report F, Section VI, for a more detailed explanation of this system.)

Special engineering visits were made by FRDC control personnel to review current commercial airline control problems as applicable to the JTF17 at the airline overhaul bases of United Airlines at San Francisco, Trans World Airlines at Kansas City, American Airlines at Tulsa, Eastern Airlines at Miami, Pan American Airlines at New York City, and to the P&WA Turbine Engine Controls and Accessories Maintenance and Overhaul Conference at Windsor Locks, Connecticut, where 15 domestic and 19 foreign airlines were represented. This liaison will be continued throughout the development of the JTF17 engine.

Examples of control problems that have been brought out through reviews of commercial service records with P&WA Service and commercial airlines are: (1) A hard coating was found to be required on relatively lightly loaded contact areas in fuel controls in order to prevent wear and schedule shifts; and (2) Forced lubrication of engine drive splines for components was required to prevent excessive wear. These are a few examples of the design features that will be provided in the JTF17 components as a result of commercial experience and background. Other commercial engine experience as applied to the JTF17 control system is discussed in paragraph D.

P&WA first began studying the engine and control system requirements for the supersonic transport in 1961. Control system programs have been conducted under contract to the Federal Aviation Agency since 1963. These programs have included analytical, computer, and design studies; conceptual designs; airframe coordination; bench testing of certain critical components; and operation of initial experimental JTF17 engines with various control configurations. Over 1600 hours of analytical computer analysis, 10,800 hours of bench testing, and 3000 hours of engine testing of control components applicable to the JTF17 engine installation have been accumulated to date. The basic control mode to be used on the JTF17 engine, the specific parameters to be used, and control schedules for the initial design were established during this program. Several vendors for each major control component have been active in the program. This work, as summarized in paragraphs C and L, and the experience of both P&WA and the active component vendors on current commercial aircraft and high Mach number military aircraft are the basis for the control system and control components proposed for the P&WA JTF17 engine. The continued analytical, computer, design, vendor, bench test, and engine test programs outlined herein for the proposed control system, and the engine-inlet tunnel tests and flight test programs of Phase III will produce the optimum control system for the JTF17 engine, with minimum cost and time required for the SST program.

The computer programs, which are a very essential requirement in control system design, are outlined in paragraph C. Detail descriptions and design and development programs for the major components are included under the separate paragraphs D through K. Control system failure modes

and control system effects studies are discussed in paragraph N. Control reliability, maintainability, service, and safety programs are discussed in paragraph O. Certain advanced technology and growth control concepts and programs are outlined in paragraph F.

P&WA purchase specifications (component procurement specifications) for the major control components for the JTF17 engine have been coordinated with the active vendors, and vendor proposals in response will be received by September in the Phase II-C program. Vendor proposal evaluation by P&WA will be completed and vendor selection will be made during the Phase II-C program. A typical P&WA component purchase specification for the engine control is provided in paragraph Q.

A typical production acceptance test procedure for the gas generator fuel pump is provided in paragraph R. Component purchase specifications and production acceptance test definitions for the major control components have been prepared and are available for review on request.

P&WA contracts with experienced control system component vendors to design, fabricate and assist with development of the major control system components in order to utilize the vendor's special background and capability regarding these components. These capabilities have been developed by working with P&WA on turbine engine components for over 15 years. In addition, both vendors now proposing on the unitized fuel and area control have developed and are supplying fuel controls for the P&WA J58 engine program, and will use this extensive high Mach number background in the development of the JTF17 engine control. Two of the pump vendors that are proposing pump design for the JTF17 engine have also had the benefit of the J58 development and operational experience. P&WA has an established, successful procedure for working with control system component vendors which follows the project engineer format used on engine development The performance, design, installation, reliability, maintainability, safety, quality control, production, and service requirements are defined by the P&WA component purchase specification. Vendor proposals are evaluated on the above items and cost and delivery schedules, and a vendor selected for each component. P&WA Project Engineering coordination and direction continues through the design, development, and qualification testing programs. The component parts list that successfully completes bench testing and engine certification and installation coordination is released to production. Any subsequent changes are subjected to a formal engineering change procedure that is defined by P&WA specifications. Such changes are developed and substantiated as directed by the P&WA project engineer. Vendor procedures are discussed in more detail in Volume V, Report C.

3. General Description of Control System

The control system proposed for the P&WA JTF17 engine for the supersonic transport has the following basic functions:

1. Controls gas generator and duct heater fuel flow, and thereby engine thrust, between full reverse and maximum duct augmentation as a function of power lever angle, burner pressure, compressor rpm, and engine inlet temperature

2. Positions the duct heater exhaust nozzle area to maintain corrected total engine airflow as a function of fan discharge total and static pressures, compressor rpm, and engine inlet temperature

NOTE:

Phase II-C computer studies indicated that air inlet control system - engine control system compatibility requirements could be met without direct communication links between the two control systems. Refer to paragraph C.

3. Provides for operation of the following auxiliary functions:

   a. Varies position of compressor inlet stator as a function of compressor rpm and compressor inlet temperature to obtain optimum compressor performance for starting, takeoff, and cruise
   b. Opens the compressor bleeds to obtain additional margin for starting the engine and closes the bleeds for optimum compressor performance at higher compressor rpm as a function of compressor rpm and engine inlet temperature
   c. Positions the reverser-suppressor as a function of power lever position in the full reverse and the stowed takeoff position
   d. Controls duct heater fuel pump speed to minimum required to provide duct heater fuel flow and pressure

4. Provides the following safety features:

   a. Reduces fuel flow in the event of malfunction causing compressor overspeed and turbine overtemperature
   b. Provides failsafe plateaus on the compressor inlet temperature servo and the speed scheduling cam and failsafe contours on the gas generator metering valve to provide safe operation in the event of failure to extreme positions
   c. Provides interlocks to prevent control power lever motion to reverse or forward unless reverser is in the correct position
   d. Provides for automatic duct heater shutoff in the event of a blow-out (Recycling of power lever to max nonaugmented or below, then back to augmented position, is required for duct heater relight.)
   e. Provides immediate reduction of fuel flow to a selected minimum, which will prevent turbine overtemperature in response to burner pressure decay rate in the event of an inlet unstart or compressor surge
   f. Provides redundant compressor inlet temperature sensors and burner pressure and fan pressure sensors for increased reliability

The proposed control mode and control parameters, which are described in more detail in paragraph D, were selected based on the analytical

programs and experience noted above and will be refined as required during the continued analytical and development programs outlined in paragraphs C thru J.

Power command inputs from the airframe to the control consist of a lever controlling forward and reverse thrust, and a separate lever controlling fuel shutoff. Suitable aircraft electrical switches will actuate the gas generator ignition system and the emergency aerodynamic brake if required. Remote adjustment of gas generator pressure ratio (EPR) will be provided that also results in adjustment of exhaust gas temperature. Remote adjustment of air flow will be provided to permit limited airflow adjustments at cruise. An automatic closed loop mode of EPR control will be available as optional equipment. A description of the control system aircraft-engine interfaces and a description of the control system operational procedure during a typical mission are provided in paragraph B.

The proposed prototype JTF17 engine control system major components are described briefly in the following paragraphs. More detailed descriptions, background information, anticipated problem areas, and a description of the proposed development program for these major control components are provided in paragraphs D through J and in Appendixes A and B.

The basic engine control (the unitized fuel and area control) includes the gas generator and duct heater fuel controls, the duct exhaust nozzle area control, and the auxiliary controls noted above. The unitized concept saves weight and cost, increases reliability by using a common control computer, simplifies engine plumbing and component mounting by locating the various controls in a common housing, and permits rapid replacement to meet the maintainability requirements. Figure 2 shows the unitized control mockup installed on the JTF17 mockup engine. Figure 3 shows the control removed at the rapid replacement manifold, and illustrates how the control can be removed from the mounting adapter, which connects the engine plumbing, without removing the plumbing.

The unitized fuel and area control utilizes the same basic type of hydromechanical hardware that has provided successful service operation of previous P&WA commercial and military turbine engines. The control and the proposed development program are described in detail in paragraph D and in Volume III, Report E, Section II.

The basic control system is hydromechanical to retain the durability demonstrated by this type of control on current engines. It has been "unitized" to improve maintainability. We will also develop a vernier digital electronic engine pressure ratio (EPR) control which will automatically adjust the gas generator fuel control schedule as required to maintain the EPR schedule selected by the flight crew. This closed loop control will also limit the exhaust gas temperature (EGT) at the maximum limit within $\pm$ 10°F indicated. Optimum fuel consumption and engine hot section life will result. The EPR control will be provided as optional airframe-mounted equipment since some airlines may prefer that the flight engineer make all such adjustments manually. The EPR control and proposed program are discussed in paragraph E.

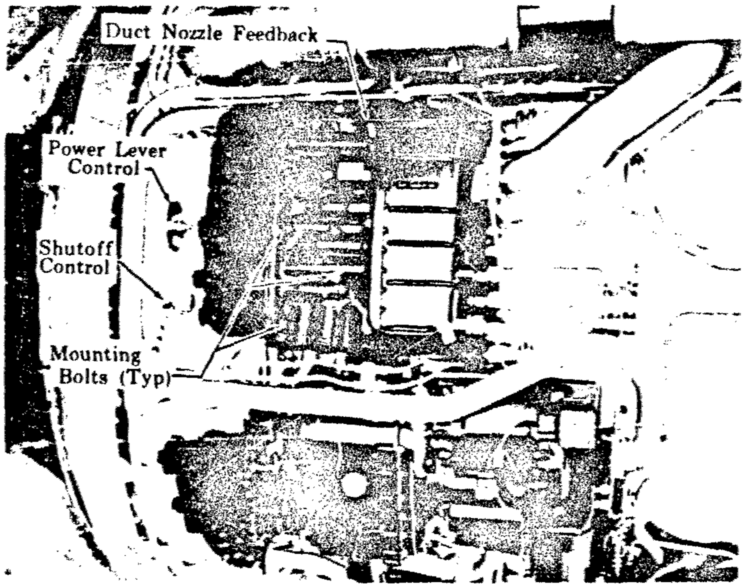

Figure 2. Unitized Fuel and Area Control Mockup Installed on the JTF17 Mockup Engine — FD 16733 BIII

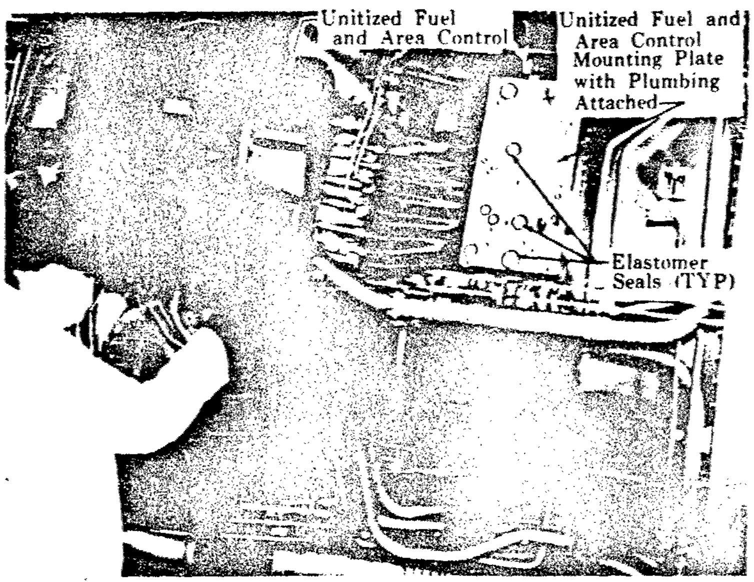

Figure 3. Unitized Fuel and Area Control Removed at Rapid Replacement Manifold — FD 16734 BIII

A digital electronic airflow control, mounted on an interface of the unitized control, will also be evaluated. This electronic control is designed to replace only the airflow computer in the unitized control and will be tested to determine whether or not the improved computer accuracies possible with digital electronics can be realized with adequate reliability in an engine-mounted control. This unit will contain its own engine-driven power supply and be capable of performing in the engine environment without external cooling supply. The control will be engine-mounted to reduce the number of electrical connections for increased reliability. The progress of the development programs will determine if the engine-mounted electronic airflow control, as described above, will be incorporated in production controls.

An engine-driven fuel pump supplies fuel for the gas generator burner. The pump is a two-stage unit with a centrifugal boost stage, a full flow filter, and a single gear stage. This pump is the same basic type that has been utilized in all previous commercial P&WA turbine engines, and has achieved TBO times of 6000 hours to date. The pump and proposed development program are described in paragraph F.

The duct heater fuel pump, which supplies fuel for the duct heater, is a two-stage, fuel-lubricated centrifugal pump driven by an air turbine. This type of fuel pump has shown excellent durability and reliability on the P&WA J58 engine in the high Mach number military applications. The turbine is driven by high compressor discharge air, which is controlled by an externally actuated butterfly valve. The turbine exhaust air passes through a vortex venturi that aerodynamically limits pump speed, without the use of moving parts, by causing the air to build up pressure in the vortex when the turbine begins to overspeed. This pressure build-up results when the swirl angle of the air leaving the turbine becomes tangential. The pump and proposed development program are described in paragraph G.

An engine-driven hydraulic pump supplies 1500 psi fuel to actuate the duct nozzle and reverser-suppressor actuators. Use of engine fuel as the hydraulic actuation fluid results in a system of the lowest possible weight by eliminating the need for hydraulic fluid heat exchangers, reservoirs, and associated plumbing. The need for a special hydraulic fluid and associated problems is also eliminated. The use of fuel in the hydraulic system also permits direct utilization of the heat rejection of the hydraulic system by returning the fuel from the hydraulic system to the gas generator fuel system; the mixed flow then flows to the burner. Fuel hydraulic systems of this type are successfully used on the P&WA J58 and TF30 engines. The hydraulic pump is a variable displacement, piston-type fuel pump, which provides maximum pump efficiency, minimum heat rejection, and rapid response in maintaining constant hydraulic system pressure. Satisfactory experience has been accumulated on a similar fuel hydraulic pump, on the P&WA J58 engine in a high Mach number military application operating at 2500 psi discharge pressure in a more severe environment. The P&WA TF30 military engine (FX-111 aircraft) utilizes a similar 2500 psi fuel piston hydraulic pump. Several British commercial jet engines use 1000 to 1200 psi piston fuel pumps with current TBO's of 6000 hours. Aircraft piston hydraulic pumps, having substantial commercial experience, supply 3000 psi aircraft hydraulic systems and

achieve up to 5000 hours TBO. The above background indicates that the JTF17 engine hydraulic pump can achieve satisfactory TBO times provided that adequate attention is paid to the problems created by the high temperature environment. The hydraulic pump and the proposed development program are discussed in more detail in paragraph H.

The electrical ignition system used to ignite both the duct heater and the gas generator is a 4-joule, 3KV, low-tension system. Satisfactory commercial and military service has been achieved with this type of ignition system. The ignition exciters are fuel-cooled. Heat transfer bench tests at maximum SST mission temperature conditions show that the electrical components in these fuel-cooled exciter assemblies operate at least 10% below their rated temperature limits. Current commercial engine exciter assemblies have achieved a TBO of 2200 hours to date. The electrical ignition system and the proposed development program are discussed in paragraph I.

Other JTF17 engine accessories are described in this proposal: The fuel-oil coolers in Volume III, Report B, Section IV; and the turbine exhaust gas thermocouples and harness, the hydraulic actuators, the aerodynamic brake actuator, and the position switches in Volume III, Report B, Section II.

The relationship of the major fuel control system components to the engine are shown in the fuel and hydraulic systems schematic shown in figure 4. This schematic shows the primary fluid connections between the components. Figure 5 shows the JTF17 engine mockup control components installed on the Boeing mockup engine. A detail fuel and hydraulic system schematic, electrical system schematic, and fuel and lube system thermal management system schematic are provided and discussed in paragraph B.

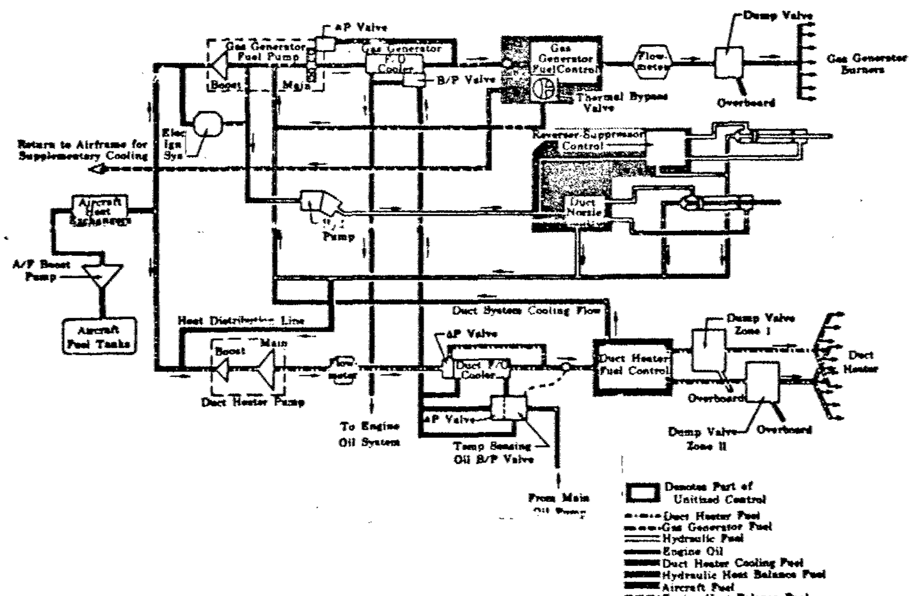

Figure 4. JTF17 Fuel, Hydraulic, and Lubrication Systems Schematic FD 16462 BIII

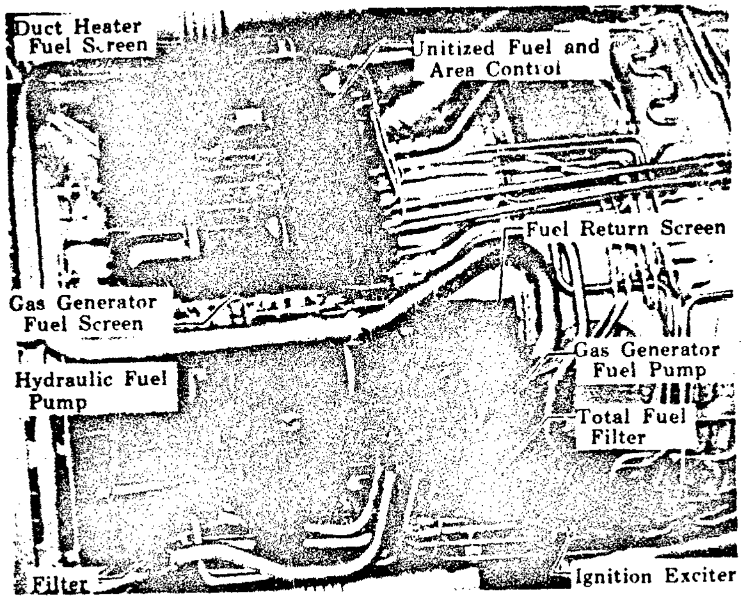

Figure 5. Mockup Control Components Installed on the JTF17 Mockup Engine - Boeing Configuration

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Development engine testing during the early part of Phase III will be supported by the utilization of modified J58 and TF30 engine control components, the same as are satisfactorily being used in the Phase II-C engine test program. These components provide the flexibility required during early engine testing, permit early evaluation of the proposed JTF17 control mode and parameters, and provide engine endurance time cn the same type hardware as will be used in the prototype control components. The early JTF17 engine test experience, as well as final engine sizing and installation coordination with the airframe manufacturer, will be incorporated into the prototype control component detail design prior to release for fabrication.

4. Component Test Plans

The component development test plan for the major control system components is based on P&WA experience in development of turbine engine components. This experience includes the development of control components in commercial use, as well as the development of the components for the supersonic P&WA J58 and TF30 military engines. Component testing follows a well established plan starting with design support tests such as materials and seal test while the unit is being designed.

This is followed by subassembly bench tests, complete unit bench tests, pump and control system bench tests, engine tests including tests with a simulated aircraft fuel system, inlet-engine tunnel tests and flight tests. Both bench and engine tests include performance and endurance testing at SST mission cycle fuel and ambient temperatures. J58 engine experience has shown that this type of component development

test program is required to define problem areas and substantiate the modifications required to correct the problems to insure that the production units will meet the design durability and reliability goals.

The major milestones of the control system component development program are:

1. Completion of a 75-hour component hot mission cycle bench test
2. Completion of aircraft inlet system and engine compatibility testing at the AEDC tunnel facility
3. Completion of a 75-hour engine flight test status test (Establishes prototype unit parts list - defines flight test units)
4. Completion of 100 hours of flight testing in an experimental SST aircraft
5. Completion of a 550-hour component bench test in Phase IV including hot mission cycle and contaminated fuel testing
6. Completion of a 150-hour engine certification test during Phase IV (Establishes production unit parts list)
7. Completion of a 5000-hour component bench endurance test of a production parts list unit during Phase IV to define and correct problems associated with high time endurance
8. Completion of 150-hour hot mission cycle bench quality assurance tests during Phase IV accomplished at P&WA option on one production component out of each lot of 50 for the first 250 delivery units and one unit out of 100 for the remaining production deliveries, selected at random by P&WA in order to demonstrate the quality of production units.

The bench test programs are defined by P&WA and are accomplished both by the vendor or by P&WA as directed by P&WA Engineering. Engine and system testing is accomplished by P&WA. Tunnel and flight tests are coordinated programs with P&WA, FAA, and the Airframe Contractor.

Table 1 shows the bench test, hot bench test, and engine test time which P&WA proposes to accomplish during the development programs for the major control components. A detail description of the development test plan and test schedule for these components is included in Volume III, Report E, Section II.

Table 1. Phase III Major Fuel System Component Estimated Development Test Time

Unitized Fuel and Area Control	1967	1968	1969	1970
Development Units at P&WA	0	16	23	25
Development Units at Vendor	1	3	5	6
P&WA Bench Test Hours	0	5000	9100	7325
Vendor Bench Test Hours	8600	21000	21500	8850
Vendor & P&WA Hot Bench Test Hours	400	2440	4220	2775
Engine Test Hours	0	1000	2850	2925
Gas Generator Fuel Pump				
Development Units at P&WA	1	21	30	35
Development Units at Vendor	1	4	7	9
P&WA Bench Test Hours	0	6500	9000	7880
Vendor Bench Test Hours	800	2500	2600	1500
Vendor & P&WA Hot Bench Test Hours	40	900	1750	1410
Engine Test Hours	0	1000	2850	2925
Duct Heater Fuel Pump				
Development Units at P&WA	1	19	25	30
Development Units at Vendor	1	4	7	9
P&WA Bench Test Hours	0	3000	3600	3600
Vendor Bench Test Hours	750	1950	1250	1610
Vendor & P&WA Hot Bench Test Hours	40	495	725	790
Engine Test Hours	0	1000	2850	2925
Hydraulic Pump				
Development Units at P&WA	1	21	30	35
Development Units at Vendor	1	4	7	9
P&WA Bench Test Hours	0	3000	3600	3600
Vendor Bench Test Hours	800	2700	2600	1600
Vendor & P&WA Hot Bench Test Hours	40	570	930	780
Engine Test Hours	0	1000	2850	2925
Ignition System				
Development sets* at P&WA	10	20	30	35
Development Units at Vendor	4	5	7	8
P&WA Bench Test Hours	2000	4000	3500	1125
Vendor Bench Test Hours	5800	6700	3300	1500
Vendor & P&WA Hot Bench Test Hours	390	1070	1020	395
Engine Test Hours	400	2000	5700	5850

* 2 exciters, 4 igniters, and 4 high-tension leads per engine

B. SYSTEM SCHEMATICS AND SYSTEM INTEGRATION

Paragraph B describes the complete fuel and control system proposed for the JTF17 turbofan engine. This engine has four fluid systems in which control functions are required: (1) the gas generator fuel system, (2) the duct heater fuel system, (3) the fuel hydraulic system, and (4) the oil lubrication system. Included in this paragraph are descriptions of the design and function of the overall system and the manner in which it is integrated into the total aircraft engine system, the complete fuel and hydraulic system, the fuel and lubricant management system, the engine control system operation procedure during a typical mission, and the electrical system wiring diagrams, as well as a summary of the aircraft engine interfaces required. Detail descriptions of the individual components are covered in other sections of this proposal. The overall system is shown schematically in figure 4, paragraph A.

1. Fuel and Hydraulic Systems

The JTF17 fuel and hydraulic systems schematic is shown in figure 1. This includes the gas generator system, the duct heater system and fuel hydraulic system. Fuel is supplied from the aircraft tanks to the boost stage of the gas generator pump. The fuel then flows through a filter to the pump high pressure stage, the hydraulic pump inlet, the ignition exciters for cooling and the duct fuel manifold quick-fill system within the fuel control.

High pressure fuel from the pump discharge is supplied to the unitized fuel control through the gas generator fuel-oil heat exchanger. A parallel bypass circuit around the heat exchanger opens at high flow levels to prevent an excessive pressure loss through the heat exchanger.

Inside the fuel control this high pressure fuel is supplied to the high compressor inlet guide vane system, the gas generator fuel metering valve and the bypass regulating valve. The bypass regulating valve normally bypasses fuel in excess of engine needs to pump interstage through a filter. However, a thermally sensitive valve within the control may reroute part of this bypass fuel to the aircraft tanks if necessary to prevent excessive fuel temperature.

Fuel to operate the computer portions of the control is taken from a wash-type micronic filter in the control fuel inlet line. The computer positions the metering valve in response to the input parameters to provide the correct metered fuel flow to the gas generator combustor.

Metered fuel for the gas generator flows through the control shutoff valve, the fuel manifold and into the gas generator combustor. A manifold drain valve is incorporated which opens after engine shutdown to drain residual fuel in the manifold overboard and prevent possibility of fuel nozzle coking.

The air-turbine-driven duct fuel pump takes fuel from the aircraft tanks and supplies high pressure fuel to the duct heater section of the unitized control. The fuel passes through the duct heater fuel-oil cooler which incorporates a parallel bypass circuit similar to that of the gas generator

fuel-oil cooler system. Because the fuel is supplied by a centrifugal pump, this section of the unitized control incorporates an in-line regulating valve rather than a bypass regulating valve.

When the duct heater is not operating, a metered amount of cooling fuel flow is recirculated through the control to the gas generator pump interstage. When the duct heater is operating, metered fuel is directed to the duct combustor through suitable shutoff valves and manifolds.

The computer portion of the duct heater control positions the duct heater metering valve to the correct position to provide the required metered fuel flow rate. A fuel manifold quick-fill system is incorporated within the control to rapidly fill the fuel manifolds prior to introducing metered flow. This uses interstage fuel and assures fast response of the duct heater. Manifold drain valves drain residual fuel overboard after shutdown.

A piston-type hydraulic pump takes filtered fuel from the gas generator pump interstage and supplies high pressure fuel through a filter to the duct exhaust nozzle control valve and the reverser-suppressor pilot valve. These valves direct the high pressure fluid to the respective actuators to position the fan duct nozzle and the reverser-suppressor as required. Hydraulic flow is then returned to gas generator pump interstage.

2. Fuel and Lube Thermal Management System

Thermal management of the fuel and oil systems is required to satisfy the following basic requirements and is accomplished by fuel oil coolers and temperature sensing fluid bypass valves.

1. Maintain acceptable fuel and oil temperatures
2. Insure maximum utilization of the two fuel heat sinks
3. Return the minimum amount of heat to the aircraft fuel tanks.

The fuel and lube thermal management system is designed to add the maximum permissible amount of heat to the fuel flowing into the engine combustion systems, and the minimum heat back to the aircraft fuel tank.

There are two fuel oil coolers; the duct heater cooler and the gas generator cooler. The duct heater fuel-oil cooler is located between the duct fuel pump and the duct heater fuel control. During aircraft climb, acceleration, and cruise, a large part of the lubrication system heat is transferred to duct heater fuel by the duct heater fuel-oil cooler. This is done to utilize the large heat sink available with duct heater fuel flows.

An oil bypass valve in the duct heater fuel-oil cooler is used to prevent the fuel from reaching overtemperature conditions by sensing fuel temperature at the discharge of the cooler. When the fuel temperature limit is reached, hot oil is bypassed around the cooler core thereby limiting fuel temperature. Under this condition, part of the lube heat load is transferred to the gas generator fuel-oil cooler which is located between the gas generator fuel pump and the gas generator fuel control.

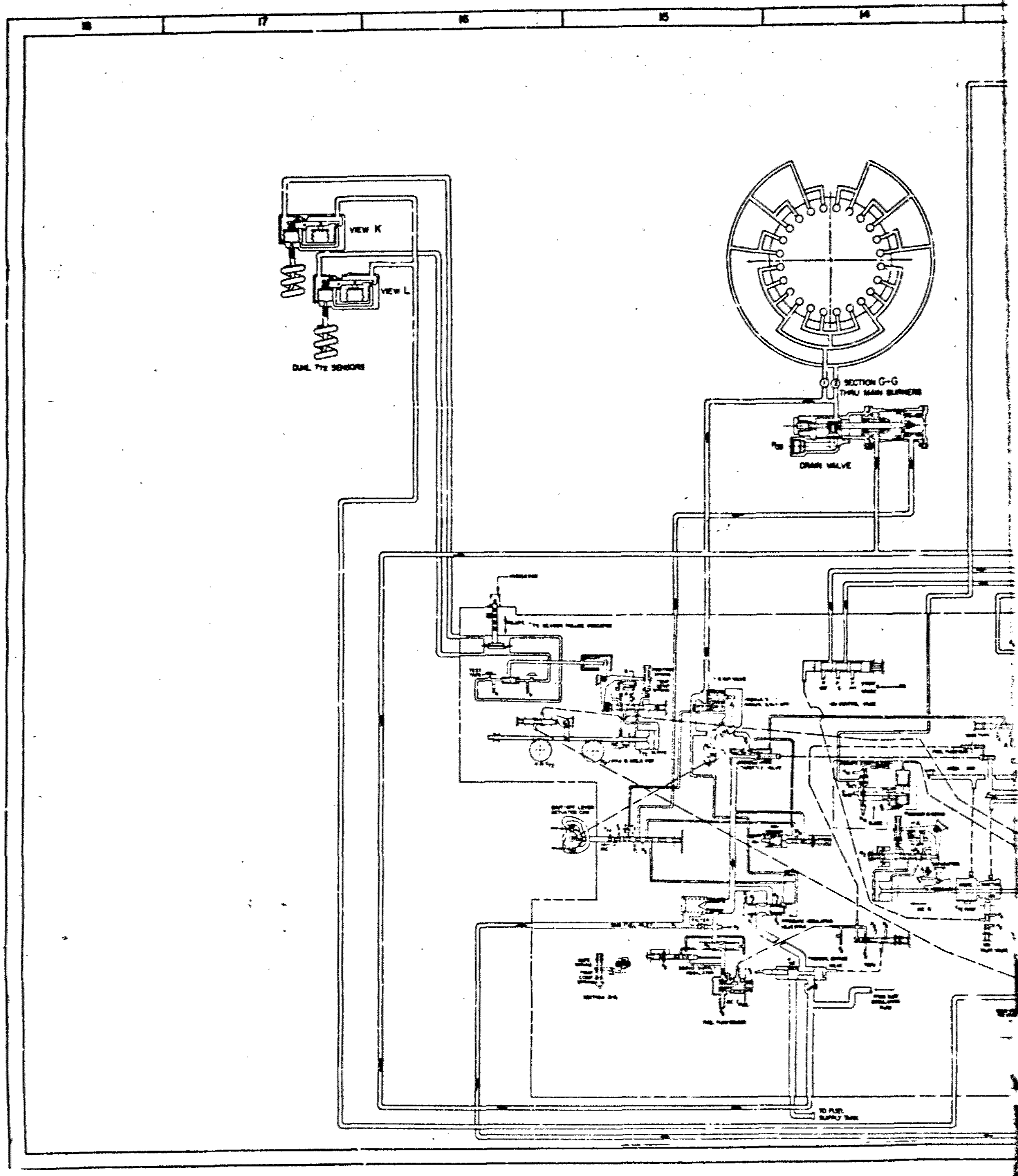
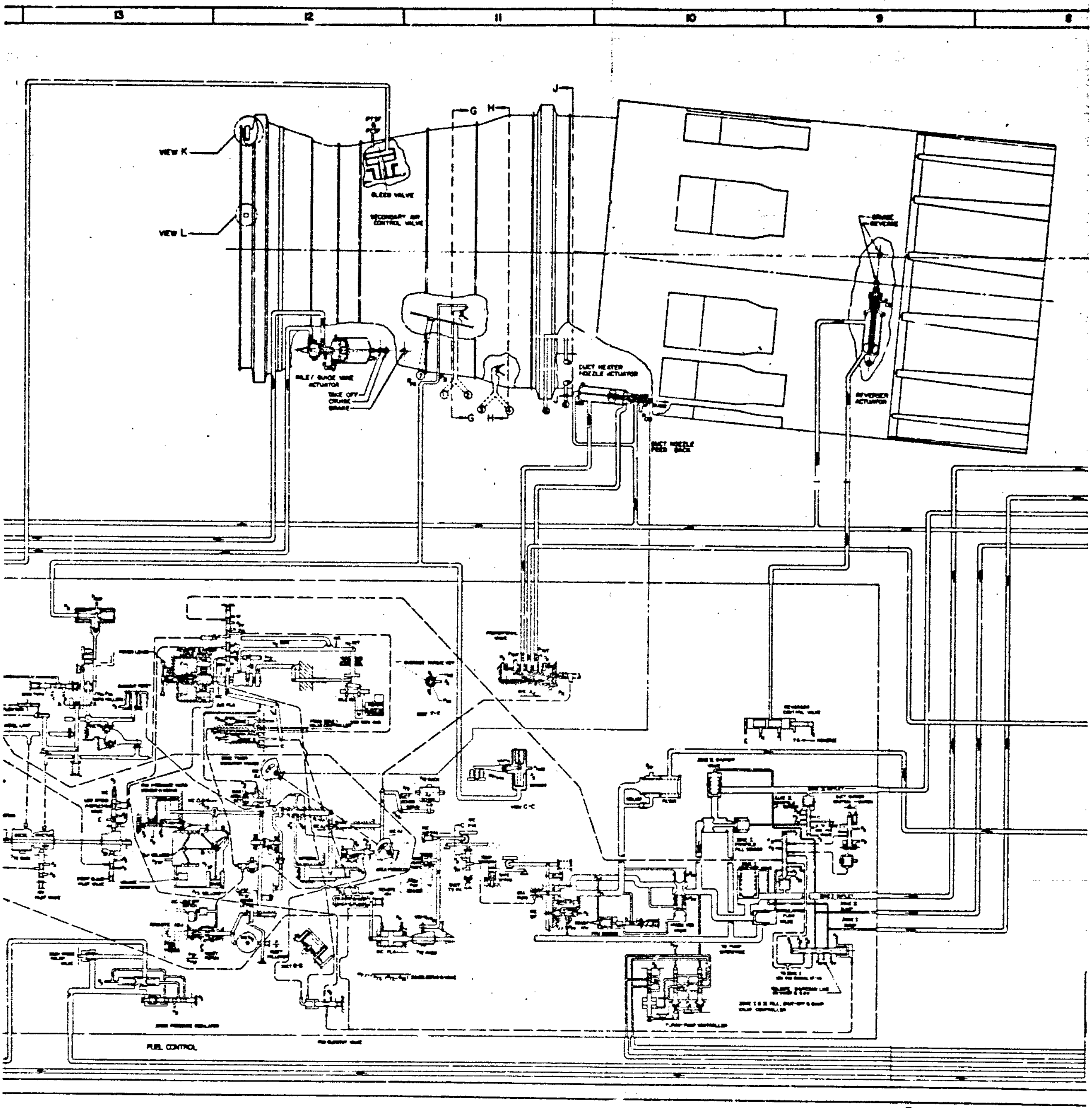
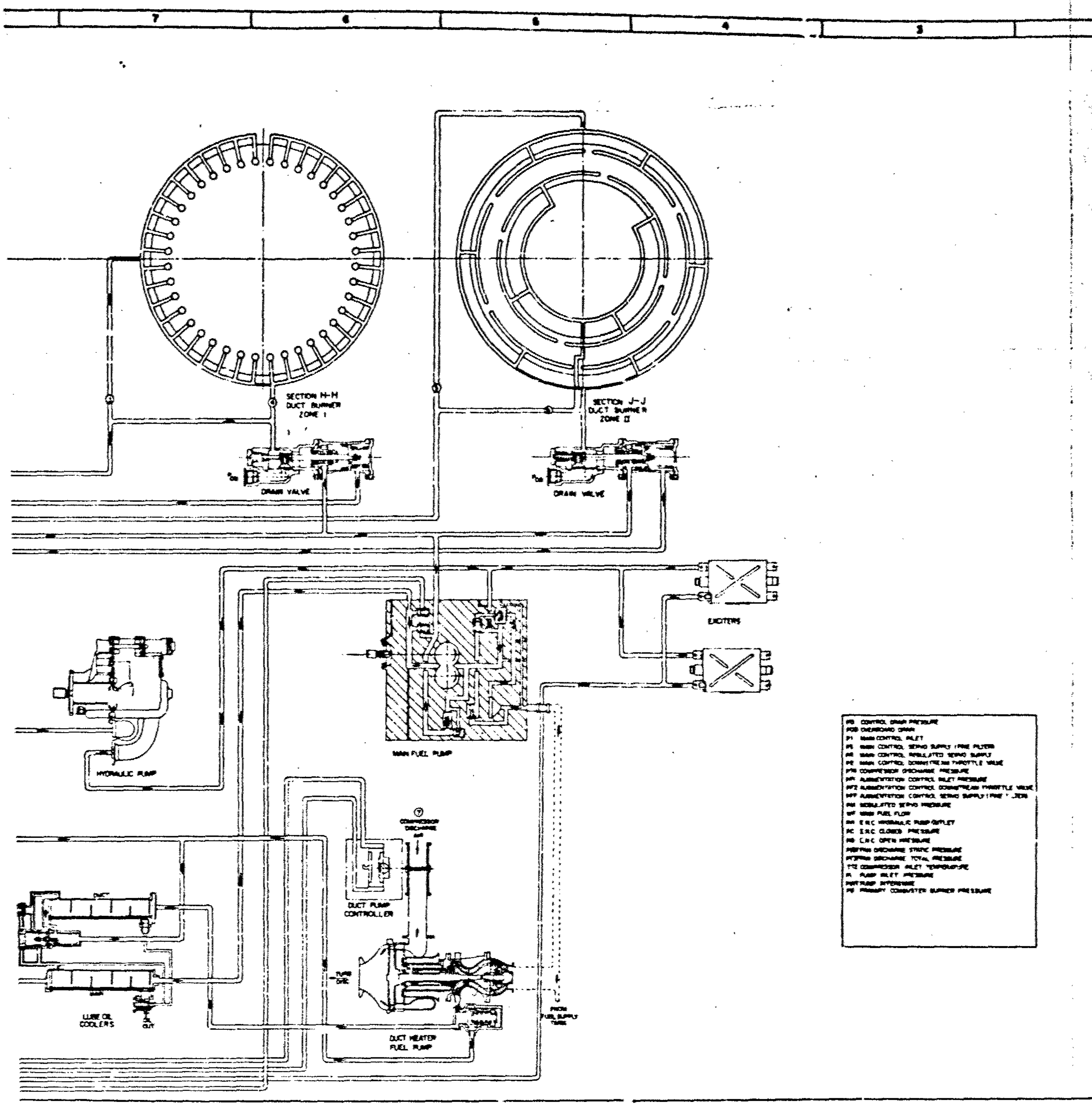


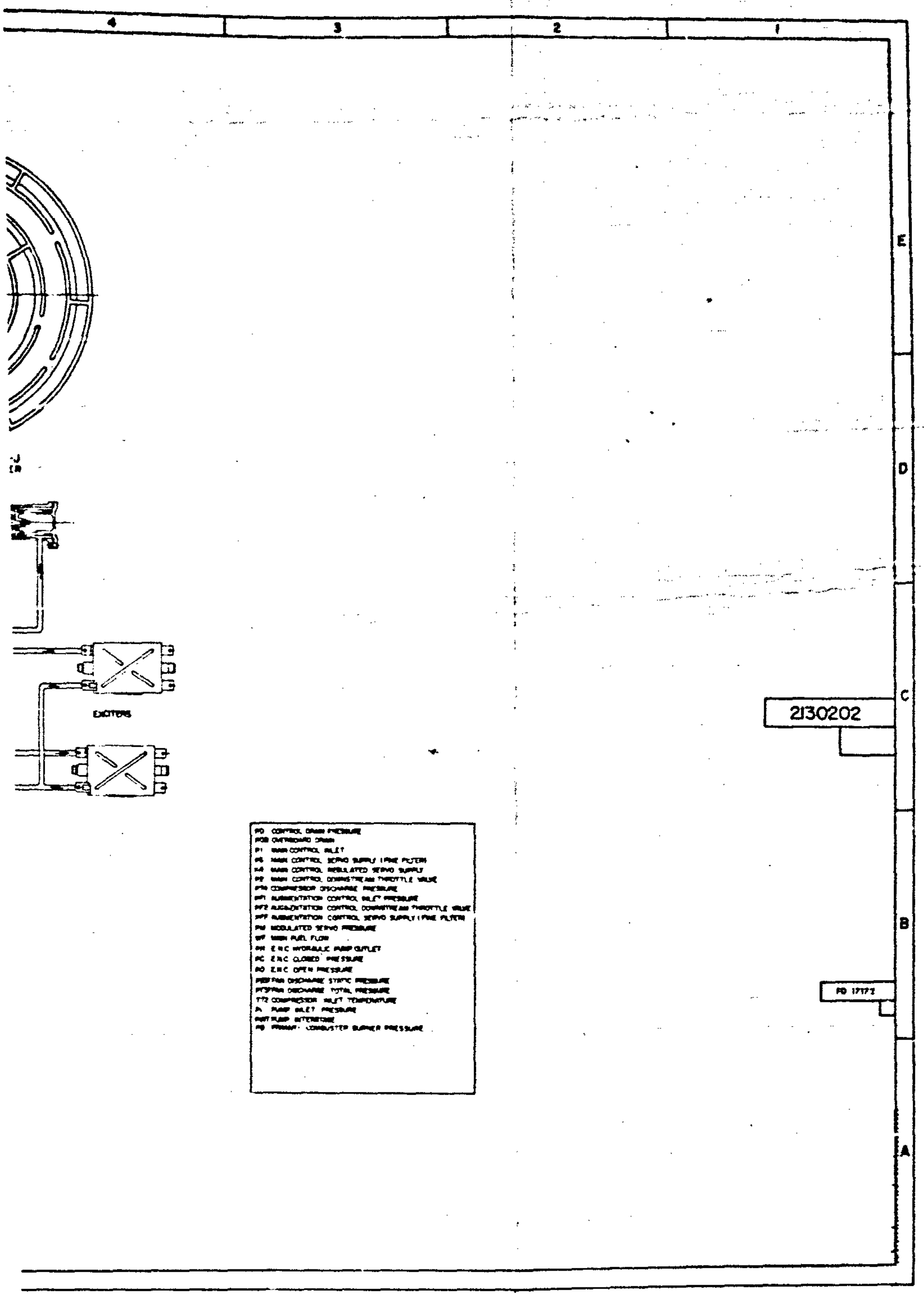
Figure 1. Fuel and Hydraulic Systems Schematic with the Hamilton Standard Division Unitized Fuel and Area Control


13
12
11
10
9
8
VIEW K
VIEW L
BLEED VALVE
SECONDARY AIR CONTROL VALVE
TAKE OFF
CRUISE
BRAKE
DUCT HEATER NOZZLE ACTUATOR
DUCT NOZZLE FEED BACK
BRAKE
REVERSE
REVERSER ACTUATOR
FUEL CONTROL

2


7
6
5
4
3
SECTION H-H
DUCT BURNER
ZONE I
SECTION J-J
DUCT BURNER
ZONE II
DRAIN VALVE
DRAIN VALVE
EXCITERS
MAIN FUEL PUMP
HYDRAULIC PUMP
COMPRESSOR
DISCHARGE
AIR
DUCT PUMP
CONTROLLER
LUBE OIL
COOLERS
OUT
DUCT HEATER
FUEL PUMP
FROM
FUEL SUPPLY
TANK
CONTROL DRAIN PRESSURE
OVERBOARD DRAIN
MAIN CONTROL INLET
COMPRESSOR DISCHARGE PRESSURE
MAIN FUEL FLOW
PUMP INLET PRESSURE

3



FD 17172
BIII

Excessive fuel temperatures are prevented in the gas generator fuel system by the use of a supplementary thermal bypass control valve within the unitized fuel and area control. This valve senses fuel temperature at the discharge of the gas generator fuel-oil cooler. When the limiting fuel temperature is reached, the thermal bypass valve diverts part of the fuel that is being returned to gas generator fuel pump to the airframe fuel tank as required to prevent excessive fuel temperatures.

To minimize the heat returned to the aircraft fuel tanks, the engine system is designed to operate with the maximum permissible fuel temperature into both gas generator and duct burner systems. To accomplish this, and maintain an optimum heat balance between the two systems, some fuel is transferred to the duct heater system from a gas generator system through a fixed orifice.

The thermal management system described above does not require external cooling in any form up to and throughout the cruise regime. During part of the descent after the cruise, fuel may be returned to the aircraft tank until the fuel flow to the engine is sufficient to handle the heat load without exceeding the fuel temperature limit.

At the present time, this engine does not incorporate an air-to-fuel heater for fuel de-icing purposes, because initial coordination with the airframe manufacturers indicated that fuel temperatures at the engine inlet would be above potential fuel icing conditions during all critical flight operations. A fuel de-icer system can be provided on the engine if need is established during detail system coordination with the airframe manufacturer.

3. Electrical System Wiring Diagram Description

The complete engine electrical requirements with circuit information is shown on the electrical installation drawing. The wiring diagram for the Lockheed engine configuration is presented in figure 2 and the Boeing engine configuration is presented in figure 3.

The figures indicate connections with the airframe for the following items:

1. Exhaust gas temperature is indicated by nine individual probes, each probe incorporating two sets of junctions with five per set. One connection with the airframe provides individual outputs that indicate exhaust gas temperature as sensed by the nine probes; each output is the electrical average of one set of junctions. Another airframe connection provides an output that is the electrical average of 45 junctions; these are the remaining set in each of the nine probes.
2. Duct exhaust nozzle position transducer
3. Low rotor speed indicator transducer
4. Two ignition exciters (The ignition system is wired in a manner which will permit only one igniter to be energized at a time if desired by the user.)
5. Unitized fuel and area control remote engine pressure ratio and airflow adjustments and duct ignition control.

6. Position switch that indicates the inlet guide vane position
7. Position switch that indicates the reverser clamshell suppressor position
8. Airframe-supplied decoupler actuator motor for the power takeoff system (Lockheed) (The diagram will be updated to include the engine-furnished instrumentation coordinated with Lockheed.)
9. Four switches that are part of the secondary airflow control system (Two switches indicate that the airflow control valves in two airflow ducts are in the closed position and the other two switches indicate that the valves in two additional ducts are also closed.) (Boeing)

4. Definition of Aircraft-Engine Control Interfaces

Control system design must recognize and define the engine-airframe interfaces. The following interface connections are required and are being provided for on the P&WA JTF17 engine.

a. Power Lever

Power lever input on the engine control requires 320 degrees of shaft rotation from the maximum reverse thrust stop to the maximum augmented forward thrust stop. The power lever provides full engine control for all engine operating conditions except fuel "on" and "off". For the Lockheed application, the power lever input is an electrical system. P&WA will provide fuel cooling for the airframe-supplied, engine-mounted electrical components for actuating the power lever.

Lockheed has also prepared an electrical servosystem that provides a direct electrical input of power lever position to the fuel control. P&WA and Lockheed are studying this system. Reduction of the 320 degrees of power lever motion in the control to the allowable range in the aircraft cockpit is being coordinated with both airframe manufacturers. The 320 degrees of control power lever motion is provided in the control to provide low thrust to power lever motion sensitivity at the control. For the Boeing application, P&WA is providing a gear and rack which converts the 5-inch linear input motion from the Boeing system to the required 320 degrees of power lever rotation.

b. Shutoff Lever

Shutoff lever input to the engine control for both engine applications requires 90 degrees of shaft rotation and either electrical or cable and pulley type inputs can be accepted.

Note that, in the event of external connection disengagement, both the power lever and shutoff lever at the unitized control will remain in the last position to which they were placed and will not change from this position unless externally moved.

c. Ignition Power

Ignition power will be supplied to the gas generator exciters from aircraft manual off-on switches. The use of a separate power source for each switch will provide maximum system redundancy.

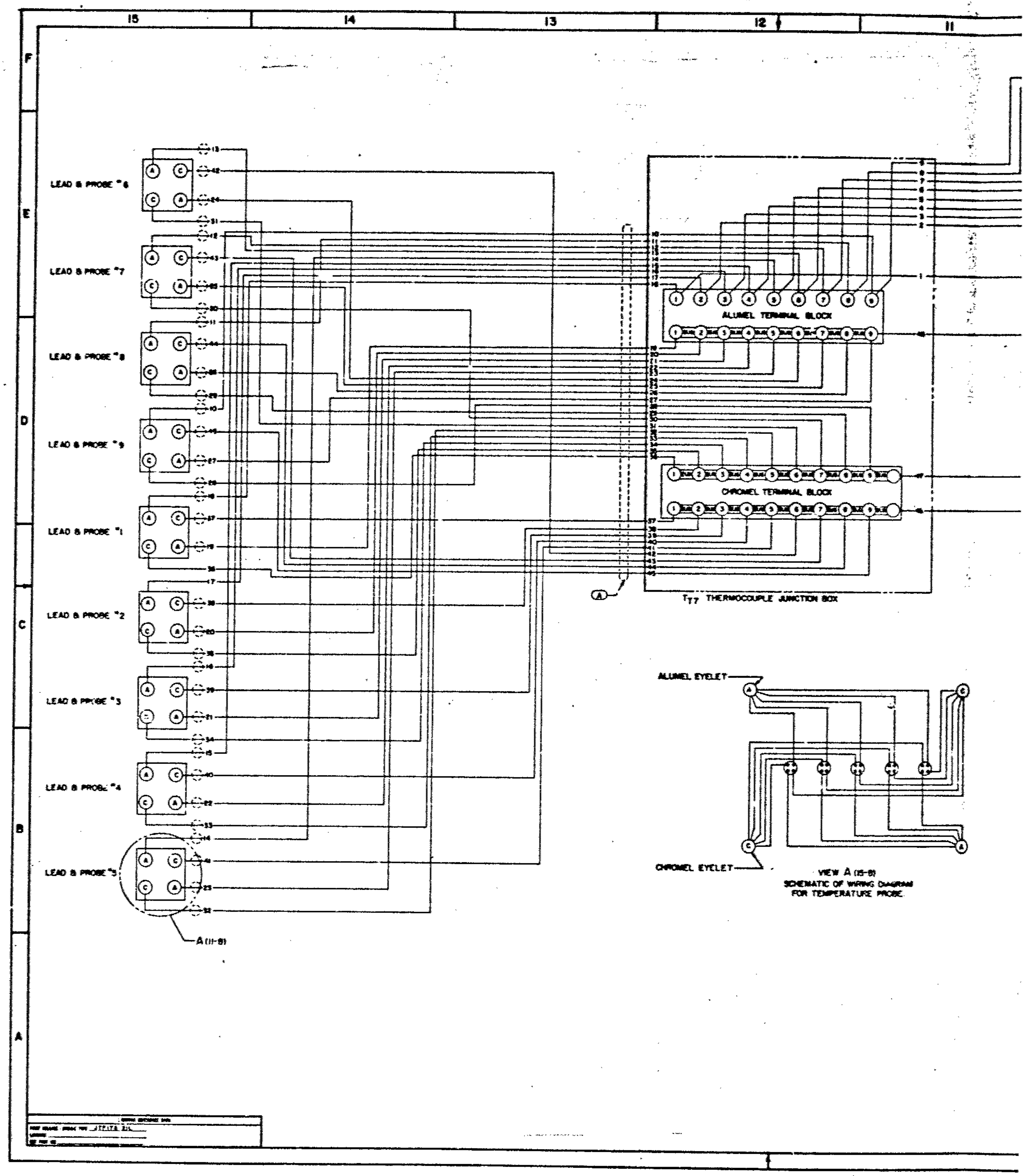
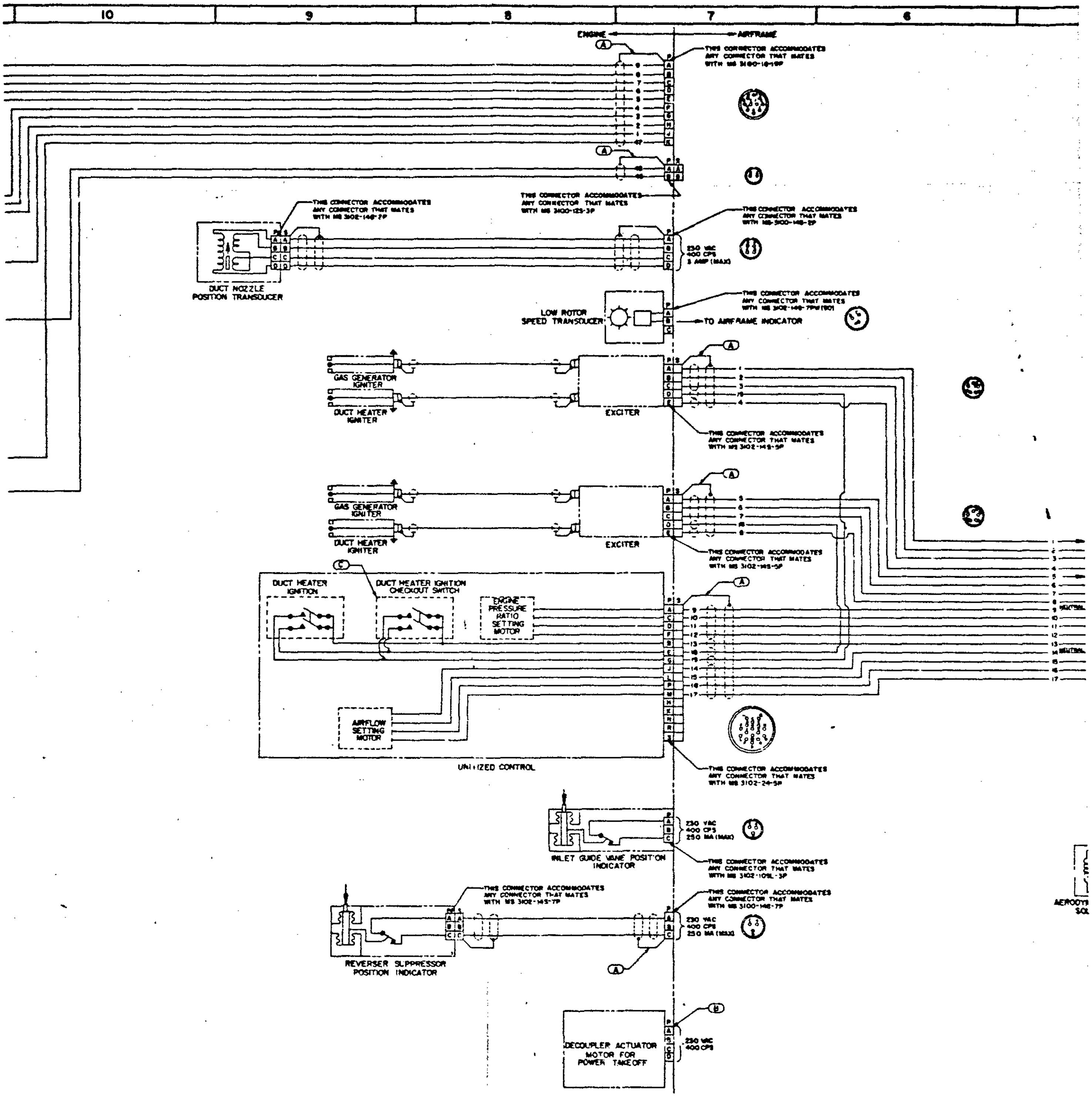


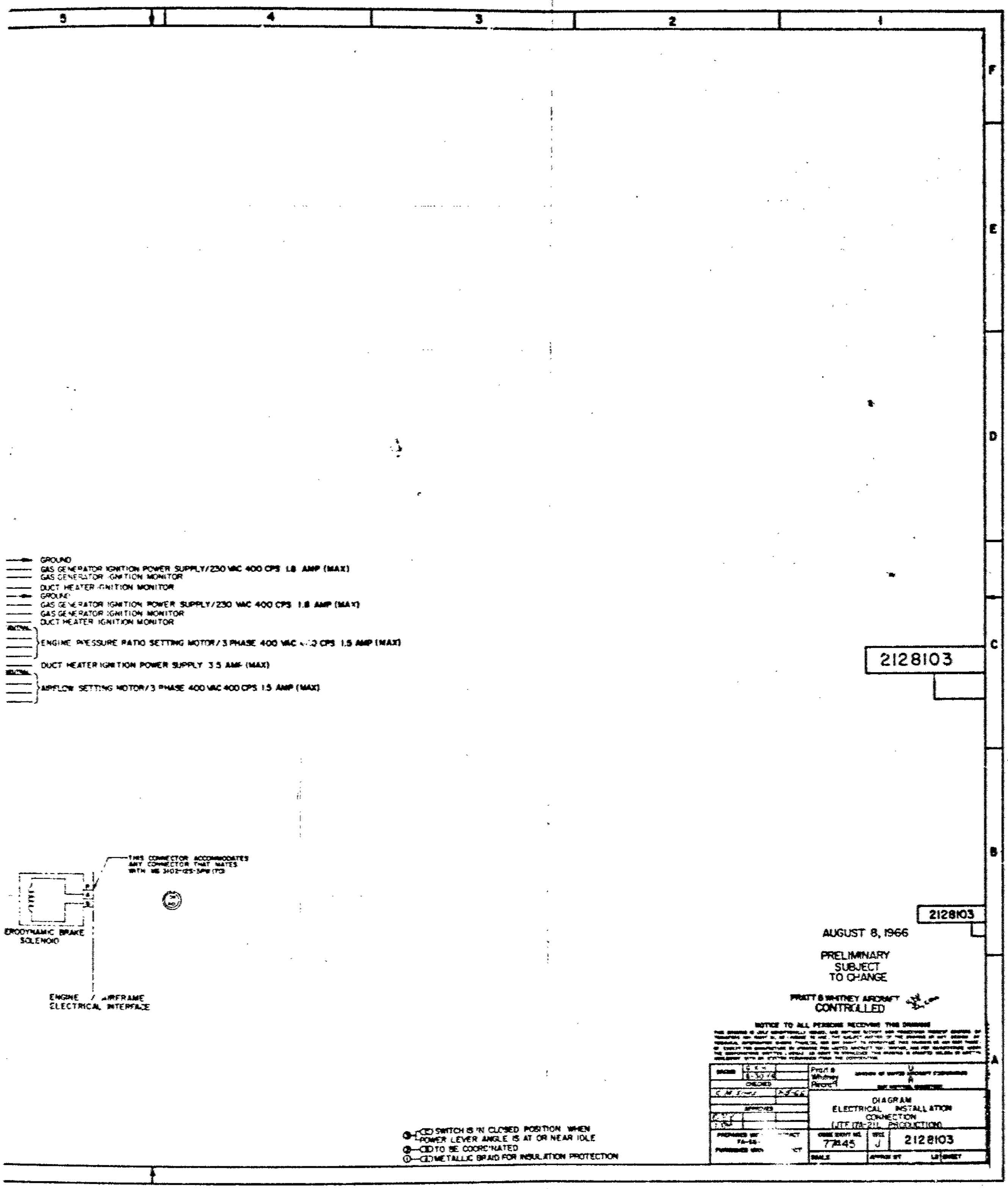
Figure 2. JTF17 Engine Wiring Diagram - Lockheed Configuration



2128103

ENGINE / AIRFRAME ELECTRICAL INTERFACE

2


5
4
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2
1
F
E
D
C
B
A
GROUND
GAS GENERATOR IGNITION POWER SUPPLY/230 VAC 400 CPS 1.8 AMP (MAX)
GAS GENERATOR IGNITION MONITOR
DUCT HEATER IGNITION MONITOR
GROUND
GAS GENERATOR IGNITION POWER SUPPLY/230 VAC 400 CPS 1.8 AMP (MAX)
GAS GENERATOR IGNITION MONITOR
DUCT HEATER IGNITION MONITOR
ENGINE PRESSURE RATIO SETTING MOTOR/3 PHASE 400 VAC 400 CPS 1.5 AMP (MAX)
DUCT HEATER IGNITION POWER SUPPLY 3.5 AMP (MAX)
AIRFLOW SETTING MOTOR/3 PHASE 400 VAC 400 CPS 1.5 AMP (MAX)
2128103
THIS CONNECTOR ACCOMMODATES ANY CONNECTOR THAT MATES WITH MS 3102-12S-3P# (73)
AERODYNAMIC BRAKE SOLENOID
ENGINE / AIRFRAME ELECTRICAL INTERFACE
2128103
AUGUST 8, 1966
PRELIMINARY
SUBJECT
TO CHANGE
PRATT & WHITNEY AIRCRAFT
CONTROLLED
NOTICE TO ALL PERSONS RECEIVING THIS DRAWING
3 SWITCH IS IN CLOSED POSITION WHEN POWER LEVER ANGLE IS AT OR NEAR IDLE
2 TO BE COORDINATED
1 METALLIC BRAID FOR INSULATION PROTECTION
CHECKED
APPROVED
Pratt & Whitney Aircraft
DIAGRAM
ELECTRICAL INSTALLATION
CONNECTION
(JTF17A-21L PRODUCTION)
77445
J
2128103
SCALE
SHEET

3

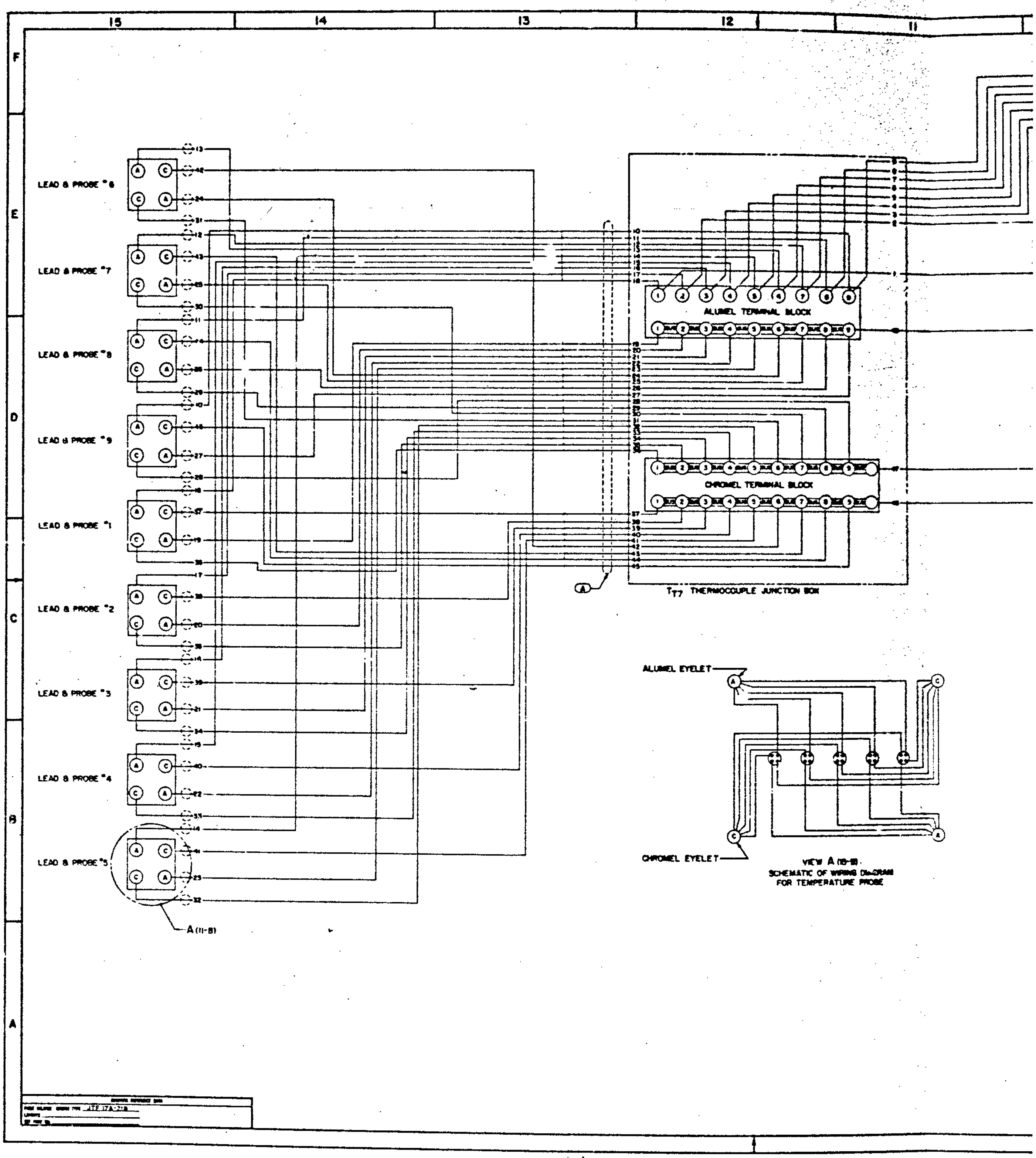
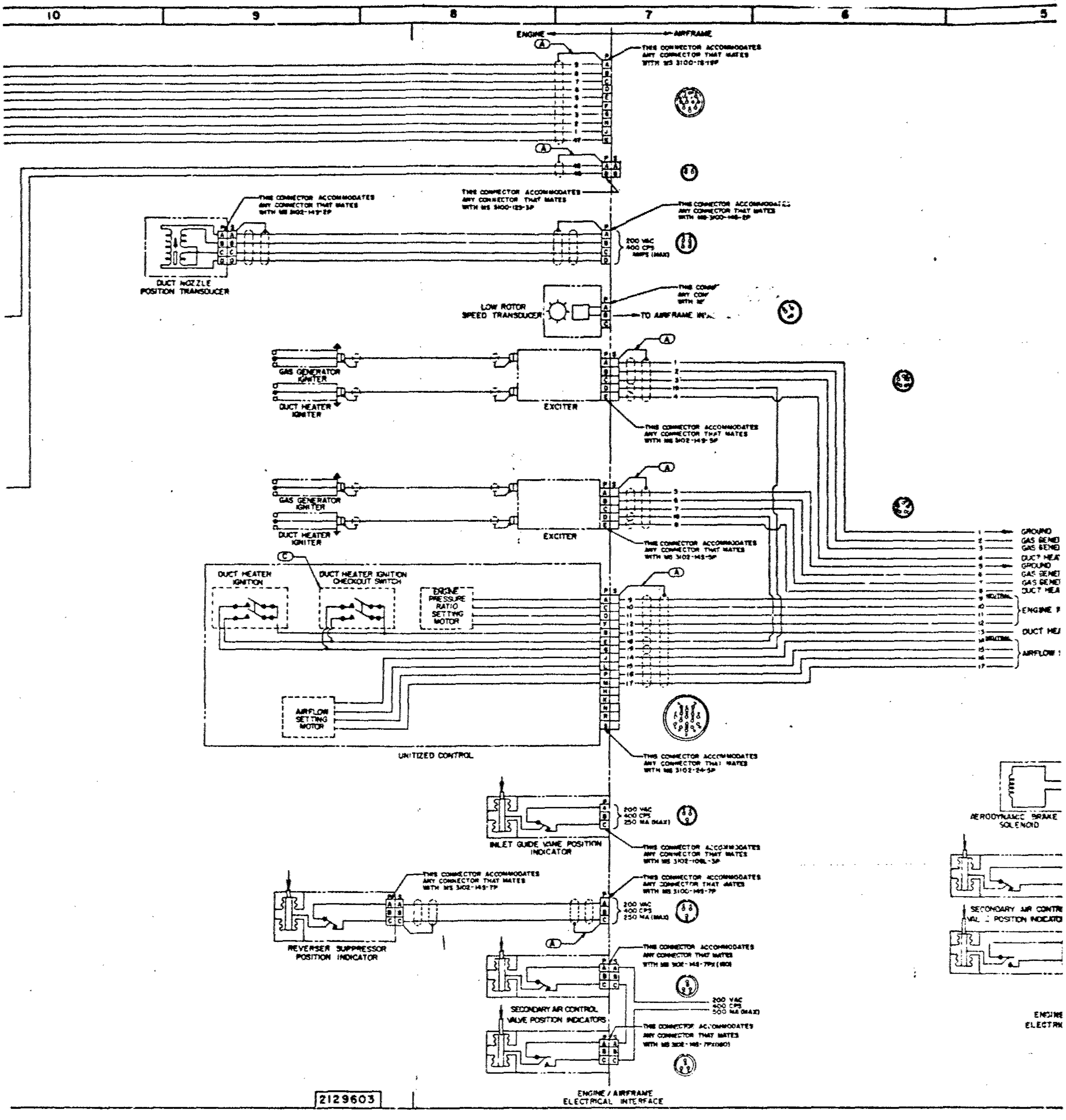


Figure 3. JTF17 Engine Wiring Diagram - Boeing Configuration


10
9
8
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5
ENGINE
AIRFRAME
DUCT NOZZLE POSITION TRANSDUCER
LOW ROTOR SPEED TRANSDUCER
GAS GENERATOR IGNITER
DUCT HEATER IGNITER
EXCITER
DUCT HEATER IGNITION
DUCT HEATER IGNITION CHECKOUT SWITCH
ENGINE PRESSURE RATIO SETTING MOTOR
AIRFLOW SETTING MOTOR
UNITIZED CONTROL
INLET GUIDE VANE POSITION INDICATOR
REVERSER SUPPRESSOR POSITION INDICATOR
SECONDARY AIR CONTROL VALVE POSITION INDICATORS
AERODYNAMIC BRAKE SOLENOID
GROUND
200 VAC
400 CPS
250 MA (MAX)
ENGINE / AIRFRAME ELECTRICAL INTERFACE
2129603

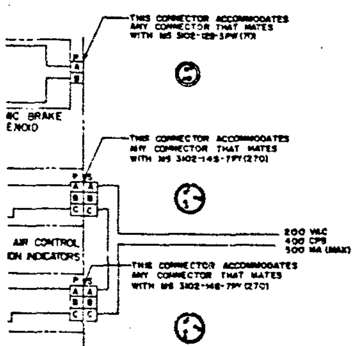
2

5 4 3 2 1

F E D C B A

GROUND
GAS GENERATOR IGNITION POWER SUPPLY / 200 VAC 400 CPS 1.8 AMP (MAX)
GAS GENERATOR IGNITION MONITOR
DUCT HEATER IGNITION MONITOR
GROUND
GAS GENERATOR IGNITION POWER SUPPLY / 200 VAC 400 CPS 1.8 AMP (MAX)
GAS GENERATOR IGNITION MONITOR
DUCT HEATER IGNITION MONITOR
ENGINE PRESSURE RATIO SETTING MOTOR / 200 VAC 400 CPS 1.5 AMP (MAX)
DUCT HEATER IGNITION POWER SUPPLY / 200 VAC 400 CPS 3.5 AMP (MAX)
AIRFLOW SETTING MOTOR / 200 VAC 400 CPS 1.5 AMP (MAX)

2129603



ENGINE / AIRFRAME
ELECTRICAL INTERFACE

SWITCH IS IN CLOSED POSITION WHEN POWER LEVER ANGLE IS AT OR NEAR IDLE
TO BE COORDINATED
METALLIC BRAID FOR INSULATION PROTECTION

2129603

AUGUST 8, 1966

PRELIMINARY
SUBJECT
TO CHANGE

PRATT & WHITNEY AIRCRAFT
CONTROLLED

NOTICE TO ALL PERSONS RECEIVING THIS DRAWING

CHECKED	Pratt & Whitney Aircraft
C. M. Lewis	DIAGRAM ELECTRICAL INSTALLATION CONNECTION (JTF 17A-21B PRODUCTION)
APPROVED	
CODE IDENT NO. 77445	SIZE J	2129603
SCALE	APPROX WT	LH SHEET

3

Ignition power for the duct exciters will be supplied through the unitized control.

d. Remote Adjustment

(1) Gas Generator Fuel Schedule Adjustment

Suitable power connections and aircraft switches are required to power and permit remote adjustment of engine pressure ratio.

(2) Remote Airflow Schedule Adjustment

Suitable power connections and aircraft switches are required to power and permit remote adjustment of corrected engine airflow, thereby providing the desired compatibility with the aircraft inlet system.

e. Fuel

Fuel is supplied from the aircraft fuel system to the inlet of the gas generator fuel pump and duct heater fuel pump through suitable connections. Fuel may be returned from the engine to the aircraft tank during descent to prevent fuel temperature from exceeding limits.

f. Component Installation Drawings

Component installation drawings, as defined in the engine installation drawings, have been coordinated with the airframe manufacturers to fit within available space envelopes in order to provide accessibility on the aircraft, eliminate large fuel lines from the bottom of the engine, provide minimum fuel line lengths, and provide suitable aircraft interfaces.

5. Control Component Arrangement and Installation

Numerous studies have been made which have resulted in an optimum arrangement of control system components on the JTF17 engine. The objectives used for the studies were to (1) keep major fuel handling components and plumbing lines from being located below the engine, as a safety feature in the event an incident occurs resulting in an engine being dragged on the ground; (2) provide easy access to the components that require routine servicing; (3) ensure components are capable of being replaced on an installed engine within one half hour; and (4) obtain acceptable routing of plumbing lines. The final component arrangement met these requirements.

6. Engine Control System Operation During Typical Mission

The pilot action required to operate the JTF17 engine, and the system sequencing which results, is best described by presenting a typical mission profile from engine startup through shutdown. Refer to Volume III, Report A, for a discussion of flight power settings.

The airframe system incorporates a power lever, a fuel shutoff lever, an ignition selection switch, a gas generator pressure ratio (EPR) adjustment switch, and an airflow adjustment switch. Engine power modulation is accomplished by the use of the power lever. Engine shutdown is accomplished through the use of the fuel shutoff lever. Remote manual adjustment of EPR and engine airflow is provided for optimizing the operation of the engine-aircraft system. The schedules and functions of the power lever and shutoff lever are shown in figures 4, 5 (Boeing), and 6 (Lockheed). A simplified electrical schematic of the ignition system is shown in figure 7. The EPR schedule of the engine is shown in figure 8. Desired variations from this schedule can be obtained with the EPR adjustment switch when the engine is operating with augmentation or by power lever modulation when the engine is operating non-augmented. If the exhaust gas temperature maximum limit schedule, as shown in figure 9, is exceeded, adjustment of EPR setting switch will eliminate the condition. The airflow setting switch permits the engine airflow to be varied between certain limits when operating in the high Mach number region to optimize the match between engine airflow with inlet airflow. The airflow schedule and permissible manual setting range is shown in figure 10 for the Lockheed configuration and figure 11 for the Boeing configuration. The EPR setting switch will be used by the pilot or flight engineer to implement the engine constant thrust rating at compressor inlet temperatures below 59 F.

Prior to engine start, the levers are positioned as follows: (1) the power lever is located at the "idle flat position," and (2) the fuel shutoff lever is in the "off" position.

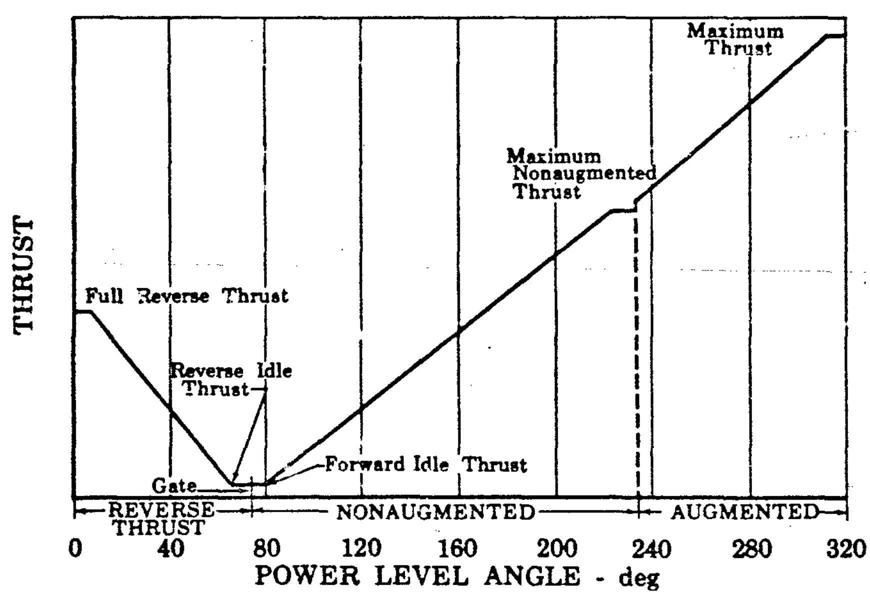


Figure 4. Proposed Power Lever Operation FD 17142 BIII

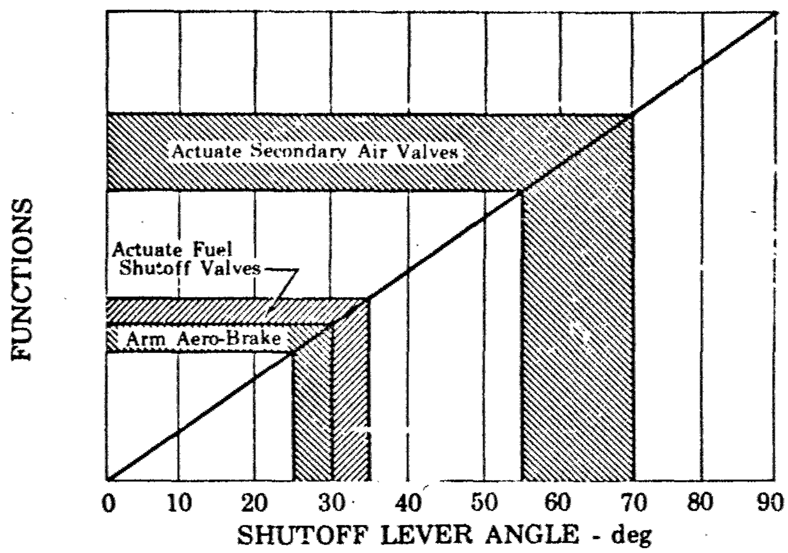


Figure 5. Proposed Shutoff Lever Operation - Boeing FD 17634 BIII

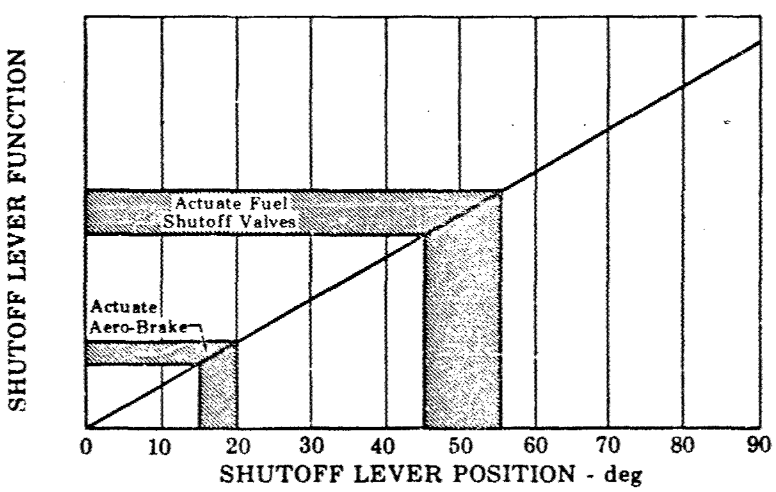


Figure 6. Proposed Shutoff Lever Operation - Lockheed FD 17646 BIII

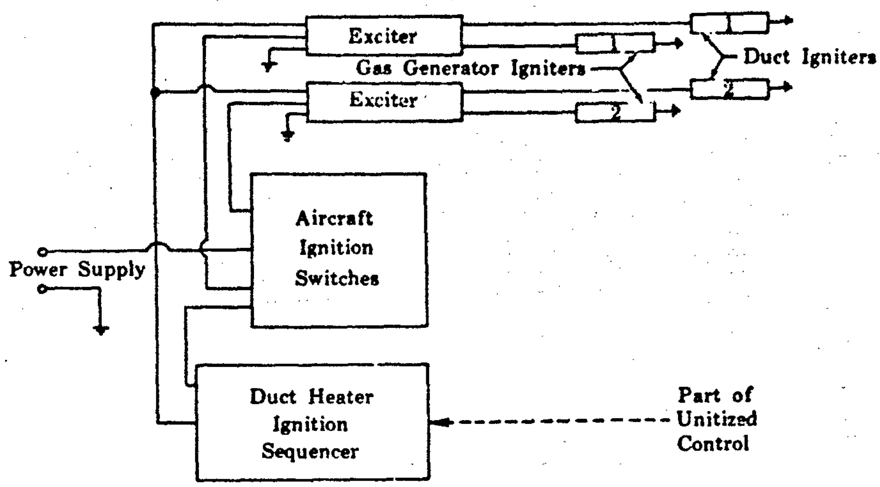


Figure 7. Ignition System Schematic FD 17140 BIII

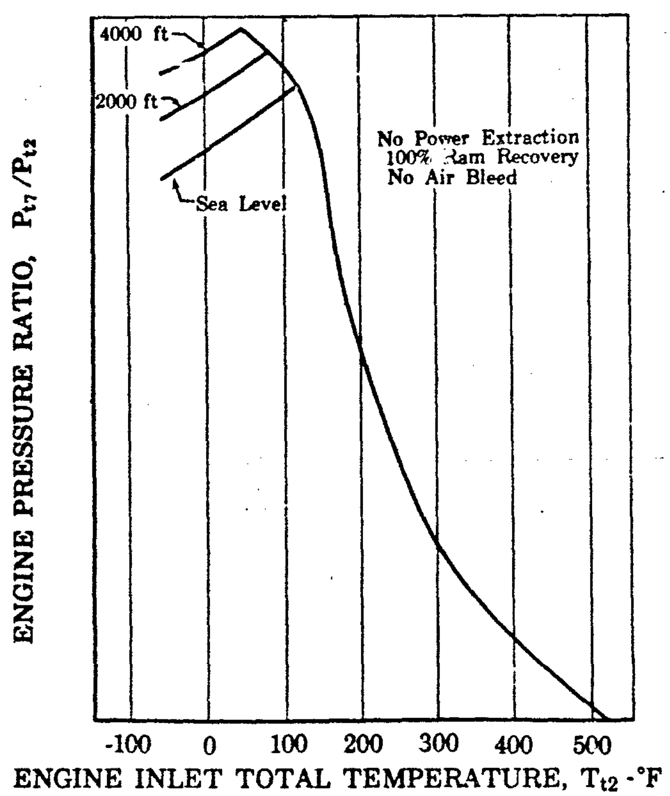


Figure 8. Estimated Performance - EPR Setting Curve FD 17139 BIII

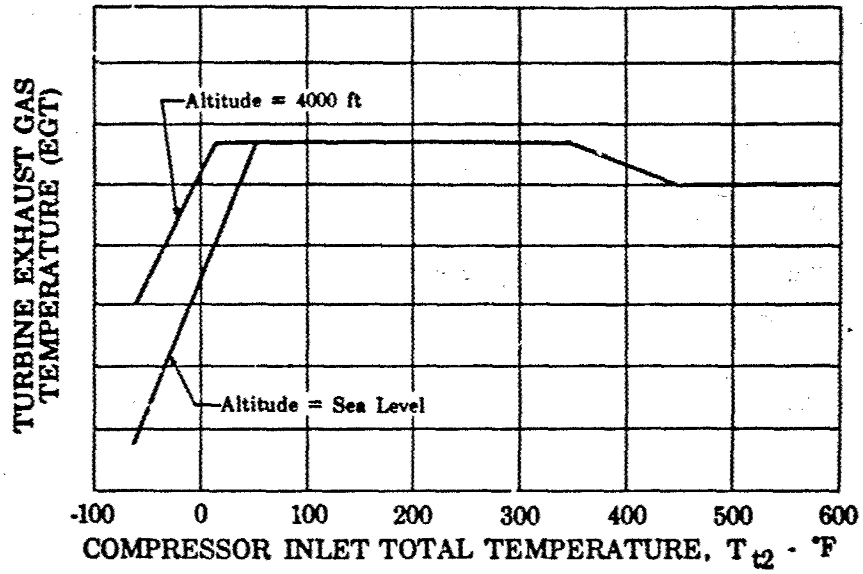


Figure 9. Estimated Performance at Maximum Turbine Exhaust Gas Temperature FD 17138 BIII

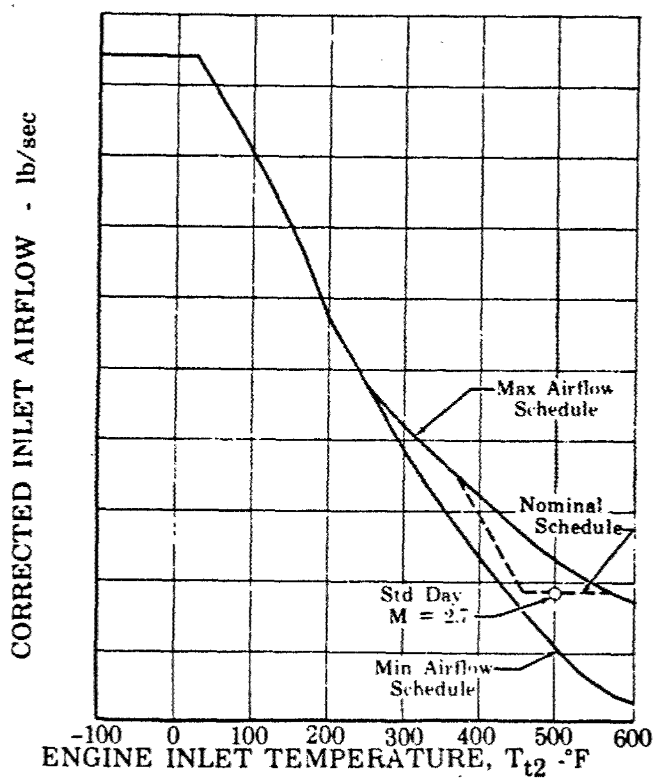


Figure 10. JTF17 Engine Proposed Airflow Schedule - Lockheed FD 17136 BIII

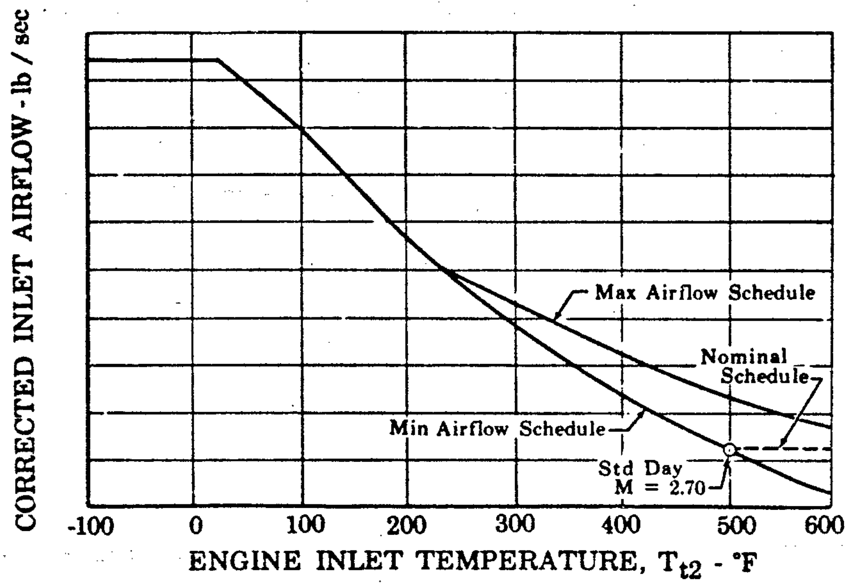


Figure 11. JTF17 Engine Proposed Airflow Schedule - Boeing

FD 17137
BIII

Engine systems are automatically positioned in the shutdown condition as follows: (1) the compressor stator vanes are in the start position, (2) the compressor bleed valves are spring-loaded open, (3) the duct nozzle is partially open, (4) the reverser-suppressor is in the takeoff position, (5) the ejector blow-in doors are open, and (6) the fuel manifold dump valves are spring-loaded open.

Engine start is accomplished by energizing the engine starter and, at the first indication of engine rotation, moving the ignition switches to the "on" position and the fuel cutoff lever to the fuel "on" position. Motion of the shutoff lever signals the gas generator fuel manifold dump valve to close, fuel is introduced to the gas generator fuel nozzles at the starting flow rate, and engine ignition occurs. Fuel pressure rises to maintain the compressor stator vanes in the start position and to set the duct exhaust nozzle area at the constant controlled area.

The engine rotors accelerate in response to fuel flow established by the starting fuel schedule. When the idle speed is reached, the speed governor prevents further fuel flow increase and controls the engine at the idle speed. The gas generator ignition switches are then placed in the "off" position. The gas generator ignition switches should remain in the "off" position unless restart is required or ignition "on" is desired during takeoff or when turbulent air conditions are encountered. However, the ignition usage cycle recommended in paragraph I, should be followed.

Power lever advancement to a position above the idle flat causes the engine to accelerate in response to the fuel flow scheduled by an acceleration limit cam in the fuel control. Part power is set by the power lever and held by the speed governor. As power is increased toward the maximum nonaugmented power, the compressor bleeds close at the proper high compressor rotor speed for the measured engine inlet temperature. Also, at a predetermined high rotor speed, the compressor stators rotate from the start to the takeoff position. When engine total airflow reaches the value scheduled for the operating condition, the duct nozzle begins to close to maintain engine airflow constant. In the maximum, nonaugmented and the augmented thrust range, the fuel/air ratio of the gas generator is held constant at a given engine inlet temperature but is biased by this temperature over the engine operating envelope.

Throughout engine operation with the duct heater off, a small fuel flow is metered through the duct heater fuel scheduling system components for cooling. The duct fuel pump speed is controlled to regulate the metering head, and the duct manifold dump valves are open venting the duct heater fuel manifolds overboard. In addition to providing the cooling function, this also permits all servosystems to be in the correct scheduling position when duct heater operation is requested.

Advancement of the power lever into the augmentation range will initiate duct heater operation with events being sequenced as defined in unitized fuel and area control description, paragraph D.

The thrust setting technique to be used during the takeoff consists of advancing the power lever to the maximum augmented position and setting engine pressure ratio to the desired value, this value being a function of engine inlet air temperature. Where thrust settings below maximum are desired, this same technique is used, followed by a reduction of the power lever to set total engine fuel flow to a value previously determined as a function of engine inlet temperature and desired thrust as shown in figure 12.

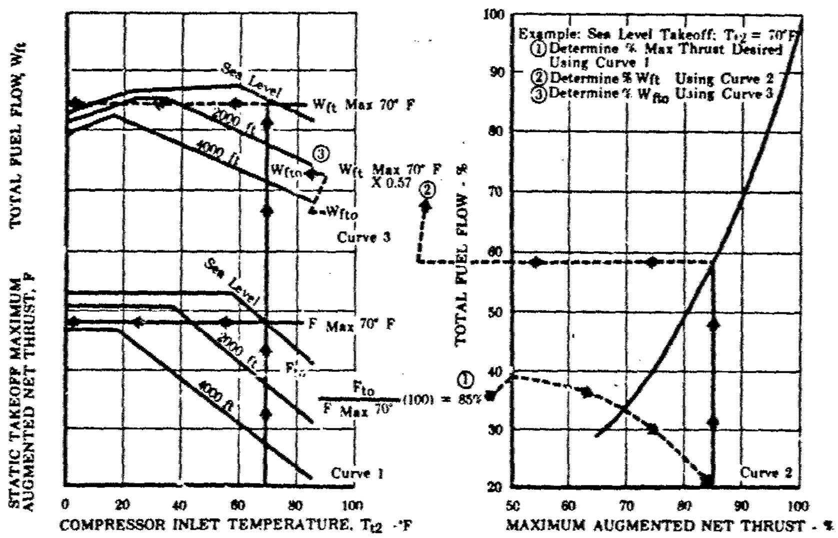


Figure 12. Estimated Performance (Takeoff Part Power Total Fuel Flow Setting) FD 17143 BIII

At a flight condition of approximately Mach 1.2, the reverser-suppressor will be aerodynamically reset to the cruise position. The compressor stator will be automatically positioned in the cruise position by the control at the scheduled N_2 and T_{T2}.

Engine inlet airflow is scheduled as a function of engine inlet total temperature as shown in figure 9 for the Lockheed Configuration and figure 10 for the Boeing Configuration. Above engine inlet temperatures of approximately 250°F, manual remote adjustment of engine airflow is provided within maximum and minimum airflow limits to permit optimum matching of the engine and inlet.

Engine thrust at cruise conditions is obtained by setting EPR and total engine fuel flow to the required values. These values will be available in the form of tables similar to tables currently being used on subsonic commercial aircraft. Table 1 shows a sample cruise thrust setting table.

Engine thrust is reduced by reducing the power lever position. Moving the power lever into the non-augmented region will shut off duct heater fuel flow at a controlled rate with the duct heater fuel manifold dump valves operating in proper sequential order.

Power lever movement to the idle flat reduces gas generator fuel flow to the deceleration schedule resulting in an engine speed decrease to idle speed and the duct exhaust nozzle being set to the fixed maximum area. As the aircraft descends along the flight path, the reverser-suppressor is reset to the takeoff position aerodynamically. The compressor stator vanes and the compressor bleed valves will automatically be positioned as a function of engine high rotor speed and engine inlet temperature. Checkout of the duct heater igniters is possible during the descent, provided the power lever is in the idle or near idle position.

On approach for landing, engine power is modulated by the pilot to maintain the desired flight path. Reverse power may be selected by retracting the power lever to the reverse idle position. Movement of the clamshell doors to the reverse position permits the power lever to be further retarded to obtain maximum reverse thrust. Ground maneuvering after reverse thrust engine operation will require the power lever to be advanced to the forward idle position. Movement of the clamshell doors to the takeoff position permits the power lever to be advanced to obtain the desired forward thrust.

The engine is normally shutdown by placing the power lever in the forward idle position, and moving the fuel shutoff lever to the "off" position. The ignition selector switches should be placed in the "off" position if the system had been energized for the landing. Shutdown can be effected with the power lever in any position by placing the fuel shutoff lever in the "off" position. This will shutoff fuel flow to both the gas generator and the duct heater.

If duct flameout occurs during duct heater operation, the sudden increase in duct airflow will energize the duct heater blowout mechanism in the control which causes the Zone I and Zone II shutoff valves

to close and opens the respective fuel manifold dump valves. Cooling fuel flow will also be reestablished through the duct heater fuel system components. In this event, the power lever must be retarded to the nonaugmented flat power lever position, or less, and then moved to an augmentation position before duct heater operation can be reinitiated.

In the event an inflight engine shutdown is performed, the engine may be restarted by placing ignition selector switches on the "on" position, the power lever at idle position and moving the shutoff lever to the "on" position. The gas generator ignition selector switches should be placed in the "off" position after engine restart is accomplished. Normal engine control by power lever modulation may then be

Table 1. Sample Cruise Thrust Setting Tables

Subsonic:	36,150 Ft.		M = 0.9
Gross Wt.	OAT - °F	-70	-52
	EPR/EGT	2.26/1370	2.26/1450
480,000	N_2/N_1	94.7/97.2	96.5/100.0
	WFT	10600	10900
	AJD/DH	3.6/Not Lit	3.6/Not Lit
	RAT/TAS	-7/516	+14/528
	EPR/EGT	1.91/1105	1.91/1180
360,000	N_2/N_1	89.6/92.7	91.5/94.7
	WFT	7800	8000
	AJD/DH	4.3/Not Lit	4.3/Not Lit
	RAT/TAS	-7/516	+14/528

Supersonic:	65,000 Ft.		M = 2.7
Gross Wt.	OAT °F	-70	-52
	EPR/EGT	0.743/1425	0.701/1425
500,000	N_2/N_1	100.0/86.2	100.4/86.5
	WFT	26000	27200
	AJD/DH	6.4/Lit	7.0/Lit
	RAT/TAS	494/1548	537/1584
	EPR/EGT	0.743/1425	0.701/1425
370,000	N_2/N_1	100.0/86.2	100.4/86.5
	WFT	19100	20200
	AJD/DH	5.6/Lit	6.2/Lit
	RAT/TAS	494/1548	537/1584

Symbol	Description	Units
OAT	Outside Air Temperature	°F
EPR	Engine Pressure Ratio (Pt7/Pt2)	None
EGT	Turbine Discharge Total Temperature	°F
N_2/N_1	High Rotor RPM/Low Rotor RPM	%
WFT	Total Fuel Flow	PPH
AJD	Duct Nozzle Jet Area	Ft^2
DH	Duct Heater	Lit/Not Lit
RAT	Ram Air Temperature (Tt2)	°F
TAS	True Airspeed	Knots

C. ANALYTICAL MODELING AND CONTROL SYSTEM SIMULATION TESTS

1. Control Mode and Parameter Selection

Prior to and during the Federal Aviation Agency sponsored supersonic transport program, P&WA has conducted detailed computer system analyses to define and evaluate all possible modes of engine control. The main function of all control systems on this engine is to control three basic engine parameters to within allowable limits over the flight envelope during both steady-state and transient operations. These parameters are gas generator turbine inlet temperature, duct heater discharge temperature, and total engine airflow. Total engine airflow must be controlled within narrow tolerances throughout the entire supersonic flight regime of the aircraft, particularly during transients, to minimize engine and aircraft inlet control requirements.

The studies conducted during the early phases of the program selected the following mode of control for a turbofan engine for use in a supersonic transport:

1. Gas generator output to be controlled by scheduling gas generator fuel flow as a function of P_b, N_2, T_{t2}, and PLA.
2. Duct heater output to be controlled by scheduling duct heater fuel flow as a function of P_b, PLA, and T_{t2}.
3. Total engine airflow scheduled as a function of T_{t2} and to be controlled on a closed-loop basis by positioning the duct exhaust nozzle as required to obtain the desired engine airflow utilizing N_2 and fan discharge pressure ratio (PT3-PS3)/PT3.

This mode of control is basically the same as the mode of control that is being used successfully on the J58 engine. Airflow control is obtained by actuating the augmentor exhaust nozzle on both engines. On the J58 engine, the parameter controlled to obtain the desired airflow is engine speed, whereas the parameter used on the JTF17 is duct pressure ratio. Rotor speed could not be used for the controlling parameter on the JTF17 engine, because total airflow is not a direct function of the speed of either rotor.

The JTF17 engine control mode and parameters are described in more detail in paragraph D.

With the basic control concept defined, further computer studies of the basic system were conducted during all phases of the contract. These studies were directed to evaluate the dynamic characteristics of the engine, control system, and the interaction of the engine and inlet.

The results of these computer studies, conducted during the FAA sponsored program have been reported, and are documented in the following reports:

1. P&WA Report No. PWA-2353 and Appendixes A and B cover the work performed under ASD contract No. AF33(657)-11189 BPS No. 3 (6199-648D).
2. P&WA Report No. PWA-2397 and Appendixes A and B cover the work performed under FAA contract No. FA-SS-64-2.
3. P&WA Report No. PWA-2600 and Appendixes A and B cover the work under FAA contract No. FA-SS-65-18.

Over 1000 hours of computer operation was accumulated during these studies.

2. Analytical Modeling and Control System Simulation Test Results in Phase 11-C

During Phase 11-C, the computer analysis of the system was continued by P&WA and the two control vendors. Studies at P&WA involved simulating the engine, the control system, and aircraft inlet at several operating conditions to evaluate vendor proposed systems and different approaches or methods of solution to the control problems.

The main results of these studies were to (1) define the activation schedule for the compressor bleeds, (2) indicate the required interlocks and permission signals required on the fast-fill system of the duct heater and define manifold filling times, (3) define interlocks and response requirements of the reverser system, (4) define input and removal rates of resets in the duct burner and nozzle area control, and (5) determine the response of the airframe inlet to engine transients. Additional studies covering inlet compatibility are discussed in Report D, Section II, paragraph C.

Some of the results of these studies are shown in figures 1 through 6. These computer traces show the engine operating at sea level and cruise, and indicate the engine transient performance in response to power lever motions.

Figure 1 shows the response of the engine at sea level to a throttle advance from idle to maximum augmentation. The engine accelerates from idle, and when the high pressure compressor rotor reaches 80% rpm, a signal is provided to initiate the fast-filling sequence of the zone I manifold of the duct burner. When the manifold is full, this occurring in approximately 1 second, a signal stops the fast-fill sequence and ignition occurs.

When zone I ignition has occurred, the fast-fill sequence of zone II manifold is initiated. This sequence is terminated when filling is complete. This method of sequencing the filling on zone II manifold provides smooth transition from zone I to zone II with essentially no time delays. After the hold for zone I fast-fill sequence, the engine can advance through the augmentation region in approximately 1.1 second. The total time of 8 seconds is necessary to advance from idle to maximum thrust, and 6.5 seconds to reach 95% of maximum thrust. The augmentation initiation sequence is described in detail in the unitized fuel and area control description in paragraph D.

Figure 2 presents the response of the engine for a "wave-off" during landing approach. The engine is operating at the approximate approach thrust when the power lever is snapped to the maximum augmentation position. The engine takes approximately 4 seconds to reach maximum thrust.

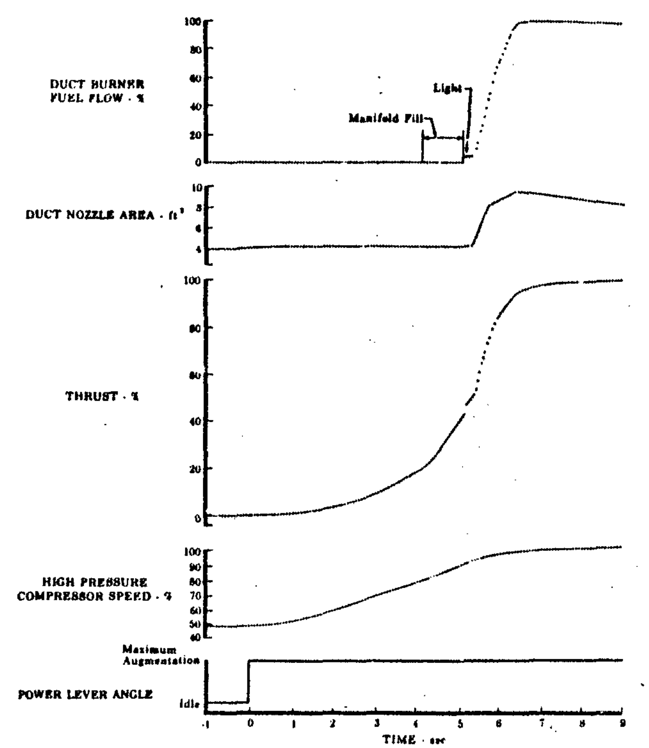


Figure 1. JTF17 Digital Simulation Engine Response to PLA Modulations from Idle to Maximum Augmentation at Sea Level Conditions

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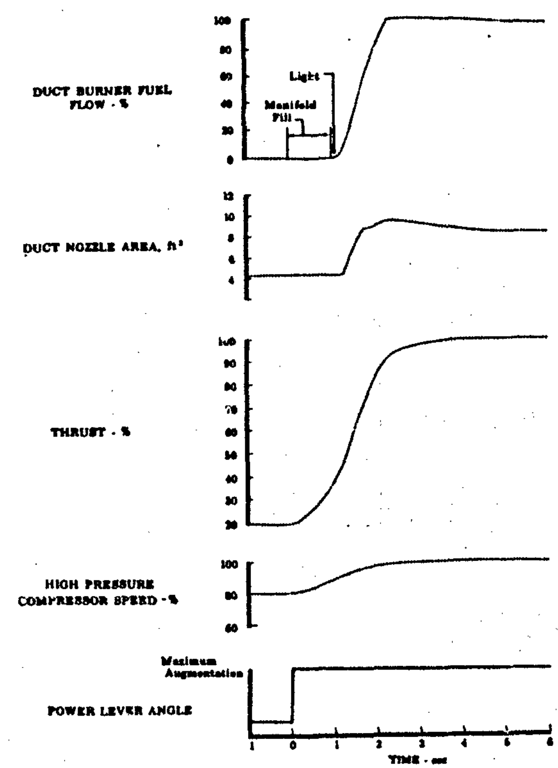

Figure 2. JTF17 Digital Simulation Engine Response to PLA Modulation from Part Power to Maximum Augmentation Sea Level Conditions

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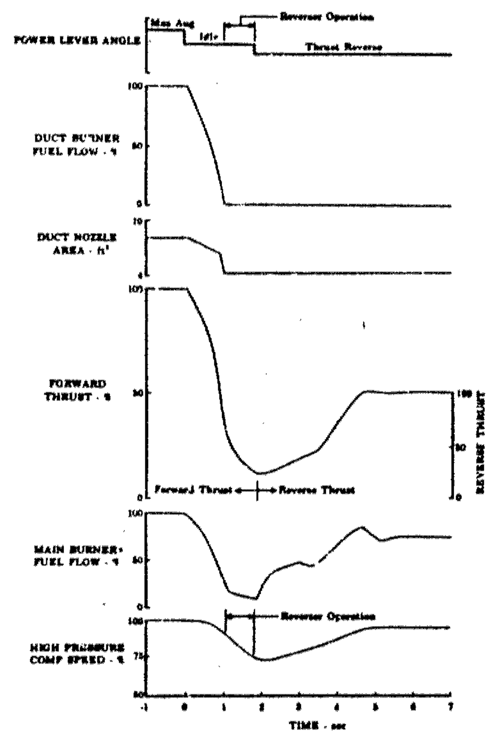

Figure 3. JTF17 Digital Simulation Engine Response to PLA Modulation from Maximum Augmented to Thrust Reverse Sea Level Conditions FD 15841 BIII

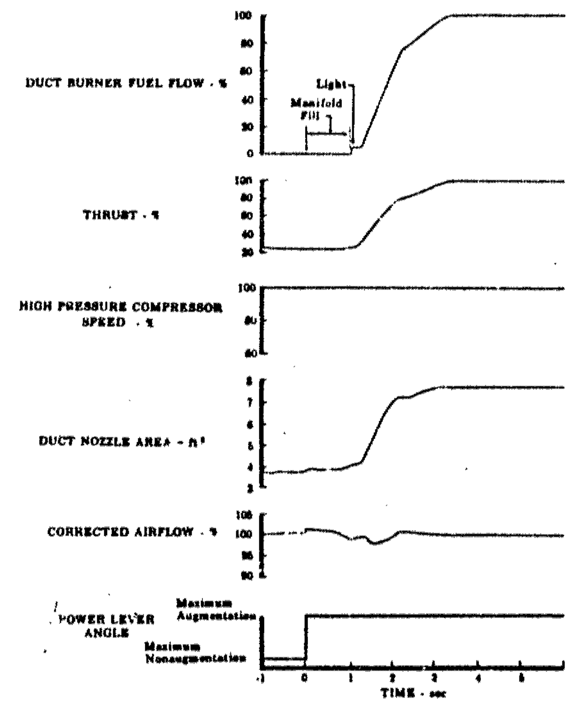

Figure 4. JTF17 Digital Simulation Engine Response to PLA Modulation from Maximum Nonaugmented to Maximum Augmentation at Cruise Conditions FD 15626A BIII

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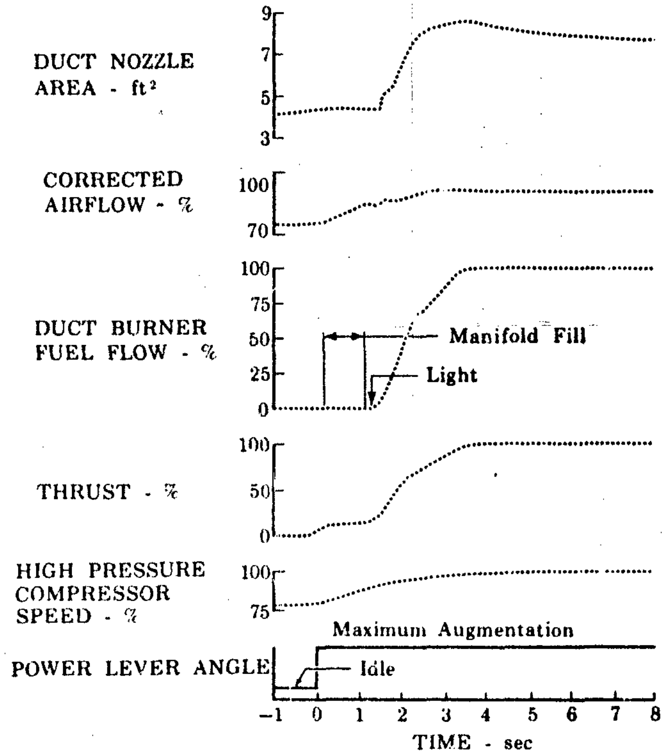


Figure 5. JTF17 Digital Simulation Engine Response to PLA Modulation from Idle to Maximum Augmentation at Cruise Conditions FD 15624A BIII

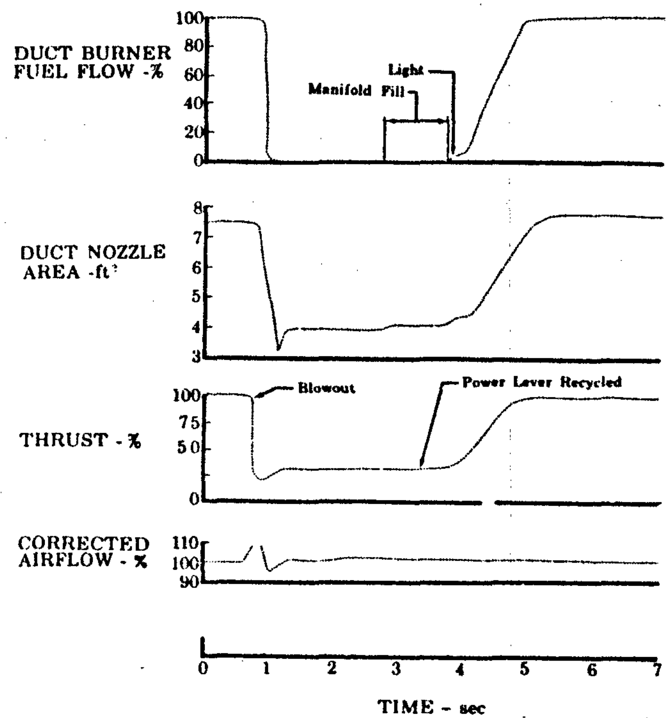


Figure 6. JTF17 Digital Simulation Duct Heater Blowout and Relight at Cruise FD 17144 BIII

Sea Level operation from maximum augmentation to reverse thrust is shown in figure 3. The engine is operating at maximum thrust when the power lever is snapped to reverse idle, at which point it must stay until two interlocking functions are performed at which time further motion to thrust reverse is permitted. When the high-pressure compressor rotor speed decelerates to 90% rpm, a signal is provided that starts the thrust reverser clamshell doors moving to the reverse thrust position. This motion requires approximately 0.75 seconds. Once the clamshells are in the reverse thrust position, the power lever can be moved to reverse thrust position. The transient from maximum augmentation to maximum reverse thrust requires approximately 4.75 seconds.

Figure 4 shows engine operation with a simulated aircraft inlet system for a transient from maximum nonaugmented to maximum augmented at cruise conditions of Mach 2.7 and 65,000 feet altitude. For this study, the engine is operating at the maximum nonaugmented condition and the power lever is snapped to the maximum augmented position. The fast-filling sequence of Zone I is begun immediately. The duct nozzle area is reset open in anticipation of the duct light, which provides additional fan and inlet stability margins during the duct heater light-off. The area reset is removed slowly after the engine has advanced well into the augmentation region. After the ignition of Zone 1, the operation through to maximum augmentation is the same as described in the sea level idle to maximum augmentation transient. Approximately 4 seconds is required for the transient from maximum nonaugmented to maximum augmented.

Figure 5 shows engine transient operation with a simulated aircraft inlet system from idle to maximum augmentation at cruise conditions of Mach 2.7 and 65,000 feet altitude. The engine operational procedure from idle to maximum augmentation is the same as the sea level idle to maximum augmentation operation. The duct nozzle area is reset open in anticipation of the duct heater light-off in order to provide additional fan and inlet stability margins.

A simulation of a duct heater blowout and relight at cruise conditions of Mach 2.7 and 65,000 feet altitude is shown in figure 6. When the duct heater blowout is encountered, the airflow increases and, at a level of approximately 4-1/2% above the desired airflow, the airflow control shuts off the duct fuel flow and opens the Zone I and II manifold dump valves. A signal is also sent to the gas generator fuel control to reduce fuel flow and, thus, reduce any possibilty of fan overspeed. The power lever must be recycled to the maximum nonaugmented position before augmented operation can be attained. The computer simulation includes an airframe air induction system.

In addition to these data, other results from both analog and digital computer studies conducted by both P&WA and the two control vendors are reported in the monthly progress reports prepared as a part of FAA Contract No. FA-SS-66-8.

Progress Report No.	Date	Report No.
1	8-10-65	FR-1492
2	9-10-65	FR-1540
3	10- 8-65	FR-1598
4	11-10-65	FR-1643
5	12-10-65	FR-1691
6	1-10-66	FR-1708
7	2-10-66	FR-1749
8	3-10-66	FR-1779
9	4-10-66	FR-1825
10	5-10-66	FR-1855
11	6-10-66	FR-1884
12	7-10-66	FR-1953

The functional block diagram used for simulating the JTF17 engine and control system on the digital computer is shown in figure 7, and the analog schematic diagrams for simulating the same system at cruise conditions are shown in figures 8 and 9. Both systems include simulation of the air induction system.

3. Definition and Description of Interface Connections Between the Engine Control System and the Air Induction Control System

During initial analytical and computer studies of the engine control-inlet control compatibility studies, it appeared that interconnections would be required between the two control systems. These interconnections were: a Mach number signal from the air inlet control system to the engine control system; and a reset signal from the engine control to the inlet control.

Theoretically, Mach number scheduling of engine airflow is very desirable for inlet airflow matching. However, a realistic evaluation of the combined effect of engine-to-engine and inlet-to-inlet airflow variation plus the engine airflow variation induced by precision errors in the Mach number signal lead to propulsion performance penalties that may be unacceptable.

A new scheduling concept has been coordinated with the airframe manufacturers. Engine airflow is scheduled by engine inlet total temperature with a flight deck adjustment available to improve inlet matching. This method of scheduling cruise airflow takes advantage of the versatility available with the turbofan cycle to provide excellent engine/inlet compatibility.

Elimination of the Mach number bias removed the need for an instrumentation link between the inlet control and the engine control.

Initial evaluation of inlet stability margins indicated that rapid airflow transients of greater than -2% could not be tolerated by the inlet during inlet started operation. Therefore, to provide stability margin, a reset signal was transmitted to the inlet control so that when the duct heater was lit or when Zone II operation was initiated, the

shock position would be relocated to a more stable location. Lockheed and Boeing have since advised that the inlet stability margins have been greatly increased to the extent that the duct heater transients are now accomplished within these margins. Therefore the inlet reset was eliminated.

As a result of these studies it has been concluded that interfaces between the inlet control system and the engine control system are not required on the JTF17 engine installation in the Boeing or the Lockheed airplane. Figure 6 shows the response of the engine to an augmentor light-off at Mach 2.7 and 65,000 feet without any interfaces between the engine and inlet control systems.

4. Plans for Combined Air Industion System, Engine, and Control Simulation Tests in Phase III.

During Phase III, the computer simulation and analysis of the engine system will be continued by both P&WA and the vendor designing and manufacturing the unitized fuel and area control. The computer work to be conducted by P&WA will emphasize control system compatibility with the engine and the compatibility of the engine with the inlet. The vendor program will be concentrated on determining the gains, response time, and constants required within the control in order for the unit to provide the necessary control of the engine.

The major portion of the study to be conducted by P&WA will be directed toward the analysis of the complete airframe-engine system to ensure that necessary system compatibility is attained. This will involve close coordination of the engine and airframe inlet control design programs. To aid in this analysis the airframe manufacturers have been supplied a mathematical computer model of the engine and its control system. This model will provide the airframe inlet control designer a tool with which he can evaluate the engine inlet interaction. Similarly, a complete computer simulation of the inlet, provided by the airframe manufacturer, will be used to evaluate engine control designs and operation in the airframe environment at P&WA.

To refine the design of the control system for the prototype engine the control schedules will be further evaluated to define steady-state operation and acceleration schedules that will give the fastest acceleration times.

A detailed simulation of each major component of the control system will also be constructed to determine the adequacy of the design before reaching the testing phase. As testing continues, the model will be revised to reflect the central characteristics determined from bench and engine testing. This component simulation will be in sufficient detail to provide a tool with which to do malfunction analysis on an individual part, such as springs, levers, etc. This will then provide information for the malfunction predictions.

The computer program at P&WA will also verify the results obtained from system studies performed by the unitized fuel and area control vendor. This verification is necessary since much of the detail design will be based on requirements that are generated by the vendor study. Thus, if a discrepancy is found in the overall system study, the necessary corrective action will be taken at the earliest opportunity.

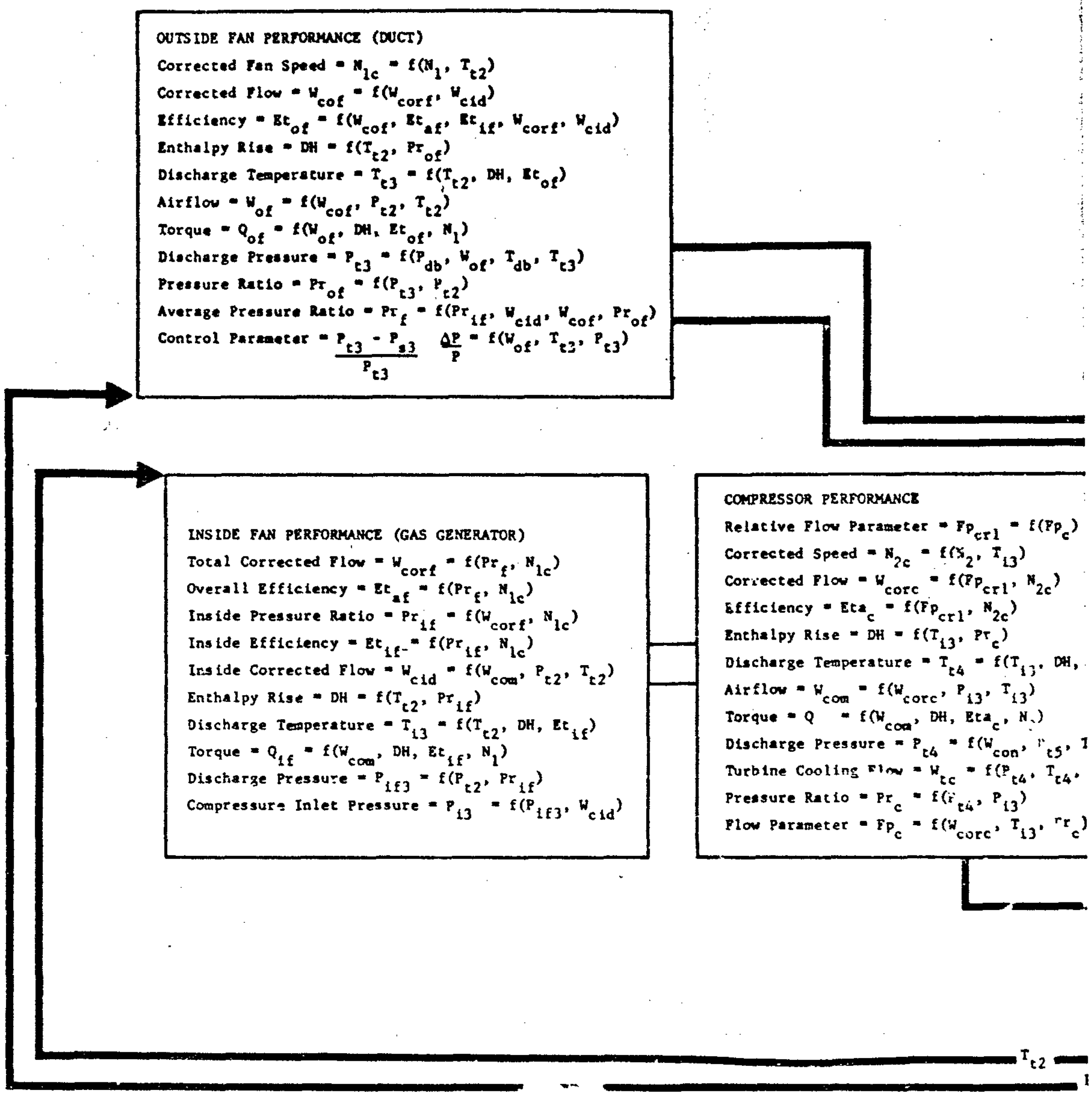
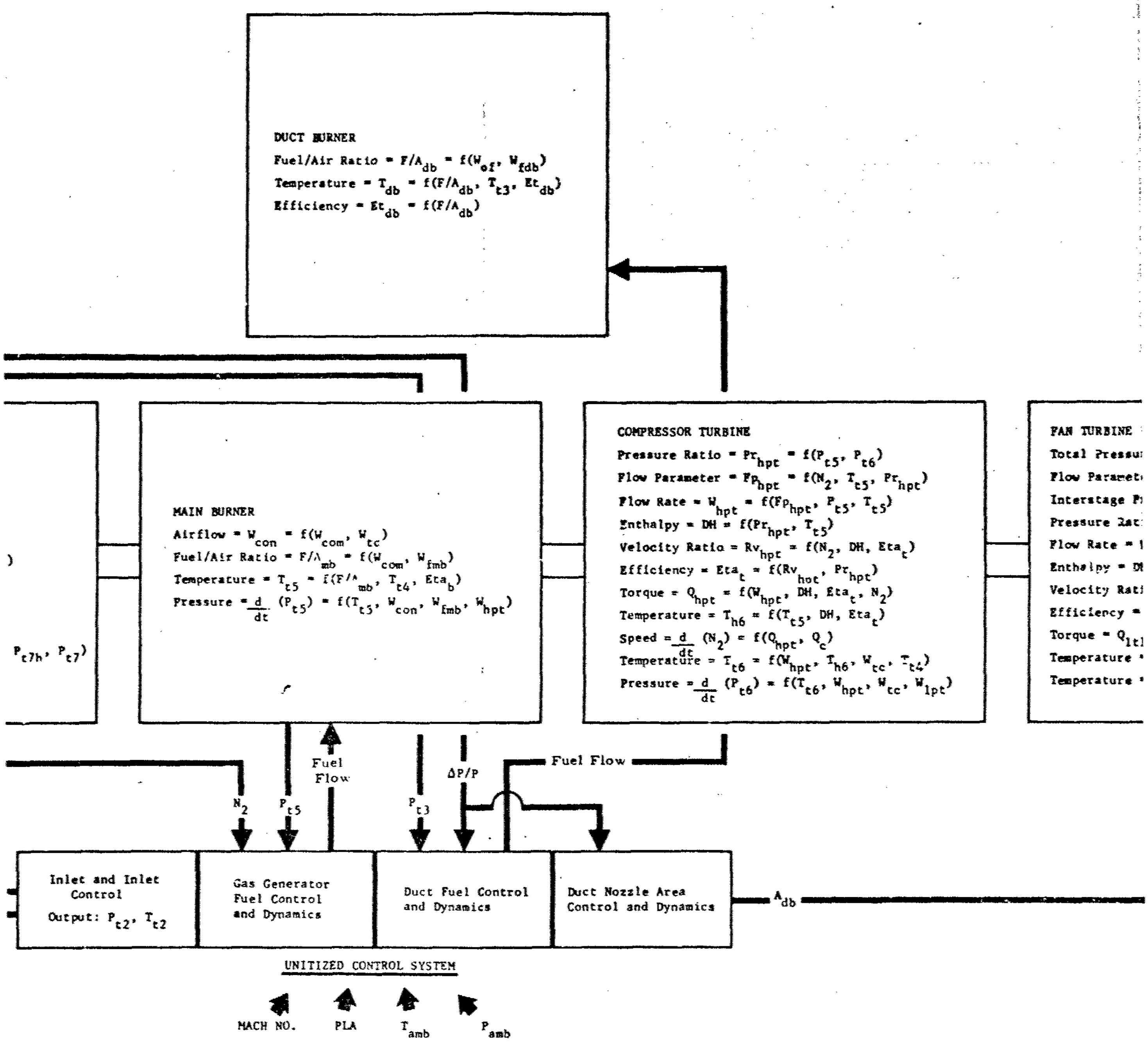


Figure 7. Functional Block Diagram Used for Simulating JTF17 Engine and Control



tem on Digital Computer

2

T STAGE)

tio = $Pr_{1pt} = f(P_{t6}, P_{t7})$

$Fp_{1pt} = f(T_{t6}, Pr_{1pt}, N_1)$

re = $P_{t7h} = f(P_{t6}, P_{t7})$

st Stage) = $Pr_{11t} = f(P_{t6}, P_{t7h})$

$= f(Fp_{1pt}, P_{t6}, T_{t6})$

$(Pr_{1t} \cdot T_{t6})$

$Rv_{1t1} = f(N_1, DH, Et_{11t})$

t $= f(Rv_{1t1})$

$(W_{1pt}, DH, Et_{11t}, N_1)$

$= f(T_{t6}, DH, Et_{1t})$

$= f(W_{1pt}, T_{7h}, W_{tc}, T_{t4})$

FAN TURBINE (SECOND STAGE)

Flow = $W_{12t} = f(W_{1pt}, W_{tc})$

Pressure Ratio = $Pr_{21t} = f(P_{t7h}, P_{t7})$

Enthalpy = $DH = f(Pr_{21t}, T_{71})$

Velocity Ratio = $Rv_{1t2} = f(N_1, DH, Et_{21t})$

Efficiency = $Et_{21t} = f(Rv_{1t2})$

Torque = $Q_{1t2} = f(W_{12t}, DH, Et_{21t}, N_1)$

Temperature = $T_{t7} = f(T_{71}, DH, Et_{21t})$

Temperature = $T_7 = f(T_{t4}, W_{tc}, W_{12t}, T_{t7})$

Flow = $W_7 = f(W_{12t}, W_{tc})$

Pressure = $P_{t7} = f(W_7, T_7, P_{t8})$

Speed = $\frac{d}{dt}(N_1) = f(Q_{1t1}, Q_{1t2}, Q_{of}, Q_{if})$

DUCT NOZZLE

Pressure = $\frac{d}{dt}(P_{db}) = f(T_{db}, W_{of}, W_{fdb}, W_{exd})$

Flow = $W_{exd} = f(T_{db}, P_{db}, A_{db}, P_{amb})$

Total Fuel/Air Ratio = $F/A_{dtt} = f(W_{of}, W_{fdb})$

Enthalpy Change = $DH = f(T_{db}, P_{db}, P_{amb})$

Gas Velocity = $V_{jdb} = f(DH)$

Specific Thrust = $T/Wa_{db} = f(F/A_{dtt}, V_{jdb}$, Aircraft Velocity

Duct Thrust = $F_{db} = f(T/Wa_{db}, W_{exd})$

$TSFC_d = f(F/A_{dtt}, T/Wa_{db})$

NOZZLE

Pressure = $\frac{d}{dt}(P_{t8}) = f(T_7, W_7, W_{ex})$

Flow = $W_{ex} = f(T_7, P_{t8}, P_{am})$

Total Fuel/Air Ratio = $F/A_{tgg} = f(W_{com}, W_{fmb})$

Enthalpy Change = $DH = f(T_7, P_{t8}, P_{am})$

Gas Velocity = $V_{jn} = f(DH)$

Specific Inrust = $T/Wa = f(F/A_{tgg}, V_{jn}$, Aircraft Velocity)

Gas Generator Thrust = $F_n = f(T/Wa, W_{ex})$

TSFCG = $f(F/A_{tgg}, T/Wa)$

Total Thrust = $T_e = f(F_n, F_{db})$

Total Fuel Flow = $W_{feng} = f(W_{fmb}, W_{fdb})$

Total TSFC = $f(W_{feng}, T_e)$

DUCT NOZZLE

Pressure = $\frac{d}{dt}(P_{db}) = f(T_{db}, \dot{W}_{of}, W_{fdb}, W_{exd})$

Flow = $W_{exd} = f(T_{db}, P_{db}, A_{db}, P_{amb})$

Total Fuel/Air Ratio = $F/A_{dtt} = f(W_{of}, W_{fdb})$

Enthalpy Change = $DH = f(T_{db}, P_{db}, P_{amb})$

Gas Velocity = $V_{jdb} = f(DH)$

Specific Thrust = $T/Wa_{db} = f(F/A_{dtt}, V_{jdb}, \text{Aircraft Velocity})$

Duct Thrust = $F_{db} = f(T/Wa_{db}, W_{exd})$

$TSFC_d = f(F/A_{dtt}, T/Wa_{db})$

THRUST

FUEL CONSUMPTION

$1t)$

$t7)$

$1f)$

NOZZLE

Pressure = $\frac{d}{dt}(P_{t8}) = f(T_7, W_7, W_{ex})$

Flow = $W_{ex} = f(T_7, P_{t8}, P_{am})$

Total Fuel/Air Ratio = $F/A_{tgg} = f(W_{com}, W_{fmb})$

Enthalpy Charge = $DH = f(T_7, P_{t8}, P_{am})$

Gas Velocity = $V_{jn} = f(DH)$

Specific Thrust = $T/Wa = f(F/A_{tgg}, V_{jn}, \text{Aircraft Velocity})$

Gas Generator Thrust = $F_n = f(T/Wa, W_{ex})$

$TSFCG = f(F/A_{tgg}, T/Wa)$

Total Thrust = $T_e = f(F_n, F_{db})$

Total Fuel Flow = $W_{feng} = f(W_{fmb}, W_{fdb})$

Total TSFC = $f(W_{feng}, T_e)$

THRUST

FUEL CONSUMPTION

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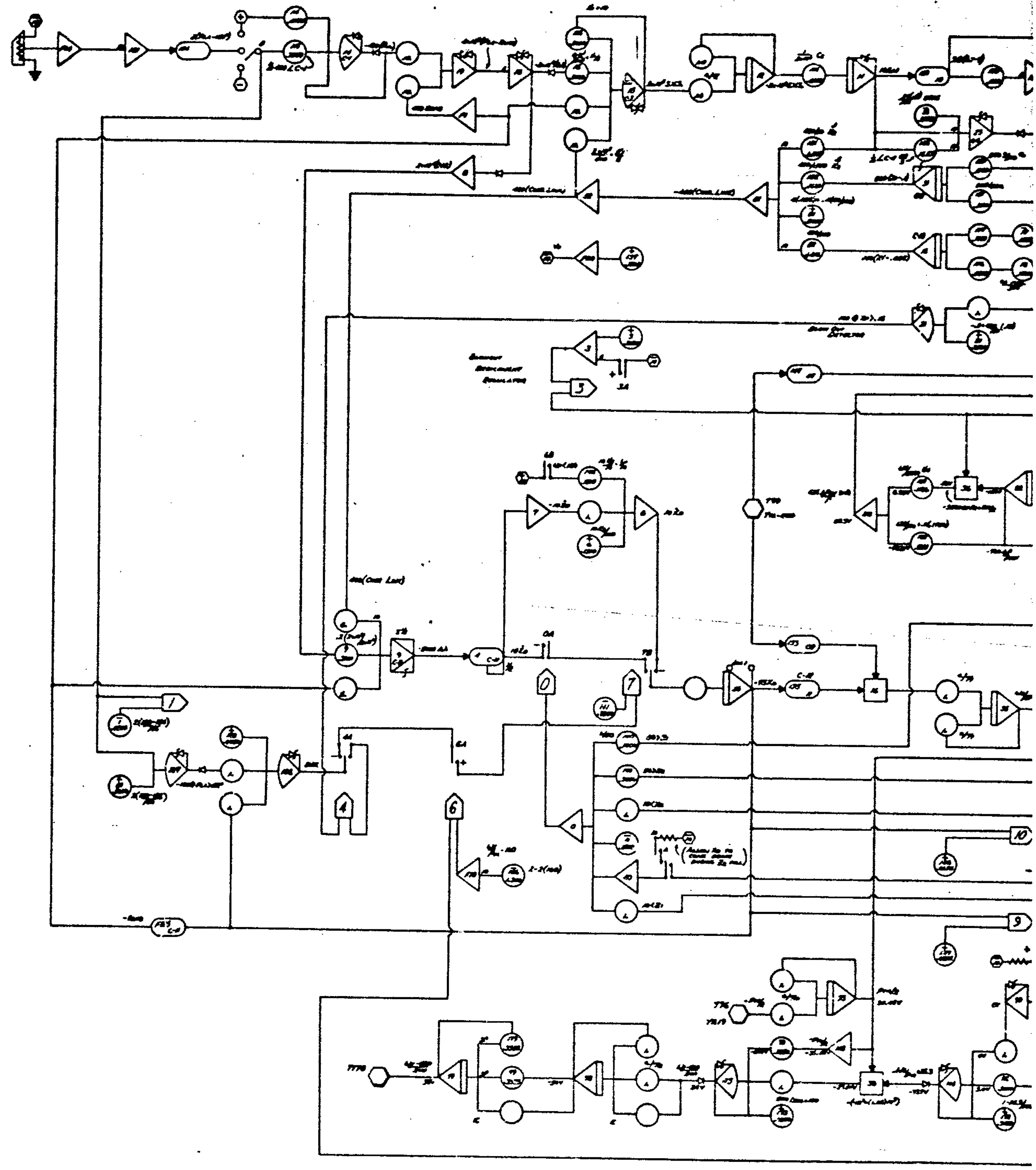
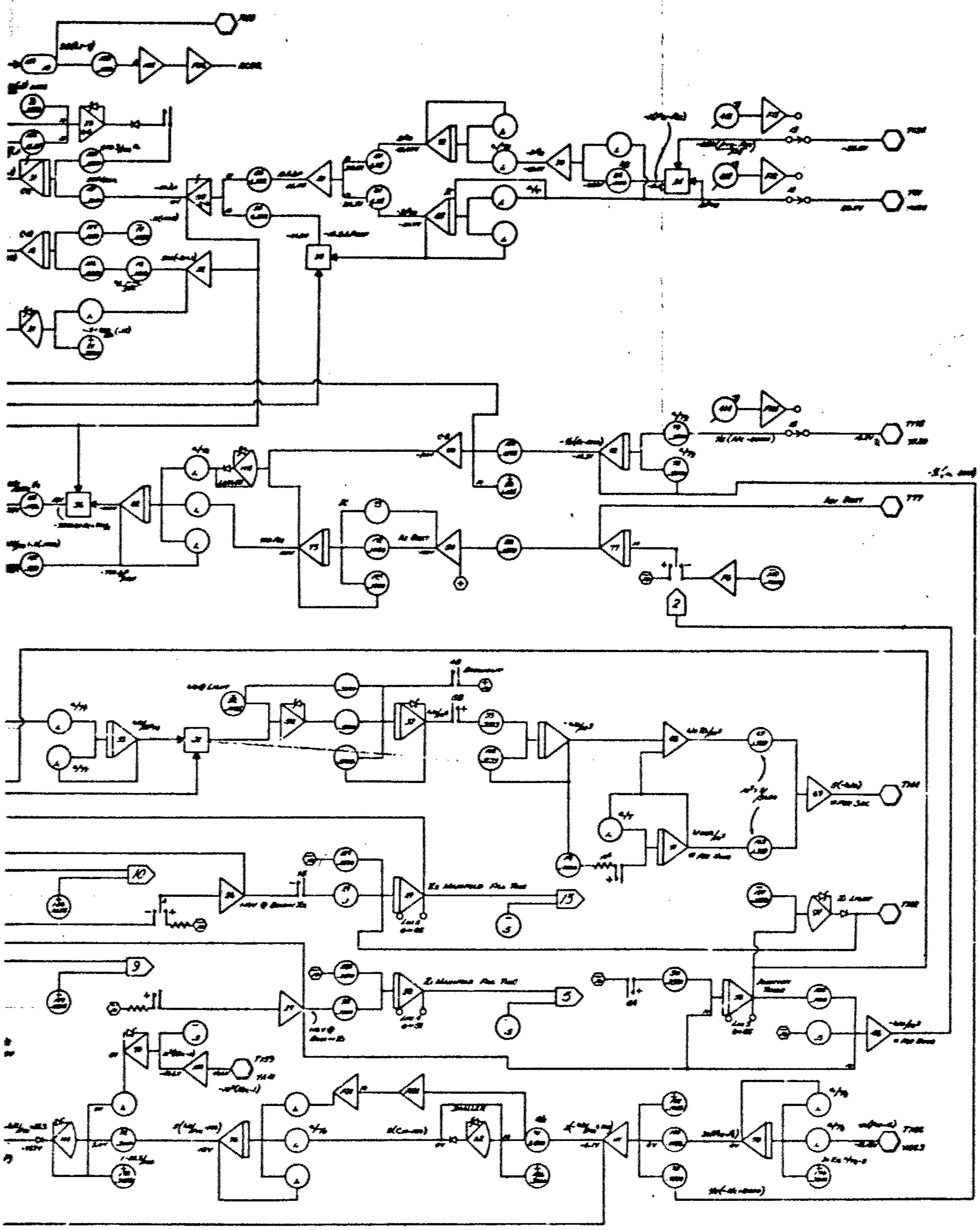


Figure 8. Analog Schematic Diagram for Simulating Aircraft Inlet and JTF17 Engine at Cruise Cond



Cruise Conditions

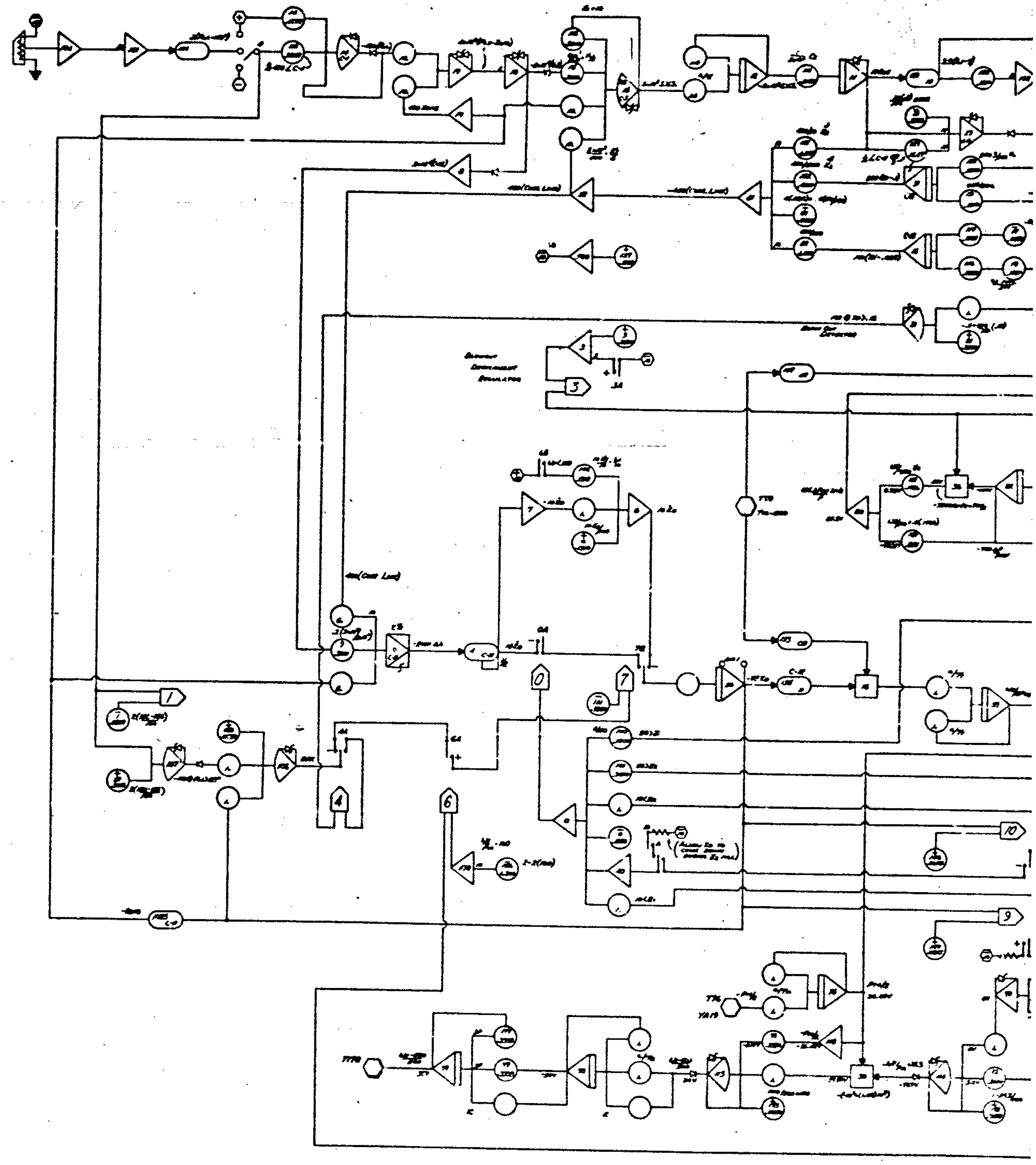
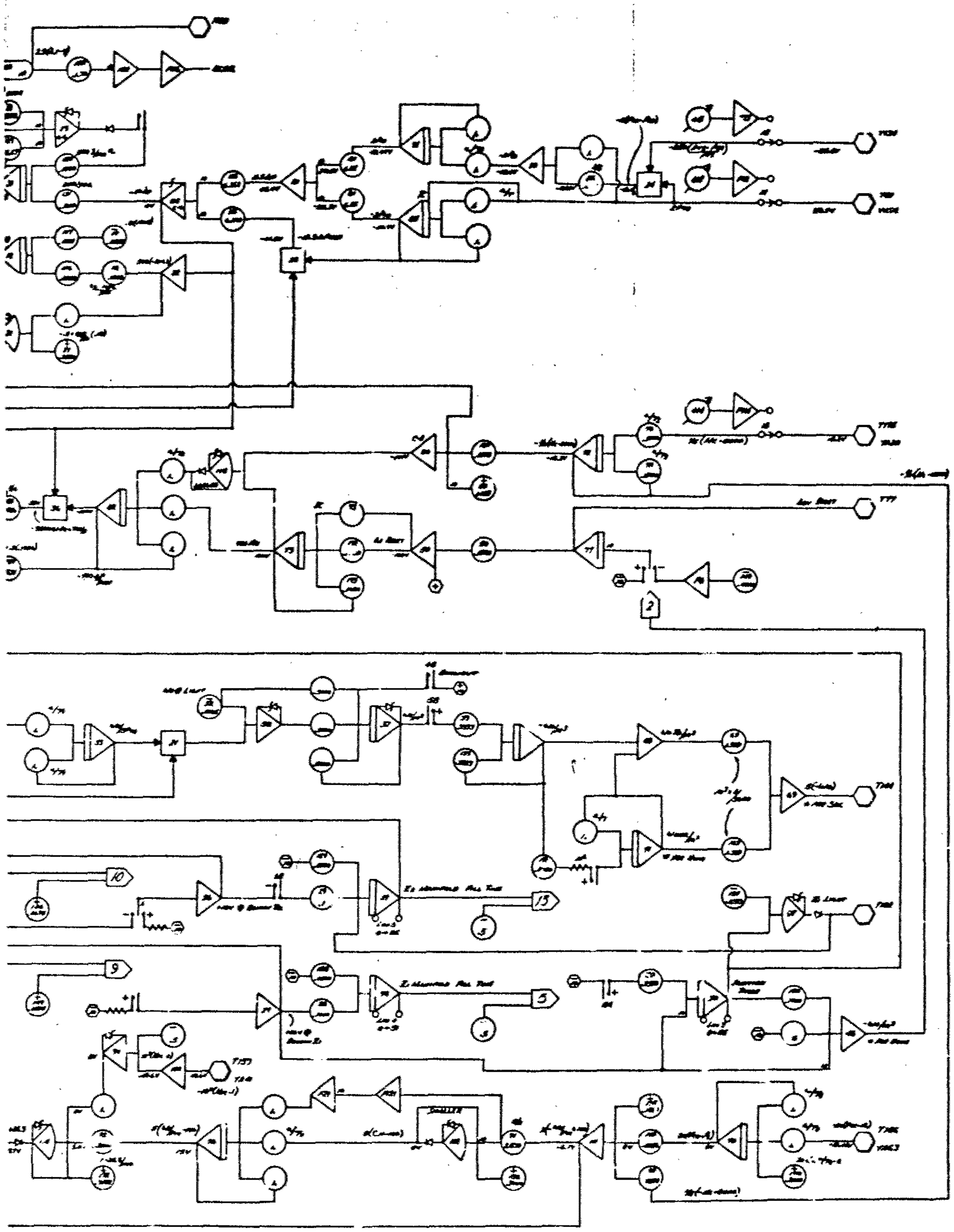


Figure 8. Analog Schematic Diagram for Simulating Aircraft Inlet and JTF17 Engine at Cruise Condi



ruise Conditions

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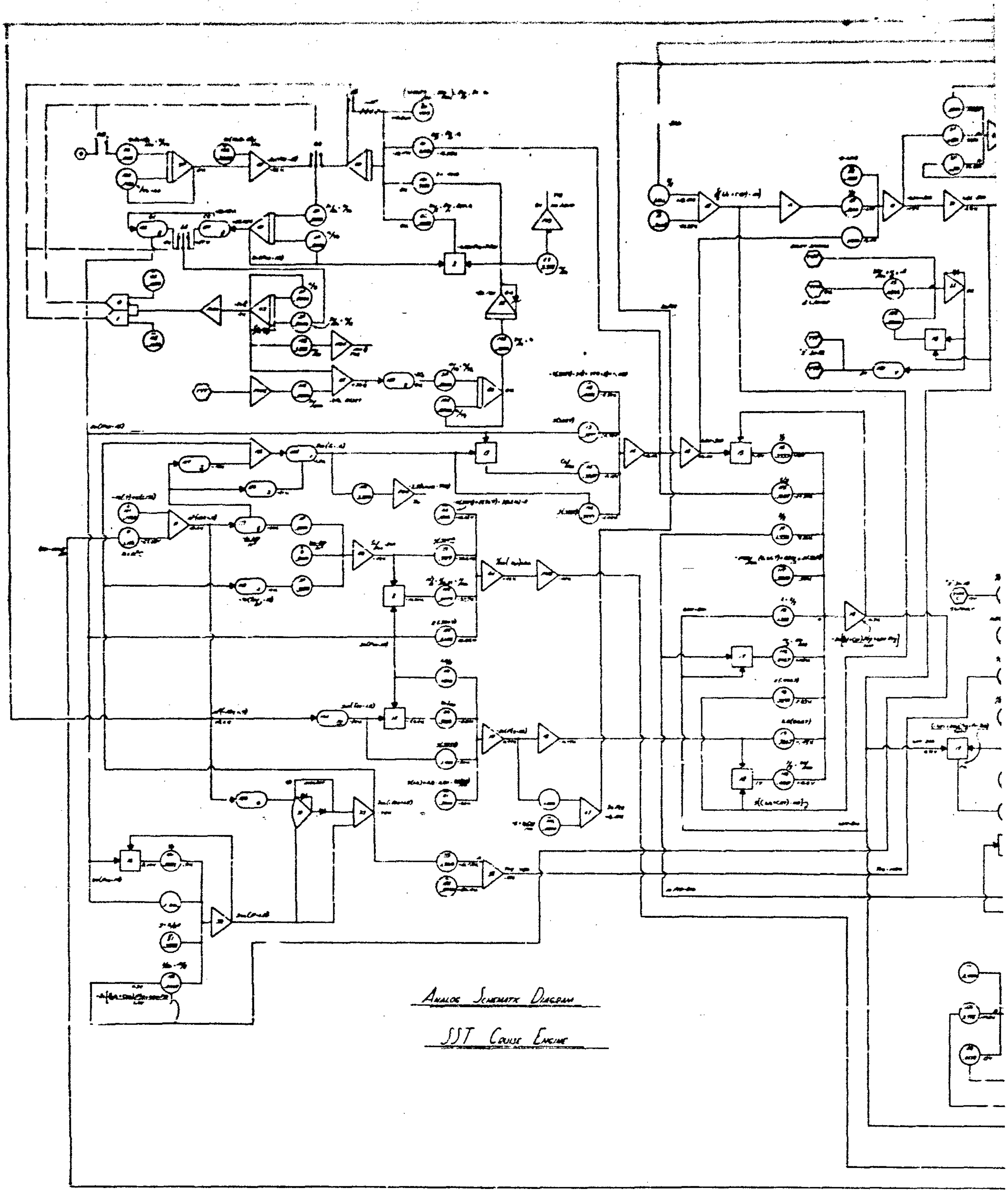
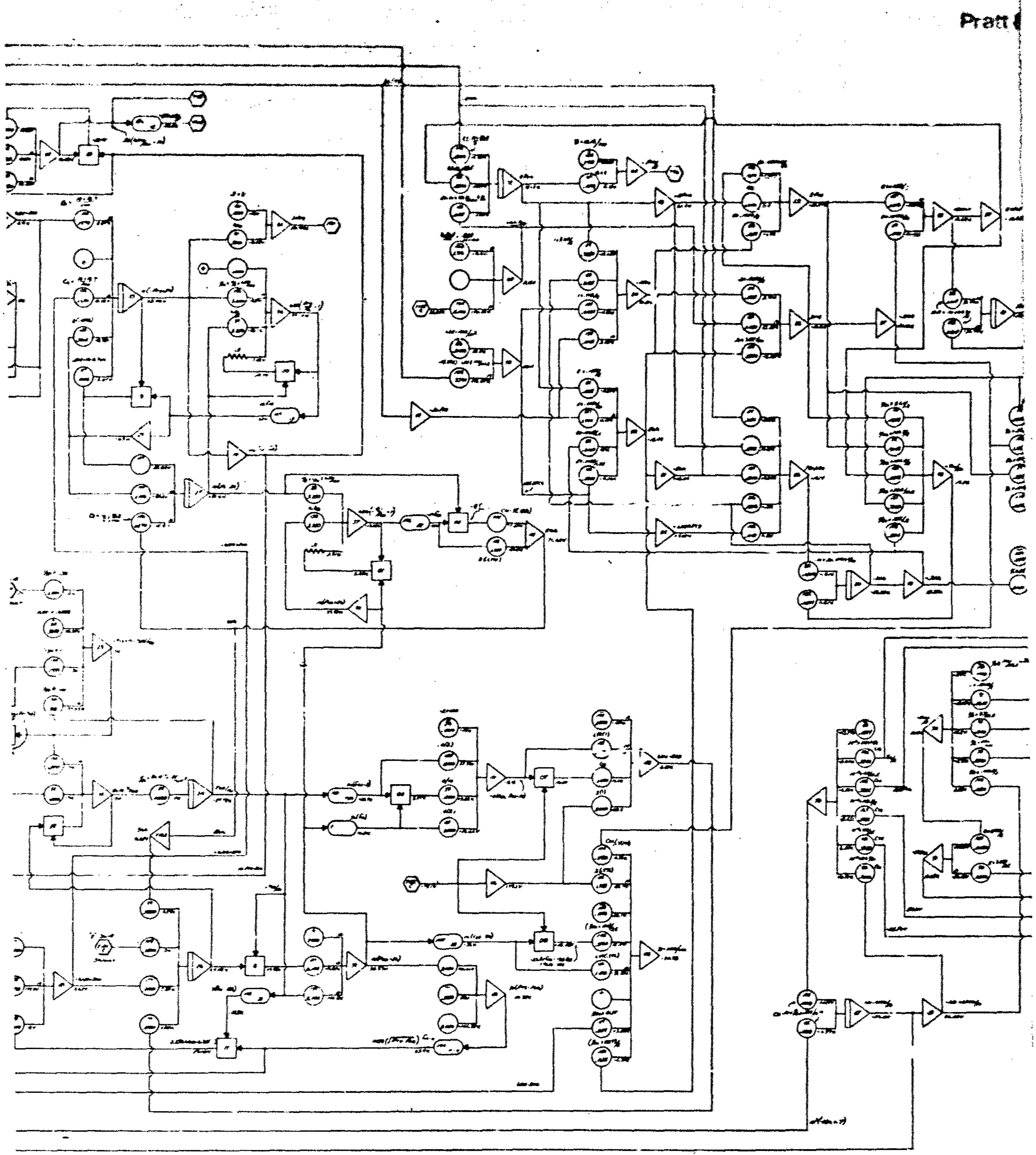
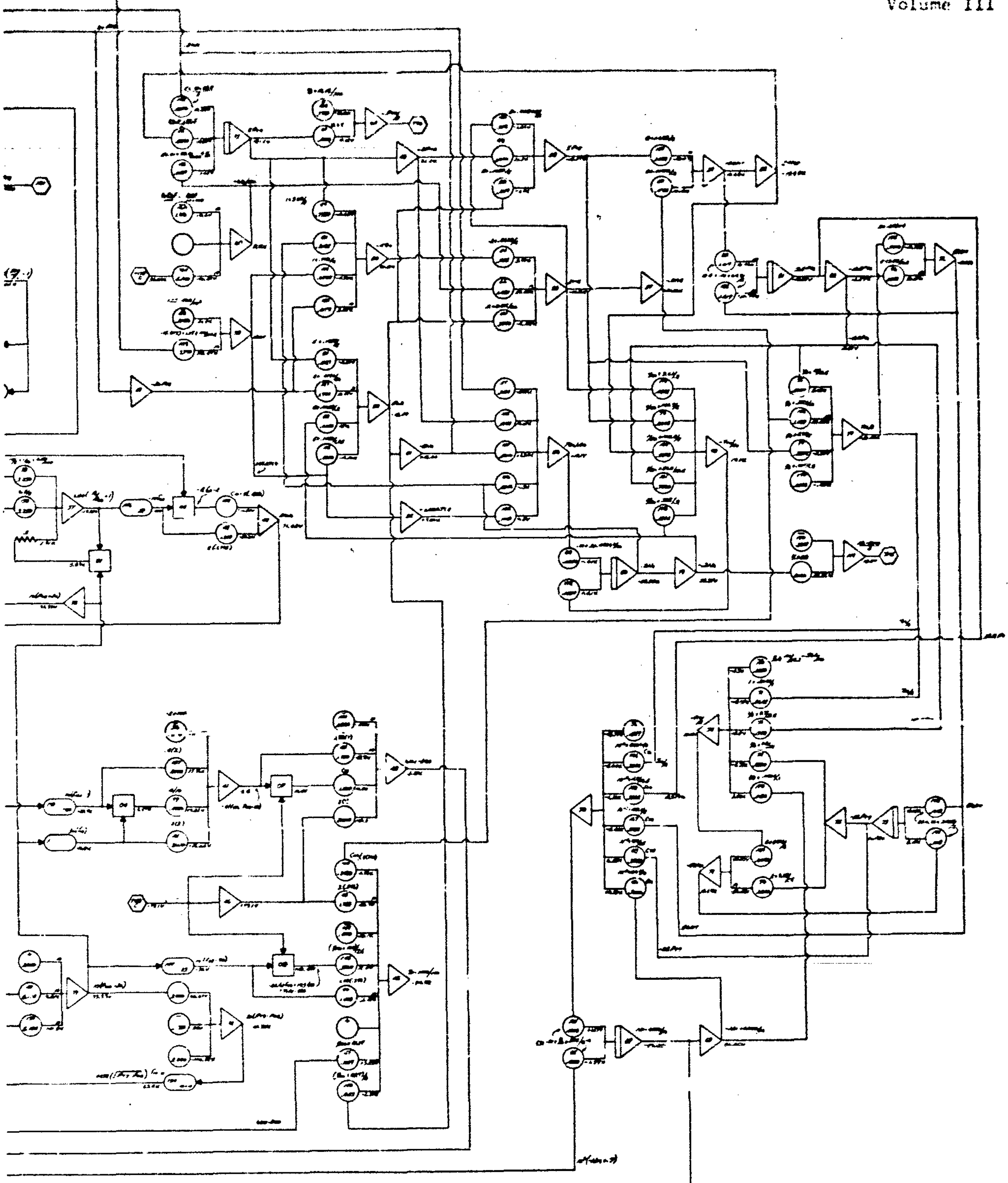


Figure 9. Analog Schematic Diagram for Simulating the Unitized Fuel and Area Control of



of the JTF17 at Cruise Conditions

2



.tions

?. Computer Facilities

Extensive computer facilities are available to P&WA for use in the investigation of SST control modes and the development of proper design within the desired control mode. A complete description of these facilities is presented in Volume V, Section II.

Section II, paragraph C also presents results of computer studies performed investigating control of the JTF17 engine and compatibility of the engine with the airframe inlet.

a. FRDC Computer Capabilities

Computers, such as Beckman Model 2133 analog and IBM 360 system Model 65, are available at FRDC and can simulate a complete JTF17 engine with supersonic inlet. Simulation of engine operation throughout the flight envelope, including engine inflight shutdown and engine windmilling and the aerodynamic brake on and off, can be made.

To illustrate the extent of a typical simulation, the control system is programed to predict the performance of each individual servo and pump, and the characteristics of each engine component as well as performance of the airframe inlet. Through a detailed analysis of each run, the control or pumps may be revised to produce more desirable systems results. Simultaneous operation of both analog and digital computer representation is available. A photograph of the FRDC analog computer control panel is shown in figure 10.

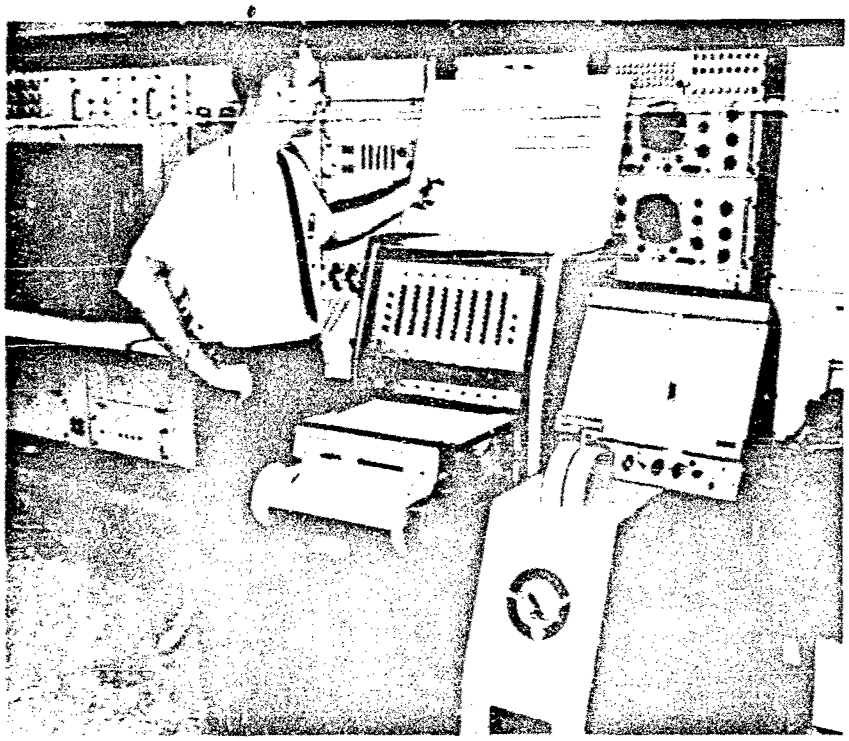


Figure 10. P&WA FRDC Analog Facility FE 12204 BIII

b. UARL Computer Facilities

In addition to the FRDC facilities, hybrid computer DEC Models PDP-1 and PDP-6 are available at United Aircraft Research Laboratories. UARL also have Beckman Model 1100 Analog facilities and are procuring Univac 1108 digital facilities.

c. Hamilton Standard Computer Facilities

Hamilton Standard computer facilities include an IBM Model 1620 digital computer and a Philbrick analog computer. The equipment is capable of complete control system simulation which includes engine pumps, individual control sensors, and a simplified engine. A photograph of the Hamilton Standard analog control panel is presented in figure 11.

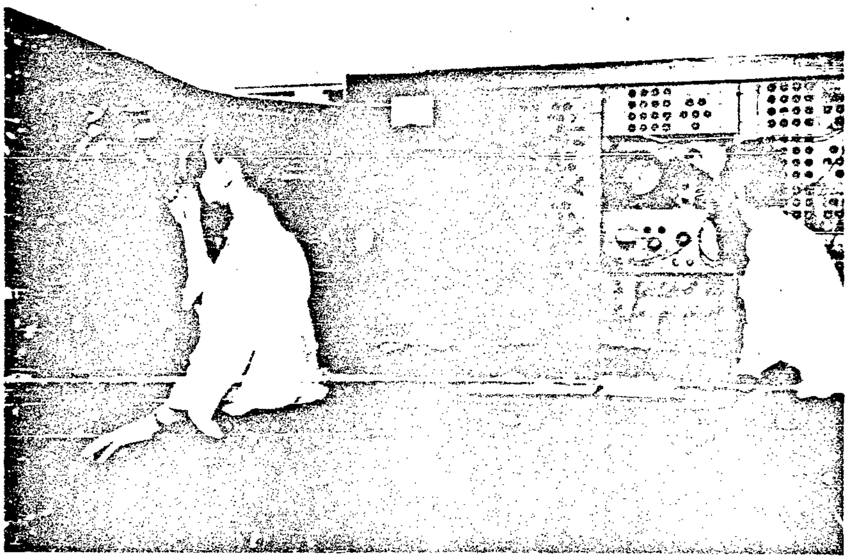


Figure 11. Hamilton Standard Analog Facility FE 56163 BIII

d. Bendix Computer Facilities

Bendix computer facilities include both IBM and CDC digital units and several EAI analog units. Univac 1107 equipment is readily available on a central basis within the city of South Bend and has been utilized by Bendix for other programs. A photograph of the Bendix analog facility is presented in figure 12.

In addition to their own facilities, each vendor has access to the UARL computer facilities.

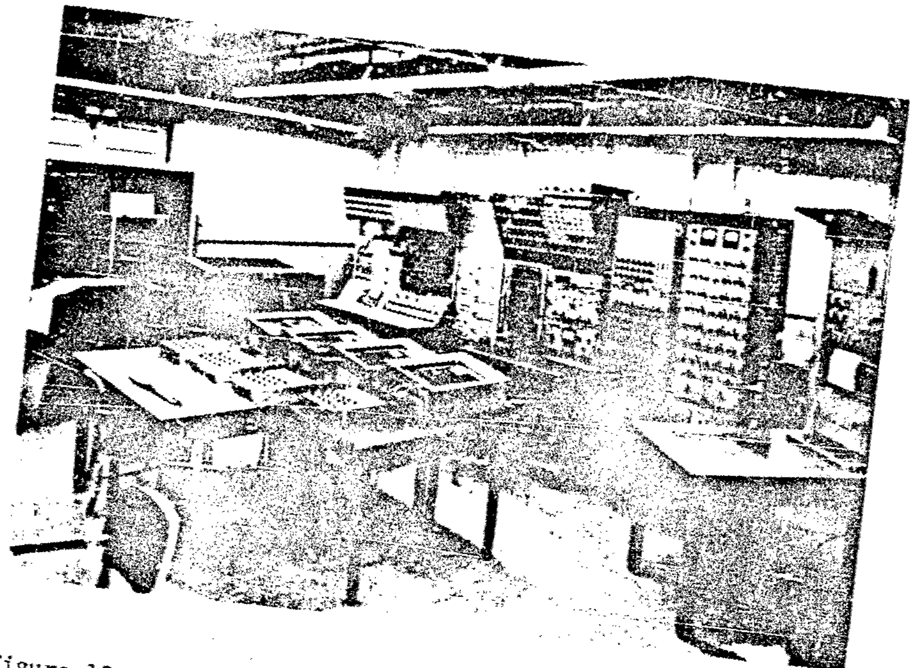


Figure 12. Bendix Analog Facility

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D. UNITIZED FUEL AND AREA CONTROL

1. Description of Unitized Fuel and Area Control

a. Introduction

The unitized fuel and area control is the major component of the control system for the JTF17 engine and incorporates all of the primary and most of the secondary controlling functions of the system. The unit has been the subject of analytical studies and design concepts by Pratt & Whitney Aircraft, Hamilton Standard Division, and Bendix Products Aerospace Division for the past three years. The result of this effort has been the definition of a control which will be capable of providing the optimum engine control over the operating envelope of the engine.

b. General Description of Unitized Fuel and Area Control

The proposed prototype JTF17 engine utilizes a hydromechanical power control system which is the same type of hardware that has provided successful service operation of previous P&WA commercial turbine engines.

The JTF17 engine control has the following basic functions:

1. Controls engine speed, turbine inlet temperature, and engine thrust between full reverse and maximum duct augmentation power as a function of PLA
2. Schedules gas generator fuel flow rates during acceleration or deceleration to keep engine operating conditions within acceptable limits during transient operations
3. Positions the duct heater exhaust nozzle area to maintain the desired corrected total engine airflow schedule
4. Positions the compressor inlet stator vanes as a function of compressor inlet temperature and high compressor speed
5. Positions the compressor bleeds as a function of high rotor speed and engine inlet temperature
6. Positions the thrust reverser-suppressor as a function of power lever angle
7. Provides for fuel cutoff at engine shutdown
8. Controls the speed of the duct heater fuel pump to the minimum required to provide duct fuel pressure and flow.

Typical schedules are shown on figures 1 through 6. Detail descriptions of the HSD unitized fuel and area control and of the Bendix unitized fuel and area control are provided in Appendix A and Appendix B, respectively.

The main component of the control system houses all of the engine controls described above and is referred to as the unitized control. It is supplied with fuel pressure and flow by three pumps: a gear type pump to supply fuel to the gas generator, a centrifugal fuel pump to supply fuel to the duct heater, and a piston-type pump to provide hydraulic pressure for duct heater exhaust nozzle and reverser-suppressor clamshell operation.

Power command inputs from the airframe to the control are mechanical and consist of a single lever controlling forward and reverse thrust, and a separate lever co~ rolling fuel shutoff. Remote adjustment of fuel flow ratio is provided to permit setting engine pressure ratio (EPR) in the maximum nonaugmented and the augmented regions.

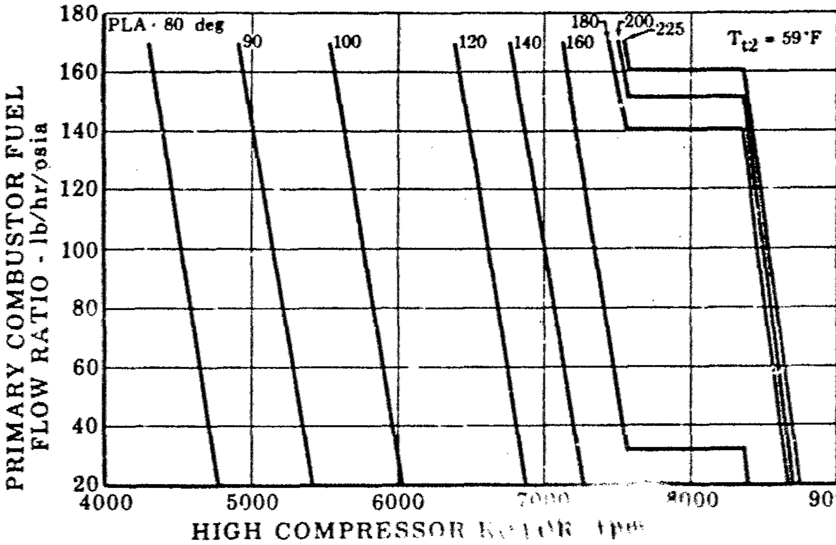

Figure 1. Estimated Po... Lever Schedule - Forward FD 16445 BIII

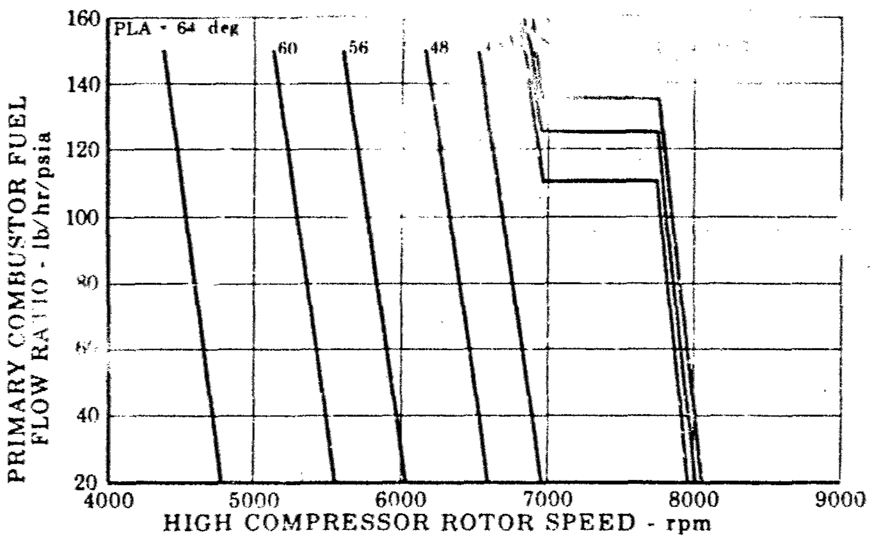

Figure 2. Estimated Power Lever Schedule - Reverse FD 16446 bIII

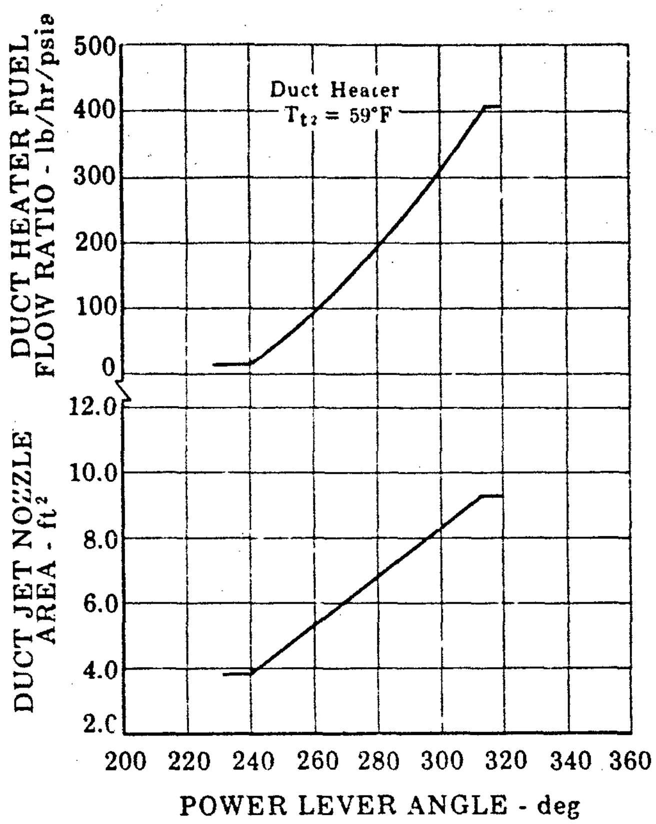

Figure 3. Estimated Duct Heater Power Lever Schedule FD 16447 BIII

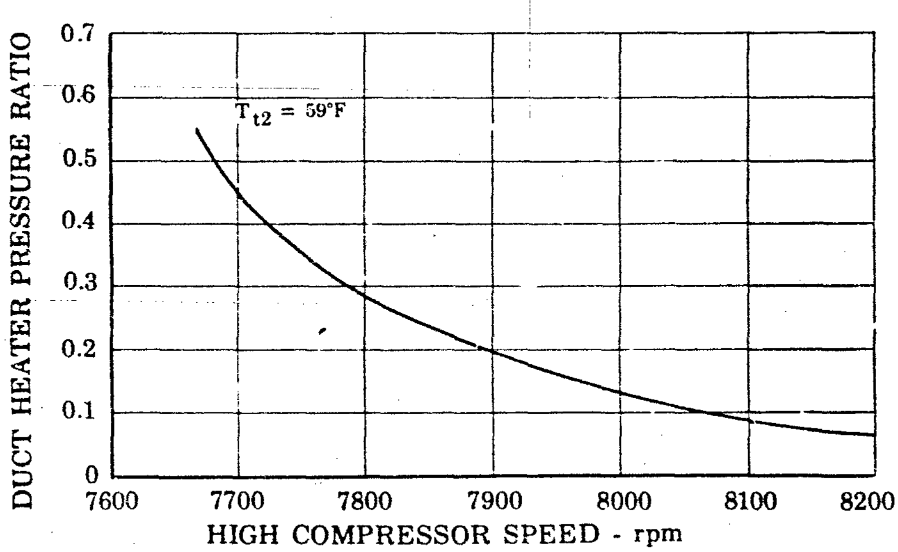

Figure 4. Estimated Duct Heater Pressure Ratio Schedule FD 16448 BIII

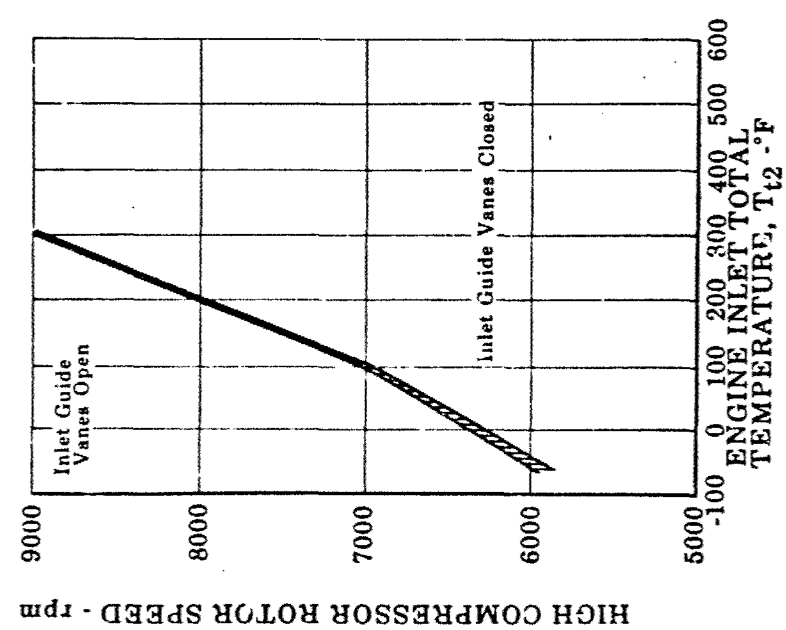

Figure 5. Estimated Inlet Guide Vane Actuation

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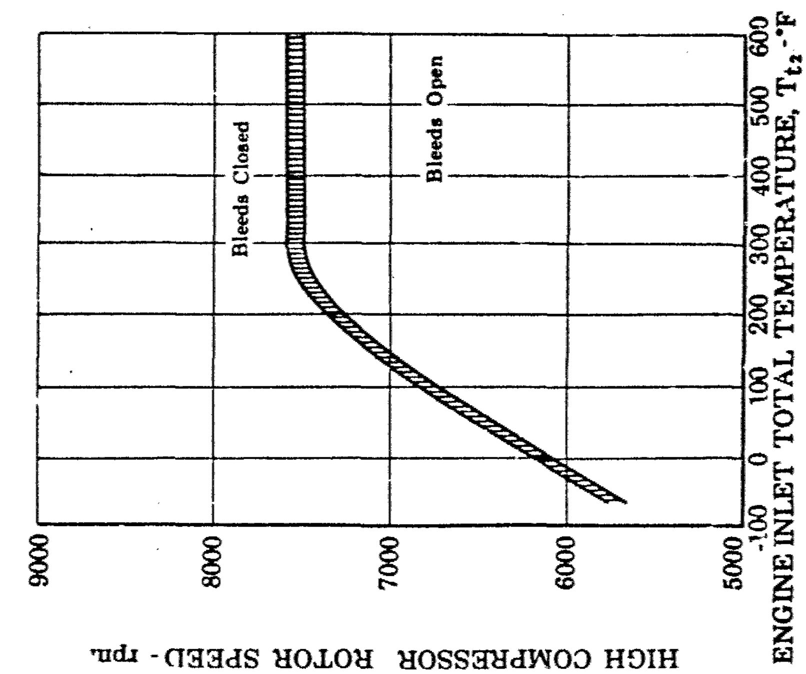

Figure 6. Estimated High Compressor Bleed Actuation

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The unitized control performs all the required computing functions in one cam linkage system that responds to input signals from the aircraft and engine. Fuel is metered to the gas generator to set the desired engine pressure ratio (EPR) for both augmented and nonaugmented operation. Fuel is metered to the duct heater to set the desired thrust augmentation. The duct heater exhaust nozzle is positioned to provide control of total engine airflow. Schedules are included to sequence the reverser-suppressor system, the high compressor inlet stator position, the compressor bleed position, and the duct heater ignition system. Safety features are incorporated to shutoff duct fuel flow and to reduce gas generator fuel flow automatically in the event of a duct heater flameout.

Figure 1 of paragraph A of this section shows a simplified representation of the airframe inlet, the JTF17 engine, and control system. Figure 4 of paragraph A of this section is a more detailed schematic showing the relation of the fuel system components to the engine. The unitized control diagram is separated in segments for convenience.

The scheduled gas generator fuel flow is metered by the throttle valve, which is positioned by the computer portion of the control in response to the input signals. A constant pressure drop is maintained across this valve by the pressure regulating valve which bypasses excess fuel back to pump interstage pressure.

To protect the system from excessive fuel temperature, a thermal bypass valve is incorporated in the unitized control. With PWA 522 (Jet A, A-1) fuel, this valve opens a supplementary bypass port when the fuel temperature at the control inlet exceeds 250°F for gas generator flows of less than 5000 pph and 285°F for gas generator flows greater than 5000 pph. This bypassed fuel is returned to the aircraft system as required to prevent excessive temperature in the engine fuel system.

Fuel is metered to the gas generator burner in response to power lever and computer system inputs when the shutoff lever is in the fuel "on" position.

Starting, accelerating, speed governing, and decelerating schedules are used to regulate this flow to protect the engine from overtemperature and overstress at all times. For starting and acceleration to the desired speed, an acceleration scheduling cam is provided. The cam is translated by engine speed and is rotated by engine inlet temperature. The cam contour provides a schedule of fuel/air ratio (Wf/Pb) that is multiplied by primary combustor pressure, Pb, to provide fuel flows to safely accelerate the engine in the minimum time. Overspeed protection is provided for the compressor high speed rotor by a steep overspeed droop slope in the acceleration cam. Typical schedules are shown on figure 7.

At all power settings below those that require maximum turbine temperature, the gas generator fuel flow is regulated by a proportional governor which senses rotor speed. This governed speed is selected by the power lever angle and biased by engine inlet temperature. See figure 7. At idle, the governed speed can be adjusted with a manual ground adjustment on the control to permit trimming of engine idle speed.

At steady-state maximum nonaugmented and all augmented conditions the gas generator fuel/air ratio is scheduled as a function of engine inlet temperature and compressor speed to provide the desired engine pressure ratio. The delivered fuel can be manually adjusted to permit adjusting gas generator pressure ratio in the duct augmented range and maximum non-augmented range if desired. The gas generator control fuel schedules are illustrated by figure 7.

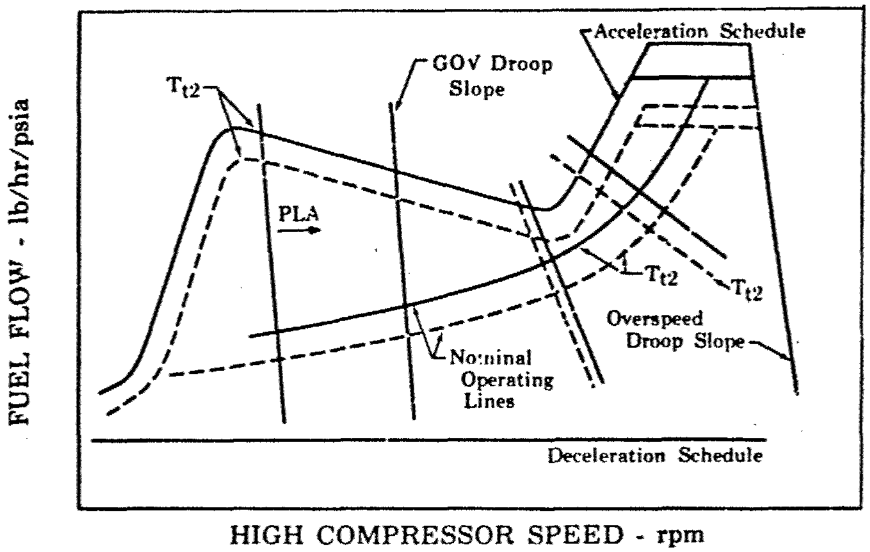

Figure 7. Estimated Gas Generator Fuel Flow Schedule FD 16452 BIII

The compressor stators are positioned in one of two positions by a fuel actuator. Cruise and takeoff positions are automatically scheduled as a function of compressor high rotor speed and engine inlet temperature. See figure 5.

The compressor bleed actuators are positioned by pneumatic pressure directed by an externally mounted control valve which is signalled by fluid pressure signals from the control. Bleeds open and closed positions are automatically scheduled as a function of compressor high rotor speed and engine inlet temperature. See figure 6.

The reverser-suppressor is actuated by hydraulic fuel actuators. Take-off and reverse position are selected by power lever positions as shown in figure 1 of paragraph B of this section. The desired duct fuel flow is scheduled by the duct control metering valve. A throttling-type regulating valve maintains a constant pressure drop across the metering valve. To minimize the amount of throttling required in the control and heat rejection to the fuel, the air supply to the duct heater turbopump is modulated to vary pump speed to the minimum required to hold a constant pressure drop across the complete duct fuel control metering section at all engine power settings.

Fuel is metered to the duct heater as a function of power lever position and T_{t2} as shown on figure 3. Power lever translates a 3D cam and T_{t2} rotates the cam, the output of which is the desired duct heater fuel flow burner pressure ratio. This ratio is multiplied by burner pressure resulting in a signal proportional to fuel flow being generated.

The duct heater incorporates two zones of fuel injection. Within the unitized control, each zone is provided with a fuel shutoff valve and a manifold rapid fill system. This latter system reduces by a significant amount the time required for augmenter transients by providing a high rate of fuel flow from the gas generator boost pump during the fill period. Each zone is also provided with separate fuel pressure signals for operating the fuel manifold dump valves.

When the power lever is advanced beyond the maximum nonaugmentation flat to the minimum duct augmentation flat, a sequencing valve in the unitized fuel control initiates the following events: (1) the Zone I manifold dump valve closes, (2) the Zone I rapid-fill valve opens, (3) the Zone I shutoff valve opens, (4) the duct exhaust nozzle resets partially open, and (5) the duct igniters are energized. Fuel is delivered to the Zone I fuel manifold at a high flow rate until a pressure signal indicates the manifold is full. The rapid-fill valve closes, the igniters are turned off, and the duct exhaust nozzle reset is removed.

Further power lever advancement increases duct fuel-air ratio and duct nozzle area on a coordinated schedule to hold the total engine airflow constant.

If the power lever is moved to the Zone II range, the Zone II fuel manifold dump valve is closed, the Zone II shutoff valve is opened, and the Zone II rapid-fill valve is opened to fill the Zone II fuel manifold. A constant fuel-air ratio is held during the Zone II rapid-fill transient. Pressure increasing in the Zone II manifold provides a signal resulting in closing of the rapid-fill valve and simultaneous routing of metered fuel to the Zone II manifold. Total duct fuel flow is divided between Zone I and Zone II by the fuel nozzle flow characteristics. Zone II fuel ignites spontaneously when the fuel enters the burner. Continued power lever advancement causes increased duct heater fuel flow, increased engine thrust, and increased duct nozzle area to maintain constant engine airflow. Maximum duct augmentation is scheduled by power lever position. Fuel flow for quick filling of both the Zone I and Zone II fuel manifolds is supplied from interstage pressure of the gas generator fuel pump. The duct heater fuel flow schedule is shown in figure 8.

The total corrected engine airflow is controlled as a function of T_{t2} to the schedule defined in the engine specification. The airflow control is achieved by actuating the variable duct exhaust nozzle. In the cruise range the nominal airflow schedule may be manually adjusted by the flight crew between maximum and minimum limits to obtain optimum inlet performance. The total corrected airflow schedule, the maximum and minimum limits, and the nominal schedule coordinated with Boeing are shown in figure 10 of paragraph B of this section, while those coordinated with Lockheed are shown in figure 9 of paragraph B of this section.

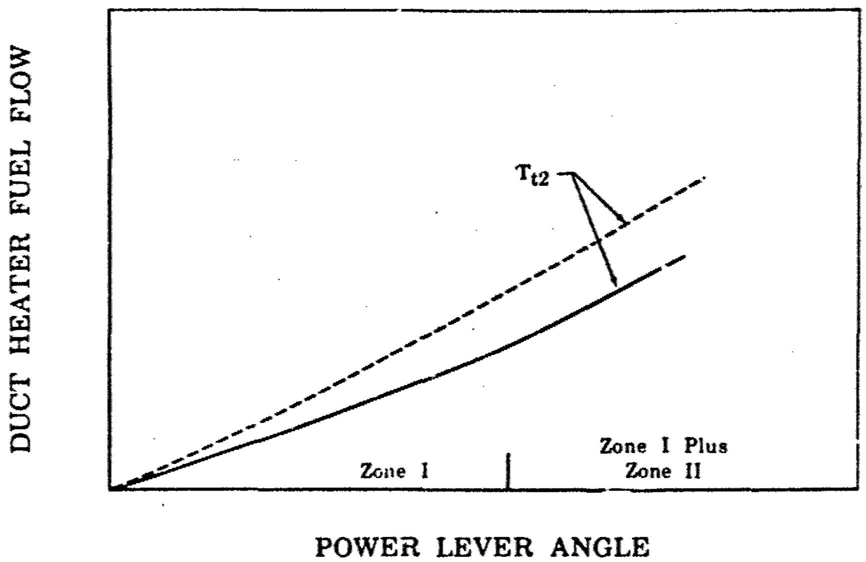

Figure 8. Duct Heater Fuel Flow Schedule FD 16453 BIII

Total engine airflow is the sum of gas generator airflow and duct airflow. Gas generator airflow is determined by sensing high rotor speed and engine inlet temperature. Knowing this airflow permits determining the duct airflow required to obtain the desired total engine airflow. Therefore, desired duct airflow parameter will be scheduled as a function of high rotor speed and engine inlet temperature, as shown on figure 4.

The duct corrected airflow is measured using the duct pressure ratio parameter, which is the difference between fan discharge total pressure and fan discharge static pressure divided by fan discharge total pressure, $(P_{t3}-P_{s3})/P_{t3}$. This same parameter is utilized in supersonic aircraft air induction controls. The unitized control will schedule the duct pressure ratio necessary to obtain the desired duct airflow. The actual duct pressure ratio will be determined by the control and compared with the scheduled pressure ratio. The difference between the pressure ratios initiates corrective action through a proportional plus integral servo and a power boost servo to reposition the duct exhaust nozzle as required in a closed loop basis to obtain the desired duct airflow.

c. Separately Mounted Components

The unitized fuel and area control system includes the following separately mounted components:

1. Turbopump controller, which regulates air supplied to the duct heater fuel pump by modulating a butterfly valve located in the pump air inlet supply.
2. Two engine inlet temperature sensors which sense temperature with a gas-filled tube. The resultant gas pressure is transduced into a fluid pressure and in turn sensed by the control for use as engine inlet temperature bias.

3. Compressor bleed control valve, which ports compressor discharge pressure or nacelle ambient pressure to the compressor bleed actuator as signalled by the control.

d. Differences Between Production and Prototype Controls

Several minor differences may exist between the prototype and production models of the unitized fuel and area control. These differences are tabulated below. The procurement specification will be the same for prototype and production controls and is included in paragraph Q of this section.

1. Prototype controls will contain pressure taps, adjustments, and schedule changing capabilities for trouble shooting and control-engine development which will not be included in the production units.
2. Prototype controls are required to have the capability of either hydromechanical or electronic computation, scheduling, and trimming of engine total airflow and associated functions with the capability of incorporating either of the two systems on a single control assembly.
3. Prototype controls are designed with the capability of adding unit insulation. Prototype testing may indicate that, due to high nacelle velocities, localized control heating may occur. Partial or total insulation can be added as necessary to prevent excessive local control temperatures.

e. Auxiliary Subsystems

(1) Engine Fuel Control Thrust Reverser Interlock

The purpose of this assembly is to interlock the engine fuel control power lever and the clamshell thrust reversers in such a way that:

1. When in forward flight, motion of the clamshells to reverse position (closed) will return the control power lever to the idle position.
2. When in reverse thrust, motion of the clamshells to the forward flight position will return the control power lever to the idle position.
3. It shall not be possible to obtain forward thrust power lever positions above idle unless the clamshells are in the forward flight position.
4. It shall not be possible to obtain reverse thrust power lever positions below idle unless the clamshells are in the reverse position.

The above requirements are met by a combination of stops and gearing arranged and connected so that the completed unit is a compact assembly approximately 4 1/2 inches in diameter by 3 inches long. The unit, which is self-contained, shielded from dirt and foreign matter, and easy to service or replace, consists of a housing with the following parts (see figure 9):

1. A throttle pulley which is coordinated with the fuel control by cables.
2. A clamshell pulley which is coordinated with the motion of the clamshell by a feedback system. This pulley is mounted on a shaft with the throttle pulley so that the stops on both pulleys rotate in the same plane and interact with each other at the proper positions.
3. A flying wedge is geared to the clamshell pulley by planetary gearing to amplify the wedge motion to fit the unitized fuel and area control power lever angle requirements. This flying wedge is located between the throttle pulley and clamshell pulley on the same shaft so that the wedge rotates in the same plane with the throttle pulley stops.

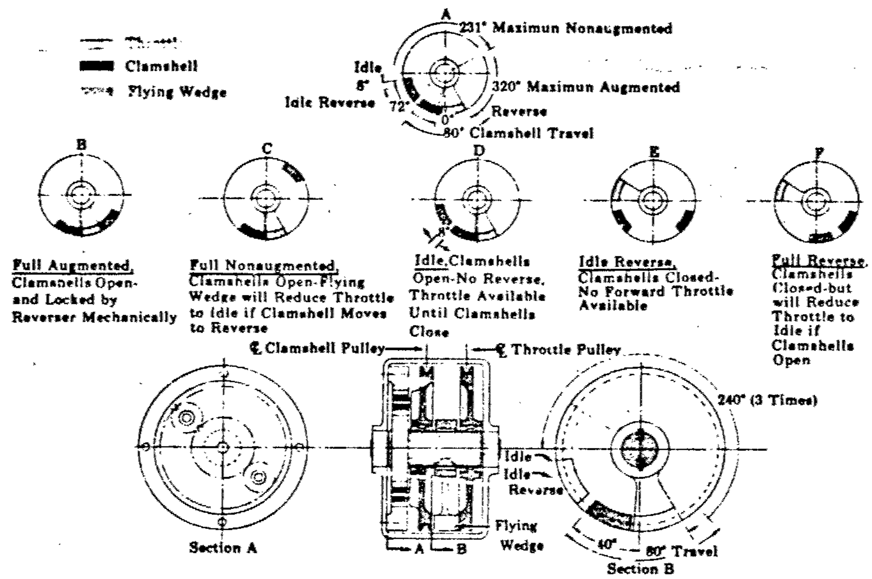

Figure 9. Throttle-Reverser Interlock Assembly FD 16436 BIII

At the clamshell open position, the clamshell pulley stop is approximately 8 degrees from the throttle pulley stop in idle position (figure 9, view D). In this 8 degrees of motion, the throttle signals the clamshells to close (reverse), which in turn moves the clamshell stop counter clockwise (figure 9, view E). If the clamshells do not move, the stop remains in position and the throttle can no longer continue in the reverse thrust direction.

In the full reverse condition (figure 9, view F), the clamshell stop is clear of the throttle stop. Motion of the clamshell toward the open position causes the clamshell stop to move clockwise, contacting the throttle stop and pushing the stop to the idle position.

During forward flight with the clamshells open, the flying wedge is at the full forward position (figure 9, view B and C) clear of all forward flight throttle stop positions. Motion of the clamshell toward the reverse position causes the clamshell stop to move counterclockwise (figure 9, view E), contracting the throttle stop and pushing the stop to the idle position.

When the clamshells are closed, the flying wedge is at the throttle stop idle position, preventing the throttle from being moved in the forward thrust direction until the clamshells are opened.

(2) Duct Airflow Sensing Probes

Total and static duct air pressure must be sensed as accurately and with as fast a response as possible because these signals are used to control the duct corrected airflow, which must be held within close tolerances to ensure aircraft engine compatibility.

(a) Total Pressure Pickups

A Kiel probe, shown in figure 10, is a scaled-up version of "off-the-shelf" probes that are used extensively for airflow measurement. The Kiel probe consists of a probe rotated 90° from the flange such that the open end faces the oncoming air. The probe is surrounded by an annular tube-like shield, which is approximately two probe diameters across, protrudes approximately two probe diameters upstream of the probe end, and extends past the 90-degree probe bend.

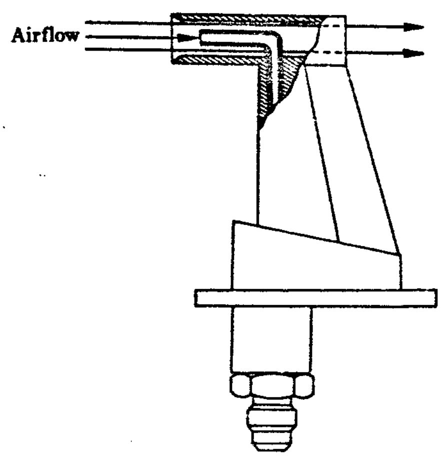

Figure 10. Kiel Total Pressure Probe

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(b) Static Pressure Pickups

Several wall static tap configurations were considered. Two holes 0.100 inch in diameter were found to produce the desired response and noise characteristics.

(3) Secondary Air Control System

A system is being provided to control the secondary airflow bypassed through ducts around the engine from the engine inlet to the ejector. This system is utilized on the Boeing engine only. The requirements of this control system are:

1. The secondary airflow bypass ducts must be closed to prevent recirculation of engine exhaust gases whenever the pressure in the ejector is greater than the pressure at the inlet of the engine. This occurs during takeoff, landing, and reverse operations.
2. The secondary airflow bypass ducts must be open sufficiently to provide cooling airflow to the ejector whenever the ejector blow-in doors are closed and the engine is being operated at maximum duct heat conditions.
3. At cruise conditions, an airflow equal to approximately 2% of the total engine corrected airflow is bypassed to optimize ejector performance.

The system chosen to fulfill the above requirements consists of six ducts each incorporating a check valve to prevent reverse flow. Four of the ducts will also incorporate butterfly valves, which are positioned by two position actuators. A valve in the unitized fuel and area control will direct fuel pressure to the actuators to position the butterfly valves in two of the ducts to the closed position as a function of compressor inlet temperature. Another valve will be incorporated in the control which will direct fuel pressure to the two remaining actuators to close the butterfly valves as a function of shut-off lever position. This function will be performed by crew action at cruise conditions. The unitized control will incorporate an interlock to prevent closing the bypass ducts when engine power is in excess of 80% of maximum augmented thrust. This interlock will prevent inadvertent closing of the bypass ducts when the duct heater is operating at a condition which would overtemperature the ejector. The resulting bypass airflow schedule will be as shown in figure 11.

The Lockheed engine incorporates check valves in the reverser-suppressor aft of the engine rear mount ring that close to prevent recirculation of exhaust gases whenever ejector pressure is higher than engine inlet air pressure. This is the only secondary airflow control provision required on the Lockheed engine.

(4) Duct Nozzle Feedback

The duct nozzle feedback system, illustrated in figure 12, incorporates a push rod which connects to the duct nozzle synchronizing ring. The motion of the push rod is converted to rotary motion near the duct nozzle and this motion is then carried forward to the unitized fuel and area control by a pulley-cable system. The bearings being used in this system are the result of an extensive development effort on the J58 engine to obtain a satisfactory bearing for the cable system between the main and afterburner controls. The bearing races are made of a cobalt alloy having a high hardness and the internal radial clearance and inner and outer

raceway curvature have been increased over that of a normal bearing to improve the bearing tolerance to operating in a hot, contaminated environment.

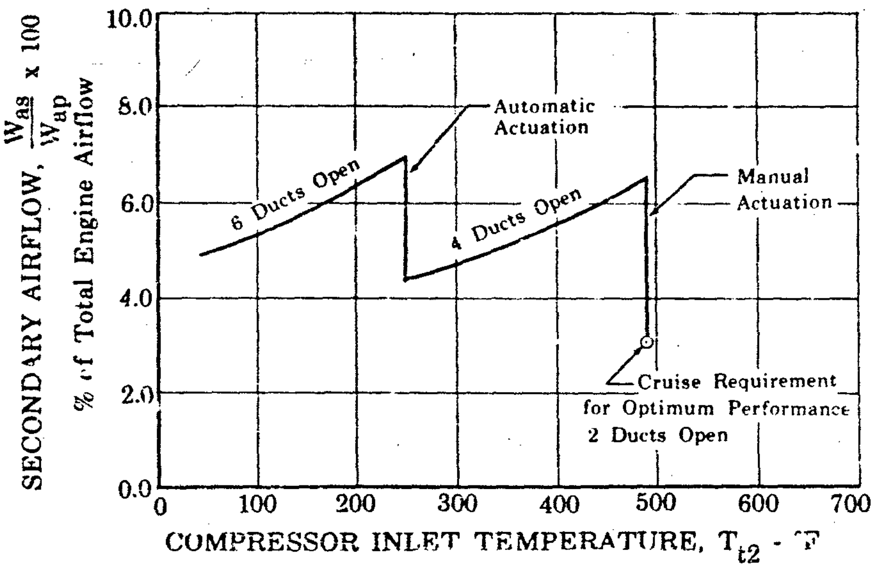


Figure 11. JTF17 Estimated Performance Actual Secondary Airflow Schedule - Boeing Configuration FD 17145 BIII

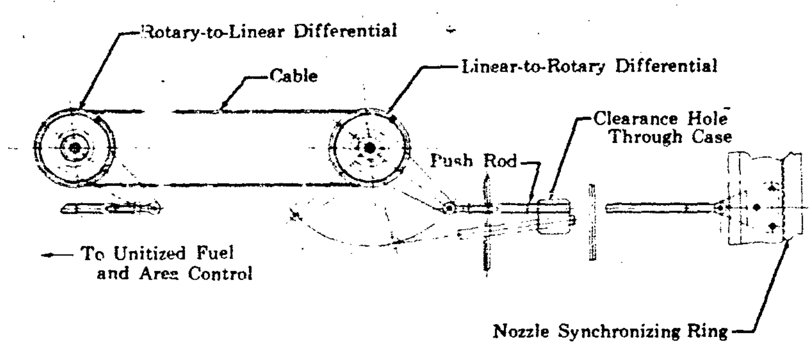


Figure 12. JTF17 Duct Heater Exhaust Nozzle Feedback System FD 16862 BIII

2. Design Objectives

The unitized fuel and area control for the JTF17 engine will be designed to meet the following objectives:

1. To provide the required control of the engine throughout the flight envelope without exceeding the operating limits of the engine and to provide the necessary compatibility with the airframe inlet

2. To provide a system which permits normal operation to be conducted with simplicity throughout the operating range of the engine
3. To provide the limiting features necessary to maintain safe engine operation during emergencies
4. To perform the necessary functions without exceeding acceptable weight and volume requirements
5. To provide a unit capable of rapid replacement on the engine and of being readily maintained and overhauled with normally replaced parts designed for ease of fabrication and low cost
6. To be capable of development into a unit that has durability equivalent to or better than current commercial aircraft turbine engine controls

3. Unitized Fuel and Area Control Design Requirements

Specific design requirements for the JTF17 control are based on past Pratt & Whitney Aircraft commercial and high Mach number experience. Procurement Specification PWA-PPS-J112 describes the technical requirements of the control and is summarized in paragraph Q of this section. The following are the design requirements for the control.

1. The control will operate with fluid conforming to PWA 522 (Jet A, A-1) and PWA Specification 533, the commercial aircraft turbine engine fuel recommended for use with the JTF17 engine.
2. The control will provide the engine with the proper schedules. Typical schedule requirements are shown in the following figures: figures 1 and 2 present the gas generator speed and maximum wf/Pb schedule for 59°F T_{t2} for forward and reverse mode operation respectively, and figure 3 presents the 59°F T_{t2} duct heater fuel and nozzle schedule. Figure 4 presents the duct heater pressure ratio schedule for 59°F T_{t2}. Figures 5 and 6 give the compressor inlet guide vane and compressor bleed schedules respectively.
3. The control will schedule gas generator fuel over the range of the flight envelope between the engine acceleration and deceleration limits. Maximum gas generator fuel flow will be 30,000 pph, and minimum gas generator fuel flow will be 1200 pph.
4. The control will operate satisfactorily with a rate of change of inlet fuel temperature of 180°F per minute from -65 to 370°F fuel temperature.

5. The control normal operating fuel pressures are as follows:

Item	Range
Pump inlet	5 to 50 psig
Pump interstage	5 to 225 psig
Gas generator control inlet	160 to 900 psig
Duct heater control inlet	120 to 1200 psig
Hydraulic supply	1500 to 1750 psig
Ambient pressure	2 to 38 psia

6. The control burst fuel pressures are as follows:

Item	Pressure, psig
Pump inlet	125
Pump interstage	450
Gas generator control inlet	2025
Duct heater control inlet	2700
Hydraulic supply	3950

Tests to demonstrate this requirement will be performed by the vendor. Burst pressure proof tests to destruction will be performed on a production fabricated unit.

7. The control will operate satisfactorily for 80 hours with fuel contaminated by 8 grams of foreign matter per 1000 gallons with the following particle size composition:

Particle Size - Microns		Percent of Total
0 - 5	0 - 5	39 ± 2 by weight
5 - 10	5 - 10	18 ± 3 by weight
10 - 20	10 - 20	16 ± 3 by weight
20 - 40	20 - 40	18 ± 3 by weight
Over 40	Over 40	9 ± 3 by weight
Through a 200-mesh screen		100 by weight

8. The control will operate satisfactorily over the range of steady state ambient temperature from -70 to 575°F and the range of pressure from 3.0 to 38 psia.

9. The operating life of the control will be a 50,000-hour housing life, with design target weights as follows:

Item	Weight, lb
Unitized control	120
Duct heater pump controller	10
Compressor inlet temperature sensors	3
Compressor bleed control valve	3

10. The control will be self-purging of any entrapped air or vapor and will prevent any accumulation of water or vapor in any air pressure sense line or control bellows installed on the engine.

11. All parts of the control requiring routine service checking, adjustment, or replacement while on the engine must be readily accessible, including fuel screens and filters and the adjustments listed below:

1. Idle speed
2. Start bleed actuation speed
3. Duct fuel flow ratio
4. Gas generator meter ΔP
5. Duct heater meter ΔP
6. Power lever indexing adjustment
7. Cutoff lever indexing adjustment
8. Maximum gas generator fuel flow ratio
9. Duct heater $\Delta P/P$ schedule

12. Design requirements for the control input parameters will be as follows:

Input Variable	Range	Units
High rot r speed (N_2)	0 to 8800	rpm
Engine inlet temperature (T_{T_2})	-70 to 650	^oF
Power lever angle	0 to 320	degrees
Shutoff lever angle	0 to 90	degrees
Duct nozzle area	0 to 7	in. actuator travel
Burner pressure (P_b)	3 to 265	psia
Duct inlet total pressure (P_{t_3})	6 to 60	psia
Duct inlet static pressure (P_{s_3})	5 to 50	psia

13. Duct heater fuel flow requirements will vary between 800 pph minimum and 90,000 pph maximum. Duct heater lightoff will occur at a fuel/air ratio of 0.002 to provide duct heater lightoff characteristics consistent with the characteristics programmed into the analog, which showed excellent lightoff compatibility with the inlet system.
14. Duct heater nozzle area will modulate over an area from 12.0 to 3.5 square feet. Control of the nozzle will be implemented by the nominal power lever schedule and biased as indicated by the duct heater pressure ratio.
15. Control of compressor bleeds, compressor inlet guide vanes, and reverser-suppressor will be provided. Bleed and vane schedules will be automatic, and the reverser-suppressor schedule will be integrated with the power lever.
16. Safety features and interlocks to be incorporated will:

    1. Prevent rotor overspeed by sensing high rotor speed and reducing fuel flow if overspeed occurs
    2. Sense increased duct airflow, which could result from a duct heater blowout, and shutoff duct heater fuel flow and reduce gas generator fuel flow to prevent low rotor overspeed
    3. Provide redundant engine inlet temperature sensors for improved reliability
    4. Provide interlocks with the reverser-suppressor as defined in Report B, Section II, paragraph F of this volume.

4. Unitized Fuel and Area Control Design Criteria

Some design criteria for the JTF17 control are as follows:

1. Bellows and springs will be designed for 100,000,000 normal operating cycles and 1,000,000 maximum transient cycles.
2. The design stress for steel parts will not exceed that stress which is equivalent to 1,000,000 fatigue cycles. The design stress for aluminum parts will not exceed that stress which is equivalent to 100,000,000 fatigue cycles.
3. Rotating pilot valves will turn at a maximum of 2000 rpm except where performance is improved by higher speed. The lower speeds will reduce pilot valve load and wear.
4. Servo systems will be designed to produce a minimum of 25 pounds force, nonrotating valves will be designed for a maximum motivating force of 10 pounds, and rotating valves will be designed for a maximum motivating force of 3 pounds.
5. All conventional straight screw threads will conform to MIL-S-8879, which incorporates a controlled root radius. This thread configuration has proved during the J58 program to be superior for high Mach number applications.
6. Each control will be designed to withstand 1.5 times normal operating pressure without any performance loss and to withstand a burst pressure 1.5 times proof pressure without fracture or sufficient deformation to cause an external leak. Burst pressure will be demonstrated on a production fabricated unit.
7. Control flow passages will be designed to prevent accumulation of entrained contamination. Consideration will also be given to prevention of entrained contamination affecting control performance by selective layout of flow passages, design and location of servo bleeds, and force washing of servos.

5. Materials Summary

The materials to be used in the unitized fuel and area control were selected on the basis of the successful experience which has been obtained on the J58 and TF30 engines and are shown in table 1. Changes from this list will be made only after thorough analysis of JTF17 requirements.

Table 1. Unitized Fuel and Area Control Materials

Material	Typical Usage
RR 350 (Hiduminium)	Hydraulic housings, covers, fuel manifolds
AMS 5616	Linkages, sleeves, pins
AMS 5508	Brackets
AMS 5673	Springs
AMS 5630	Cams, cam followers
AMS 5673	Springs
AMS 7245	Inserts
AMS 5591	Tubing
AMS 5625	Bolts
A 286	Screws
Kentanium K 162B	Wear plates
Carboloy 44A	Wear plates
Teflon, Miplon	Seals

Several housing materials and fabrication techniques were considered early in the program, including primarily cast steel, fabricated steel, fabricated titanium, cast aluminum, and forged aluminum. The major criteria used to evaluate the relative advantages of each of these materials were life, weight, and cost effectiveness. In analytically evaluating the life of the housing materials, particular attention was given to time-temperature estimates of the environment and to the stress, creep, and fatigue characteristics of the material. A rigid thermodynamic study was conducted on a typical housing section, using the various combinations of fuel temperature, environmental air temperature, air velocity, and housing material. As a result accurate values of skin temperature, seal temperature, and temperature profile were determined. With these characteristics established, it was possible to start at a desired housing life, establish allowable stress levels for each material, and work back to a housing configuration.

As the housing detail continued to evolve from these studies, and the schematic concept and arrangement of the control became well enough established, the most promising materials and techniques were each used in detailing a typical control housing for weight and cost evaluation. As housing detailing commenced, two facts became evident. These were (1) the higher allowable stresses of steel could not always be achieved because of necessary casting and fabricating tolerances, and (2) many sealing areas in steel and titanium housings were inherently heavier than aluminum because seal temperatures required the use of metallic seals rather than the elastomeric seals used with aluminum. Although a great deal of study was directed toward fabricated housings, which proved to be extremely competitive with aluminum on a weight basis, the initial cost and repair and modification cost eliminated fabricated housings from consideration for this particular application.

The conclusion of the housing material study was to use aluminum housings cast from a high temperature alloy, RR 350 (Hiduminium). A properly designed aluminum housing has definite advantages in weight, cost, repairability, and sealing techniques, as well as being able to meet the life requirements with a wide margin.

The procurement specification for the unitized control required P&WA approval prior to using certain stainless steels, that become brittle when heat-treated. An example is AMS 5630, a material which has proved to be very satisfactory for cams. Use of the material will be permitted if means are used to ensure that the part will not be subjected to excessive impact loads. The specification also requires similar approval prior to the use of copper or copper alloys because of the detrimental effect of the element on the thermal stability of turbine engine fuels.

Both unitized control vendors have conducted tests on high temperature elastomeric compounds which meet the environmental and pressure requirements to be encountered in the supersonic transport application. Hamilton Standard has a proprietory fluorosilicon compound which will be used in static seal locations. Bendix is using a fluorocarbon compound seal, which is performing satisfactorily in the initial experimental JTF17 engine. Both vendors will use a glass-filled fluorocarbon for dynamic seal applications.

The control will permanently capture in close-fitting chambers "O" ring type seals to prevent seal fretting from pressure pulsations or the seals being pulled out of grooves by high velocity fuel. These problems have been encountered in current commercial controls and have been eliminated by close control of seal chamber sizes.

Several design concepts are being reviewed regarding a method to incorporate steel inserts into the aluminum castings to permit the use of steel fittings and metal gaskets for the plumbing connections. These concepts are shown in figure 13.

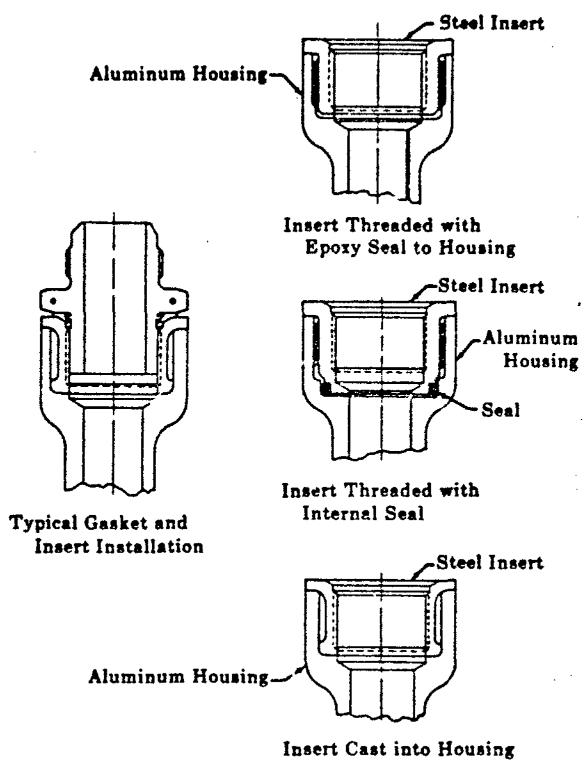


Figure 13. Design Concepts for Incorporating Control Component Plumbing Connections — FD 16749 BIII

6. Design Approach and Status

a. General

The control for a duct-burning turbofan engine for use on a supersonic transport aircraft has been under study by P&WA in accordance with Government contracts since 1961. Two fuel control vendors, Hamilton Standard of United Aircraft Corporation and Bendix Products, Aerospace Division, were contracted by P&WA to assist in the study program.

The initial work performed on a control system for a SST engine consisted of evaluating several modes of control and selecting the best of these modes for further investigation. A preliminary purchase specifi-

cation was written for the unitized fuel and area control and was provided to the two fuel control vendors. The work initially performed by the vendors consisted of three parts:

1. Performing a design study of the control defined by the preliminary purchase specification to obtain a schematic diagram of a unit which is capable of being developed into a flightworthy unit.
2. Studying the concept of incorporating all the functions required of the gas generator control system and the duct heater control system into a single package.
3. Initiating programs to investigate, design and develop new design concepts which might be applicable to a control for a supersonic transport engine.

Both Hamilton Standard and Bendix generated schematic diagrams, external configuration drawings, and preliminary internal configuration drawings. This work was supported by computer effort which helped to define both general and specific requirements for the control.

Both vendors concluded that the unitized control concept has definite advantage over the multiple control concept. This subject is discussed in more detail later in this section.

Additional data are presented in paragraph C of this Section covering the analytical and computer work performed to define the control configuration.

b. Hydromechanical Selection

The design approach selected for the unitized fuel and area control defined a hydromechanical component that utilizes fuel supplied by the engine fuel pumps for operating the computing and scheduling mechanisms. This approach was selected to make maximum use of the unequalled experience with this type of control that P&WA and the two control vendors have accumulated. The materials, techniques, and experience gained in development of housings, seals, bearings, and environmental compensation to satisfy the higher temperature and altitude requirements of the J58 and the extended usage requirements of commercial airline operation will be used in the design program for the unitized control.

The Mach 3 experience has clearly shown which areas of engine controls will require extensive development effort to obtain a satisfactory design. The major cause of the problems, which will be in addition to those normally expected for an engine control system for commercial application, will be the increased ambient air and fuel temperatures which will be encountered. Incidents were encountered in the J58 program in which control accuracy, stability, and durability were decreased by the elevated temperatures.

Variations in fuel temperatures affect such items as spring loads, lever deflections, fuel metering within a closed loop servo, and casting growth. All of these items can decrease the accuracy of control output functions if the proper elimination of or compensation for these varia-

tions is not provided. Both Hamilton Standard and Bendix have gained considerable experience relative to such compensations; this experience will be applied to the design of the unitized fuel and area control and will have a definite effect on the development program of the control. Similar accuracy variations can also be encountered as the result of housing growth as a function of ambient air temperature changes. Once again, these effects will be minimized by proper design.

Elevated temperature operation durability problems encountered in the J58 program were overcome by utilization of different materials, incorporation of hardened inserts at critical wear points, reduced fabrication tolerance for critical dimensions, and improved and more closely controlled quality control procedures. This last item became evident during the early production phase of the J58 control program. All of the techniques developed during the J58 program to obtain improved control system durability will be directly applied to the design of the unitized fuel and area control.

The experience obtained from the J58 program will be utilized by implementing the following specific considerations into the design of the unitized control, thereby providing a firm base toward obtaining the desired reliability and durability goals:

1. The critical computing portions of the control will not carry any of the mounting or plumbing loads and will be supported entirely by internal members and enclosed by fuel containing vessels.
2. Bolts with a hardness in excess of 38 Rockwell C will not be used due to susceptibility to stress corrosion. Attempts will be made for all screws with the same thread size to be the same or sufficiently different in length to insure that the screws cannot be misused.
3. Seals will have low installation forces. Metal chevron seals will not be used for dynamic applications. Elastomeric seals of a high-temperature fluorosilicone or fluoricarbon compound will be used where seal temperatures will be less than 500°F.
4. Adequate wall sections to maintain stresses and deflections within safe limits will be preferred over ribbed sections since uniform sections are easier to fabricate and less susceptible to high stress concentrations due to pressure loading or thermal gradients.
5. Housing temperatures will be maintained at desired values by internal fuel cooling with care being taken to ensure this fuel does not impinge upon control linkages to cause scheduling errors. If housing temperatures cannot be maintained, as demonstrated by component and engine testing, external thermal insulation will be considered.
6. External servo tubes or fitting standpipes brazed into the control housing will not be used.
7. Sensing compressor inlet temperature and transducing the signal to a fuel pressure will be accomplished on a redundant basis.
8. Servos will be fabricated as modules, where possible, to permit individual calibrations and prevent erroneous servo performance which might result from unequal thermal effects. A servo piston

will not operate in a bore that is common to two housings. Means will be provided to measure servo piston travel during bench calibrations, where practical.

9. Large hard-surface cams will be used to provide signal-to-error ratios as high as possible. Wear points of linkage systems will incorporate hardened inserts for bearings to minimize wear. Also, linkage systems will incorporate ball bearings at the critical pivot points. Linkage system design will be directed toward obtaining links that have as little deflection as possible.
10. Replaceable sleeves for major valves and pistons will be utilized, facilitating required changes and replacement due to excessive wear and improving the system maintainability. Retention of such sleeves will be adequate to prevent movement that results in erroneous control performance.
11. Materials susceptible to stress corrosion will be used with extreme caution. Chrome surface rubbing on another chrome surface will be avoided. Deep nitriding for case hardening will be discouraged due to susceptibility to chipping. Liquid nitriding will be the preferred method of case hardening.
12. The system will be designed to tolerate fluctuations in the reference and supply pressures. Pressure signals will cover as large a range as possible to maintain the signal-to-noise ratio as high as possible.
13. Fine grit vapor blasting has been beneficial in deburring sliding or rotating steel parts and will be used extensively.
14. Half-ball valves, spool valves, and servos of the null position type will have all parts grounded to a common surface so that thermal gradients do not cause a change in null location.
15. Pressure regulating valves will have variable sensitivity controlled by the contour of the port or valve so that stability and control of pressure is uniform over a large turndown ratio.
16. Servo system loops will be closed at the final output of the servo when possible to keep servo accuracy at an optimum.
17. Temperature-sensitive devices used for compensating for changes in fuel density will be located so that the fuel sample is typical of that which is passing through the metering valve.
18. The engine inlet temperature sensor will have a time constant goal of 3 seconds or less to ensure control scheduling accuracies are maintained during transient operating conditions.

In addition to its high Mach number experience, P&WA will use its commercial aircraft turbine engine experience as a source of data relative to the performance of fuel controls that have been subjected to high time operation. Maintenance and overhaul conferences are held on an annual basis by airline representatives and P&WA to cover turbine engine accessories and components. Data from such conferences and other meetings with airline representatives have indicated problem areas that have been encountered on current commercial fuel controls.

The following specific considerations will be utilized in the design of the control based on this commercial operating experience:

1. Sufficient filtration capability will be provided. The need to control the radial position of a screen or filter so that a shadow from a seam does not affect control performance will be eliminated. All leakage past critical parts will be filtered fuel.
2. The design of metering valves and/or lands will ensure that the metering will occur at the desired location. Pressure pickup points for the metering head pressure sensor will be well defined machined areas to ensure that consistency exists between controls.
3. The number of bellows assemblies will be kept to a minimum. Fabrication of the bellows will be reviewed to determine the need for fabrication in an inert atmosphere to prevent corrosion.
4. Carbon faced seals on shafts will be protected from direct flow of contaminated fuel by the use of shielding or by the design of the seal location to obtain the desired life.
5. A prime design objective will be to locate adjustments and filters so that they are readily accessible. Locking means, other than safety wire, for the nuts or screws which retain filter covers will be considered to facilitate rapid removal and installation of the filters. Care will be exercised in this design to assure that locking reliability is not impaired by repeated usage and extreme temperatures.
6. Loaded snap rings will not be used. Other snap rings shall be retained to ensure that the ring does not come out inadvertently.
7. The use of laminated shims will not be permitted.
8. External plumbing connections will not be located along the sides of the control adjacent to the nacelle wall or engine castings.
9. Rigging pins for accurately defining certain power lever shaft positions will be incorporated as a permanent part of the control.
10. Power lever shaft seals will be the carbon face type. Diametrical clearance with elastomeric seals will be used only if temperature permits. All shaft seals will be doubled with an overboard drain between the inboard and outboard seal.
11. Forced lubrication of the drive spline and improved control of gearbox drive pad machining is planned to eliminate excessive drive spline wear.
12. Ice accumulation in cavities for bellows will be eliminated by designing the cavity to drain through the pressure sense line, thus eliminating fuel scheduling errors that result from the ice preventing proper sense bellows operation.
13. The use of hardened inserts at critical wear points in linkage systems will be utilized to eliminate a cause of output schedule shifts.

Throughout the development program of the unitized control, reviews will be held to learn of the additional experience obtained from the J58 engine and commercial aircraft turbine engines and thereby keep abreast of new developments that might affect the design of the unitized control.

The selection of the hydromechanical type of unitized fuel and area control has the following advantages:

1. Implementation of design concepts into physical hardware of this type has been accomplished for many successful aircraft turbine engine controls.
2. Controls of this type are currently on an operational basis and are being subjected to additional extreme environmental conditions, providing a known base from which to judge other types of controls.
3. Advancements in design and packaging have resulted in a high density compact control capable of being developed into a reliable unit providing many desirable features relative to maintainability.
4. Inclusion of the control into airlines overhaul facilities can be readily accomplished since the component is the same type as the controls currently being handled by the facilities.

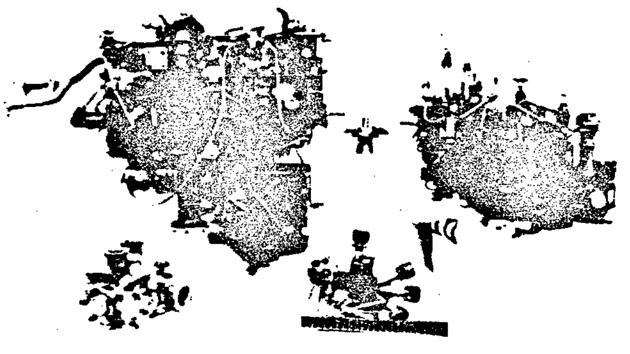
c. Unitized Control Concept

Prior to the initial phase of the supersonic transport program sponsored by the Federal Aviation Authority, the concept of incorporating all the major functions required of an engine control system into a single package rather than a number of individual controls was conceived. A reduction in total size and weight was believed to be possible with such a concept. As part of the Phase I work performed on the supersonic transport program, P&WA subcontracted with Hamilton Standard and Bendix to conduct studies to determine the feasibility and the advantages and disadvantages of such a concept.

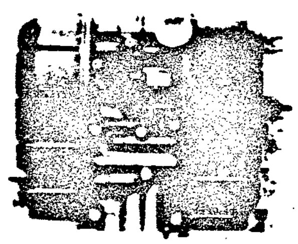
The study revealed a significant reduction in weight, a significant improvement in reliability, simplification of the system, and cost reductions could be realized by utilizing the unitized control concept. The saving is the result of elimination of duplicate servos for parameters such as engine pressures and compressor inlet temperature; reduced quantity of plumbing, associated fittings, and support brackets; and reduced total casting volume. A comparative weight study was made at that time between the estimated weight of the unitized control for a supersonic transport engine application and the actual weight of the equivalent components and interconnecting plumbing for the TF30 engine control system. The unitized control weight was approximately 30% less than the TF30 control system even with the need for the greatly increased flow requirements for the supersonic application.

Separate controls require numerous additional hydraulic lines and connections that always are potential sources of fuel leakage. The unitized control, as currently defined, requires 31 connections to other parts of the engine or airframe (3 lever or pulley connections, 23 fuel connections, 3 pneumatic connections, 1 speed input, and 1 electrical connection). For this mode of control to be accomplished with two major components, namely the gas generator control and the duct heater fuel and nozzle control, the total number of connections would be 46. This breaks down to 5 lever or pulley connections, 33 fuel connections, 4 pneumatic connections, 2 speed inputs, and 2 electrical connections.

The decision to utilize a unitized fuel and area control instead of separate controls for the supersonic transport application was made early in the study program. As additional personnel became active in the program, the comparison was reviewed and re-reviewed with the same conclusions being reached each time: the unitized concept will result in a considerable saving of weight and space with the resultant simplification and increased reliability. Figure 14 shows a comparison between the Hamilton Standard unitized fuel and area control and the equivalent components for the J58 engine (the pictures have a one-to-one relationship relative to size).



J58 Fuel and Area Controls



JTF17 Unitized Fuel and Area Control

Figure 14. JTF17 Unitized Fuel and Area Control Compared to Equivalent J58 Fuel Controls (Comparative Size Scale 1:1) FD 16747 BIII

The unitized concept reduces the number of parts in the control system, which results in a marked improvement in reliability and reduces the maintenance. A reliability comparison was made between the two control concepts by applying data directly from commercial turbine engine fuel control experience with no allowances being made for the change in operating environmental conditions. Such data were considered valid for comparative purposes, but not for absolute level values. The comparison revealed the reliability of the unitized control to be approximately 36% better than that of the separate control system. The reason for this is the larger reduction in overall complexity achieved by using the unitized concept. Table 2 presents a summary of this comparison. This improved reliability should greatly improve the overall reliability of the control system.

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Table 2. Reliability Comparison

Component	Uncorrected Failure Rate (Estimated) (Failures/10^6 Hours) Unitized Control	Separate Controls Main	Separate Controls Duct Area
Remote T_{t2} Sensor	14	14	
T_{t2} Servo	8.5	8.5	8.5
P_{t4} Sensor Servo	15.5	15.5	15.5
Speed Sensor and Servo	9	9	9
PRV	12.5	12.5	
RU Servo and Pilot Valve	5		5
Rate Limit Valve	1		1
Fan Blow-out Valves	2		2
Light-off Valves (2)	2		2
Fail Safe SOV	1		1
T.V.	12.5		12.5
PRV	11.5		12.5
$\Delta P/P$ Servo	10.5		10.5
$\Delta P/P$ Integrating Piston	4.0		4.0
Area Pilot Valve and Servo	4		4
$\Delta P/P$ Servo and Pilot Valve	6		9
Bleed Act. Pilot Valve	1	1	
$\Delta P/P$ Reset Piston	1		1
N Arming Signal	1	1	
Subtotal	122.0	61.5	97.5
Add 55% for seals	67.1	33.8	53.5
Add 67% for linkage wear	81.7	41.2	65.5
Component total	270.8	136.5	216.5
System total	270.8	353.0	

d. Rapid Replacement Concept

A design goal to replace a unitized fuel and area control on an engine within a 30-minute period resulted in the rapid replacement concept. This concept requires that all fuel connections to the control pass through a single plane. Fuel seals for this parting surface will be retained by a seal plate, with any seal being replaceable in the plate in the event the seal is damaged. All electrical connections to the control enter through one connector. However, the pneumatic connections to the control, of which there are three, were maintained as individual connections to provide the ability to ensure air leakage is not encountered. The rotory inputs to the control, such as power lever, shutoff lever, exhaust nozzle area feedback, and high rotor speed, incorporate means which permit quick removal of the control. The concept

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has been reviewed by both unitized control vendors and has resulted in control packaging concepts which meet the design goal. This concept not only reduces time required to change the control, but reduces maintenance cost and greatly increases overhaul bench utilization. The tooling necessary to replace the unitized control on an installed engine will be designed.

e. Servo Modules

Another design concept which has been utilized from the initial phase of the unitized control study is the utilization of servo modules. These modules are complete closed-loop servos that can be assembled and calibrated as a subassembly unit. This concept has enabled both vendors to obtain servo arrangement and packaging that is compatible with both the unitized and quick disconnect concepts. Calibration of these modules as subassemblies will provide definite reductions in development effort and calibration times, which in turn will improve the maintainability and reduce overhaul time and cost of the unit. A design goal that is part of this concept is to utilize the same parts in several modules, thereby improving the reliability and logistical aspects of the control.

f. Piston, Valve and Bore Design

The design of pistons and the bores in which they operate will utilize the extensive experience accumulated from the J58 engine. Close clearance metal-to-metal fits with adequate concentricity control, proper material selection, and vapor blasting of each surface has proved to be reliable. Similar requirements also have been proved necessary for rotating pilot valves. Experience has shown that hot fuel testing prior to control acceptance has been proved as a means to detect poor quality parts.

Production facilities and techniques have been developed over the past eight years which allow the economical and reliable manufacture of these valves and pistons. Moreover, design techniques have evolved which make it possible to produce difficult and variable system characteristics from a single valve.

g. Linkage Systems

The question of pivot and bearing life in linkage systems is primarily a question of contact stress and relative motion - a lightly loaded, quiet, slow moving pivot system will last indefinitely, but a heavily loaded, vibrating pivot system can be subject to rapid wear. The initial approach is to avoid designing excessive stresses into a pivot, particularly where the system requires a high gain and more or less continuous motion. In the event that a pivot must carry a fairly high contact stress, for example where point contact is desirable, then hardened carbide inserts will be employed to assure long life.

The same considerations as described above also apply to cam contours and cam followers. A great deal of experience has been accumulated using through-hardened AMS-5630 cams traced by AMS-5630 rolling ball followers. Successful experience dictates follower diameters, cam rise angles, and contact stresses. Because of the tremendous number of hours of commerical experience using these techniques and the successful

application of these techniques to other high temperature environments, no new difficulties are foreseen for the supersonic transport application.

h. Fuel Filtration

Suitable wash type servo flow filters for gas generator and duct heater servo flows will be provided to retain contamination of the size that might otherwise cause a control malfunction. The servo filters will have a check valve type bypass feature; if the filter becomes plugged due to excessive contamination or icing, the control will not be starved of servo pressure. The filter bypass system design will prevent entrance of the accumulated contaminant into the control during bypassing conditions. The control will also incorporate full flow filters of 20-mesh filtration at the inlet to both the gas generator and duct heater fuel scheduling systems. Installation, removal, and identification of the filter will be in accordance with present commercial practice. The design will ensure that removal of the filter will not leave accumulated contaminant in the control.

i. Airflow Control

The hydromechanical techniques used for the closed loop airflow control are based on experience with similar airflow controls on supersonic aircraft inlets. The required tolerance for the control of JTF17 engine airflow for compatibility with the airframe inlet, based on inlet-engine dynamic simulation studies, can be achieved with the hydromechanical control. However, should compatibility testing with actual hardware show that closer control is required, and alternative control with improved accuracy must be available.

Studies have been conducted to evaluate alternative ways of accomplishing the improved accuracy that may be required. These studies have indicated that an engine mounted airflow computer using digital electronic techniques offers significant accuracy advantages.

Vendor designs of the unitized fuel and area control include an alternate digital electronic assembly, which is mounted on the unitized fuel and area control, to provide airflow control. To accomplish this, the prototype unitized fuel control will be designed to have an interface so that the hydromechanical airflow module can be removed and the electronic airflow module installed to work with the same basic hydromechanical unitized fuel and area control. The electronic hardware will be constructed and evaluated on a test program paralleling the hydromechanical hardware program. The decision as to the type of unit to be incorporated in production engines will be made in the prototype phase on the basis of experience with actual hardware during both ground and flight inlet-engine compatibility testing.

j. Development Plan

The overall development plan to be used for the unitized fuel and area control will consist of basically the same steps and procedures that have been used successfully for similar controls for previous engines. The initial step is the preparation of a control procurement specification which is submitted to the vendors to be used as the basis of their

proposals. The choice of vendors requested to submit proposals is based on proved ability to produce similar components. The proposals submitted by the vendors will be evaluated, and a vendor will be selected upon conclusion of the P&WA evaluation. Vendor selection for the unitized fuel and area control will be completed during Phase II-C.

Upon selection of the control vendor, the procedure to be used for working with the vendor will follow the project engineering format that is used for engine development. Coordination meetings will be held on a continuing basis with the vendor to be certain the requirements specified in the control procurement specification are being met satisfactorily. In addition to these meetings, communications with the vendor will be maintained on a daily basis by letters of transmittal and telephone conversations. The procurement specification defines performance, design, installation, reliability, maintainability, safety, quality control, production, and service requirements. The communications with the vendor will ensure that these requirements are met at the appropriate time within the development program. Direction by P&WA of the vendor program will be continued throughout the complete development program. The parts list of the control which successfully completes the engine FTS will be released for use on the prototype JTF17 engine. The parts list of the control which successfully completes the engine type certification test will be released for use on production JTF17 engine.

Any subsequent changes are subject to a formal engineering change procedure that is defined by P&WA specifications. Such changes will be developed and substantiated as directed by the P&WA project engineer. Vendor procedures are discussed in more detail in Volume V, Report C, Section VI of this proposal.

k. Unitized Control Status

The requirements for unitized fuel and area control have been defined as the result of the analytical and computer effort described in Report B, Section III, paragraph C of this proposal. This definition is provided in PWA-PPS-J112, the purchase specification for the control, a summary of which is presented in paragraph Q of this section. The specification has been provided to both Hamilton Standard and Bendix for use in submitting proposals to design, fabricate, and develop the control.

Hamilton Standard and Bendix have maintained parallel programs during Phase II-C and are essentially at the same point. Schematic diagrams have been generated and show the basic concepts which would be used to implement the control. Numerous layout drawings have been made by both concerns to show the concepts are feasible. This is particularly true relative to packaging the control to meet the rapid replacement concept. Considerable effort has been expended on analytical studies to further implement the design of the control. As the requirements and concepts for the control changed, as the result of the continuing studies performed by P&WA, both vendors updated their efforts as required to remain current in the overall program.

Hamilton Standard and Bendix have submitted proposals to P&WA covering the unitized fuel and area control, and evaluation of the proposals is in process. Each proposal is being studied in detail, both on its own merit and on a comparative basis. At the conclusion of these studies, a vendor will be chosen. The selection will be based on the overall technical approach that best demonstrates an understanding of the problem and best fulfills the requirements set forth in the purchase specification, as well as on overall costs of the development program and projected costs of prototype and production units. Vendor selection will be made before the end of Phase II-C, and the selected vendor will start detail design immediately on Phase III go-ahead.

7. Design Concepts Layout Drawings

a. Bendix Drawings

The following figures present typical layout drawings prepared by Bendix to show the design concepts planned for particular areas:

(1) Figure 15

This figure shows the design of the pilot valve which controls the pressure to the reverser-suppressor actuators. The pilot valve is shown in the forward thrust position with P_{fo} pressure being supplied to port "A".

The valve will be actuated by a high pressure being supplied to P_{fx}, causing the valve to translate, and P_{fh} pressure being supplied to port "A", which will operate the reverser-suppressor actuators. This basic pilot valve design will also be used for controlling the high compressor inlet stators. Refer to figure 1, Appendix B for definition of the pressures in figures 15 and 16.

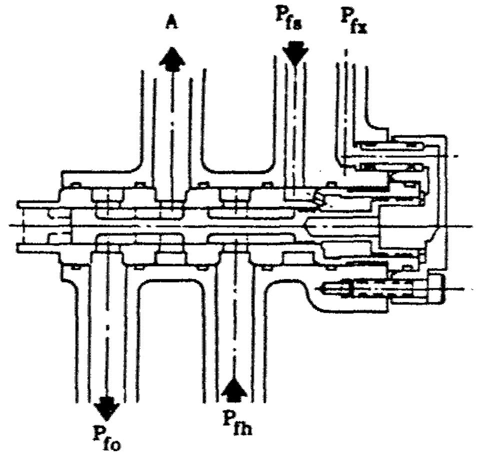


Figure 15. Reverser-Suppressor Control Valve Layout (Unitized Fuel and Area Control Bendix) FD 17146 BIII

(2) Figure 16

This figure shows the design of the pilot valve which controls the hydraulic pressures to the duct exhaust nozzle actuators. This valve will

be hydraulically positioned by the small ball valve at the left end of the pilot valve, the ball valve controlling the servo pressure at the right end of the pilot valve.

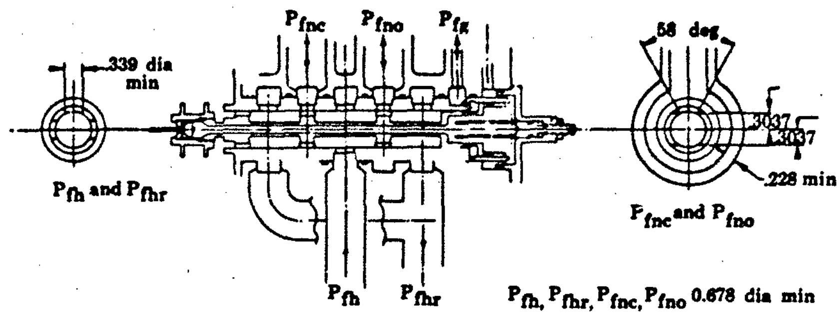


Figure 16. Duct Heater Exhaust Nozzle Control Valve Layout (Unitized Fuel and Area Control - Bendix)

FD 17147
BIII

(3) Figure 17

This figure shows the duct heater fuel metering valve and pressure regulator valve that will be fabricated and tested during the latter portion of Phase II-C. The metering valve is in the left end of the fixture and the regulating valve is in the right end.

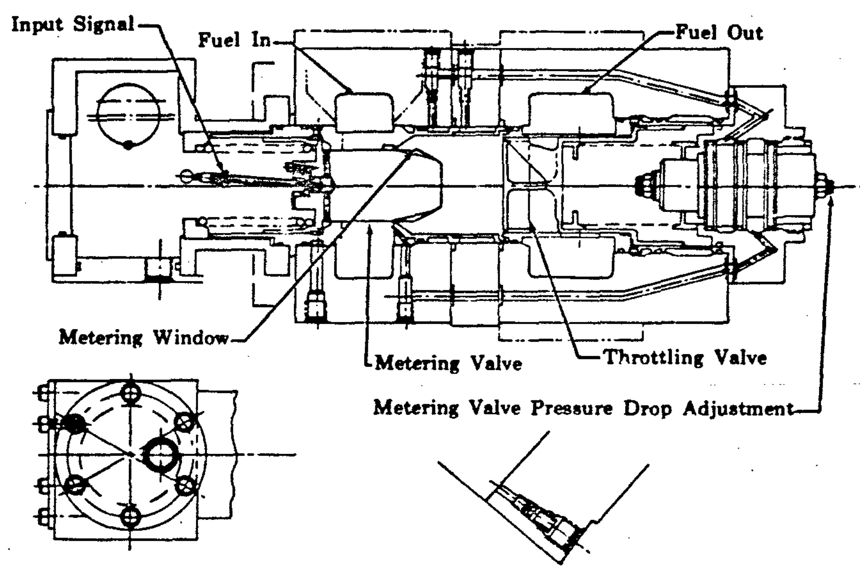


Figure 17. Duct Heater Fuel Metering Valve and Throttling Valve (Unitized Fuel and Area Control - Bendix)

FD 17148

(4) Figure 18

This figure shows the installation drawing of the Bendix unitized fuel and area control which incorporates the quick-disconnect concept.

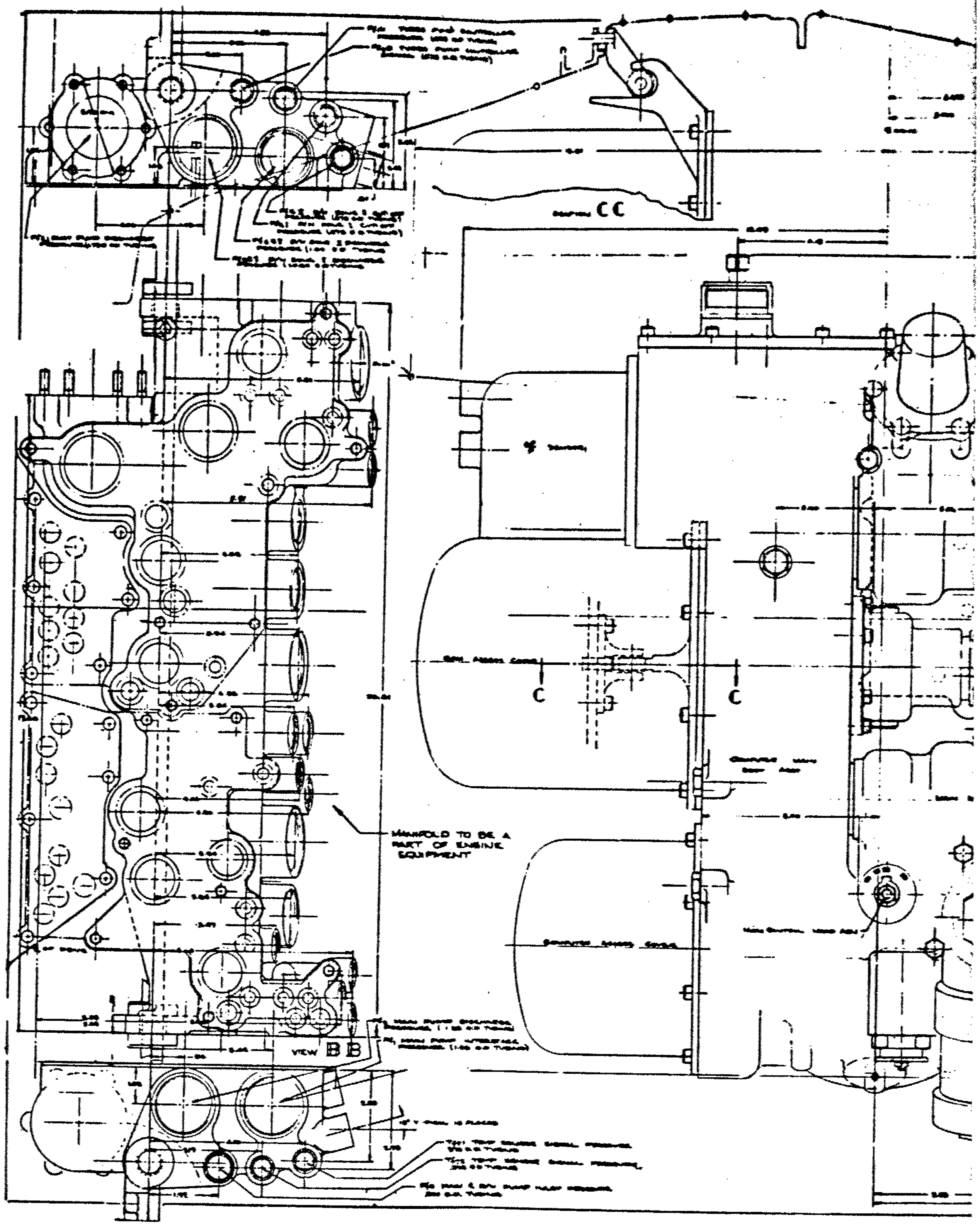
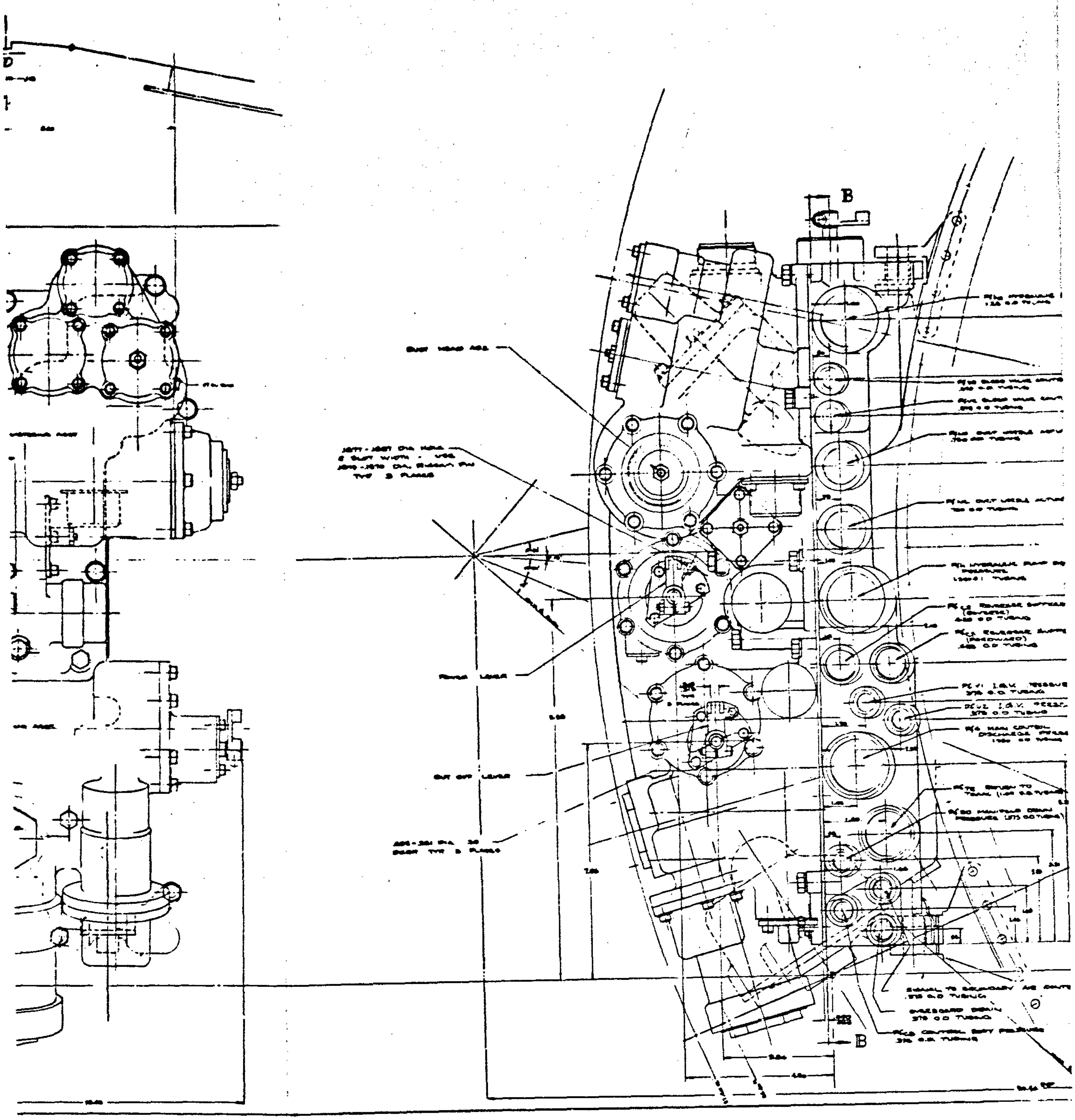


Figure 18. JTF17 Engine Unitized Fuel and Area Control Installation Layout - Bendix



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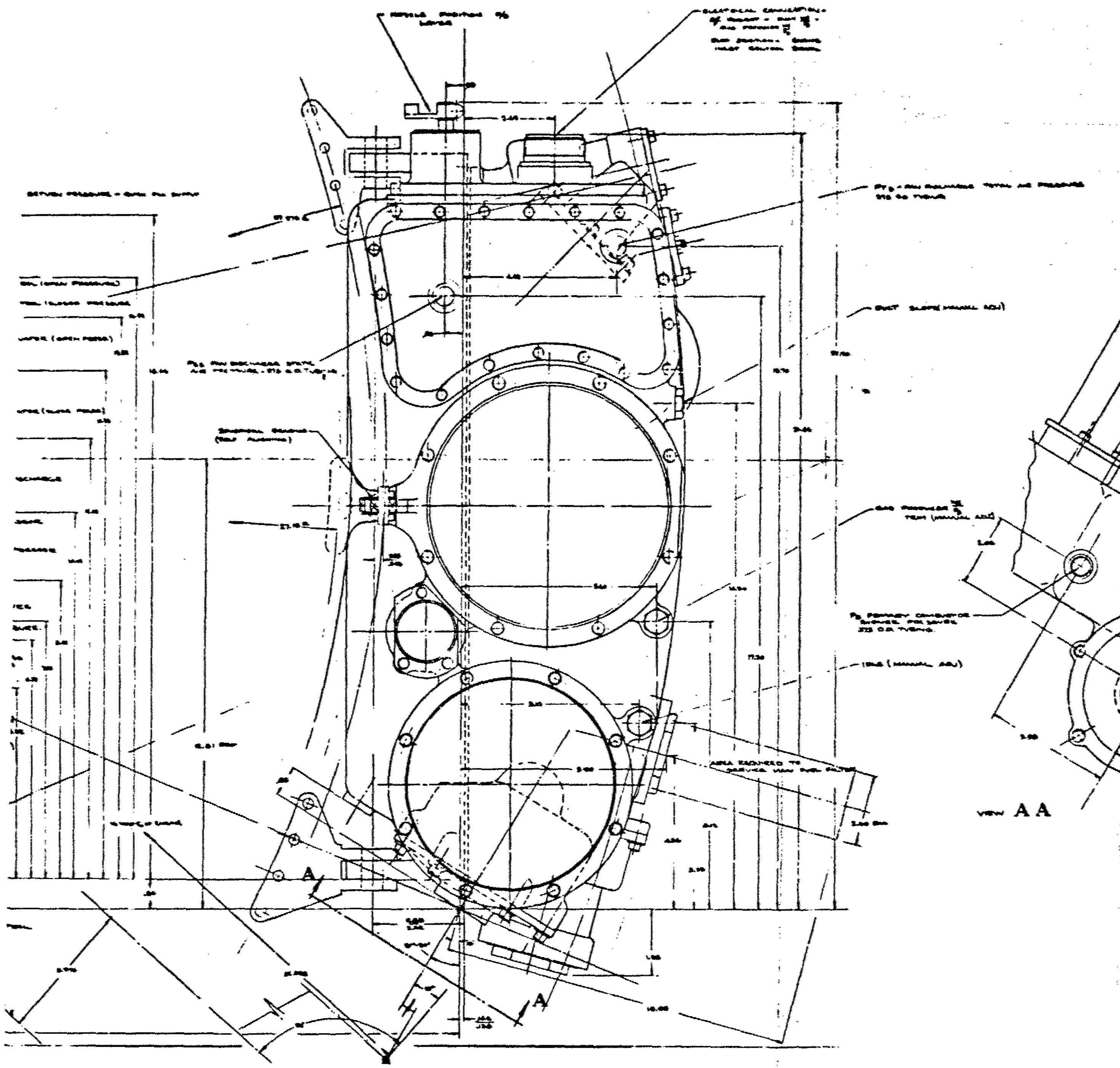

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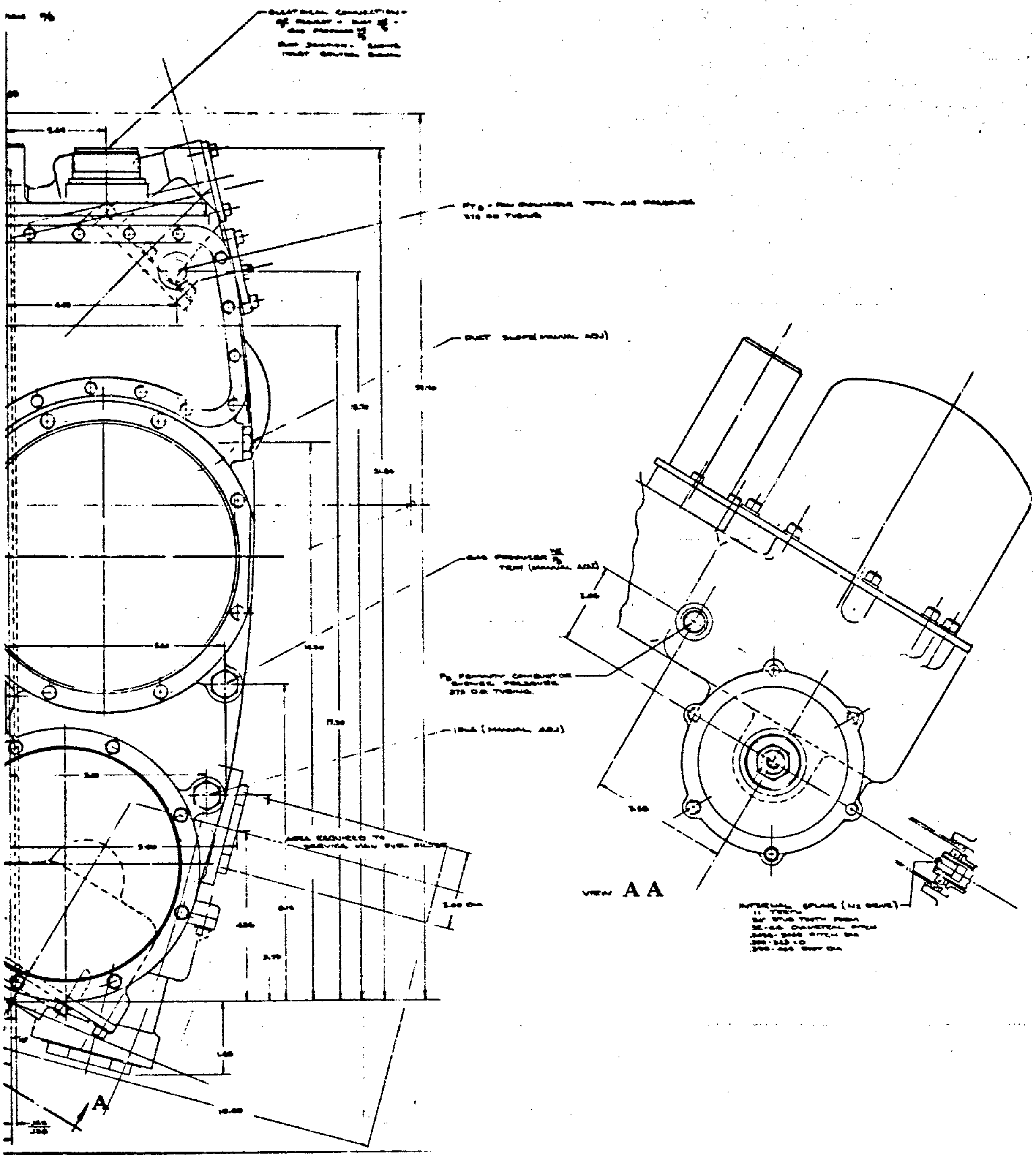

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b. Hamilton Standard Drawings

The following figures present typical layout drawings prepared by Hamilton Standard to show the design concepts planned for particular areas:

(1) Figure 19

This figure shows the arrangement of the major parts of the gas generator fuel metering system and other components in the same area.

(2) Figure 20

This figure shows the arrangement of the fuel metering details for the duct heater system.

(3) Figure 21

This figure shows the basic design and relative locations of the two shutoff valves in the duct heater fuel system.

(4) Figure 22

This figure presents a concept for locating several servos within the gas generator computing section and part of the accompanying linkage system.

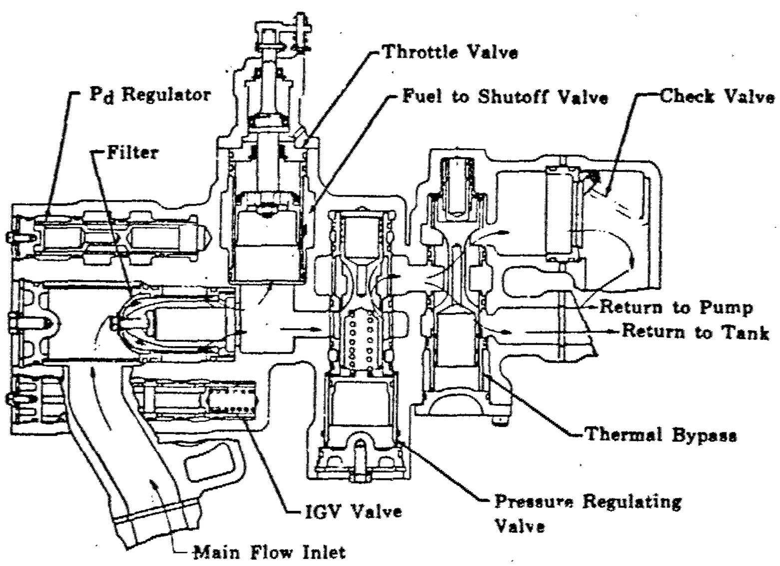


Figure 19. Gas Generator Fuel Metering System Arrangement (Unitized Fuel and Area Control - Hamilton Standard) FD 17152 BIII

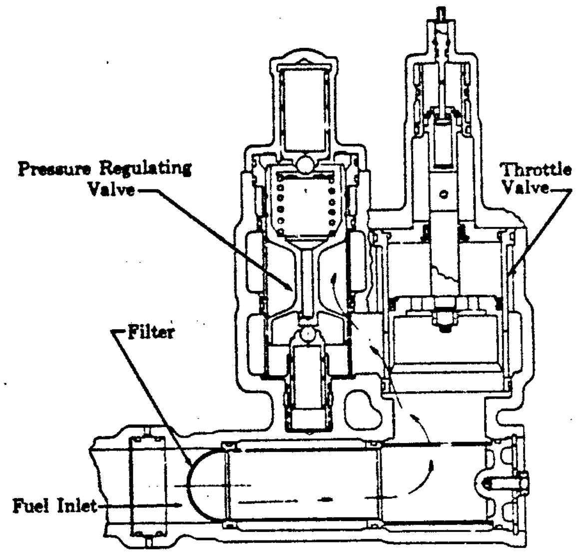


Figure 20. Duct Heater Fuel Metering System Arrangement (Unitized Fuel and Area Control - Hamilton Standard)

FD 17150
BIII

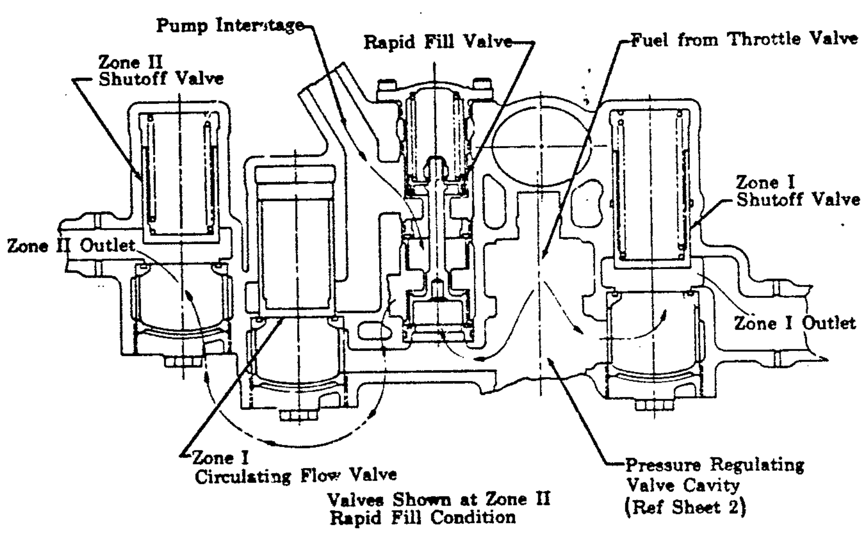


Figure 21. Duct Heater Fuel Shutoff Arrangement (Unitized Fuel and Area Control - Hamilton Standard)

FD 17151
BIII

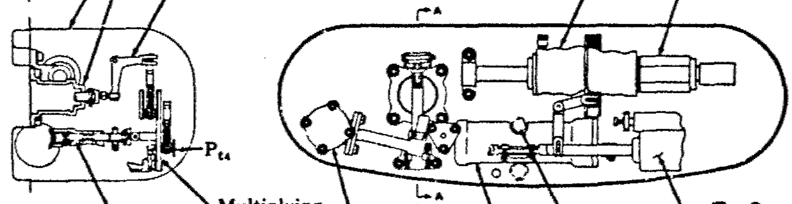


Figure 22. Gas Generator Computing Section (Unitized Fuel and Area Control - Hamilton Standard) FD 17149 BIII

(5) Figure 23

This figure shows a mockup of the casting cores required in the hydraulic housing of the control.

(6) Figure 24

This figure shows the installation drawing of the Hamilton Standard unitized fuel and area control.

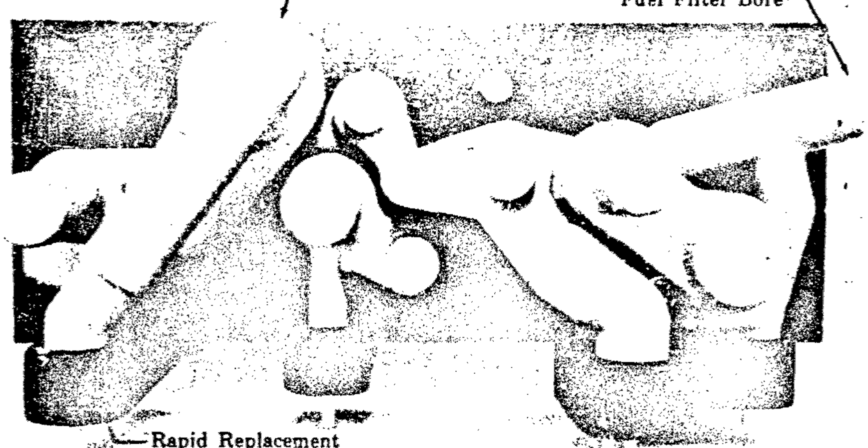


Figure 23. Hydraulic Casting Cores Mockup (Unitized Fuel and Area Control - Hamilton Standard) FD 16739 BIII

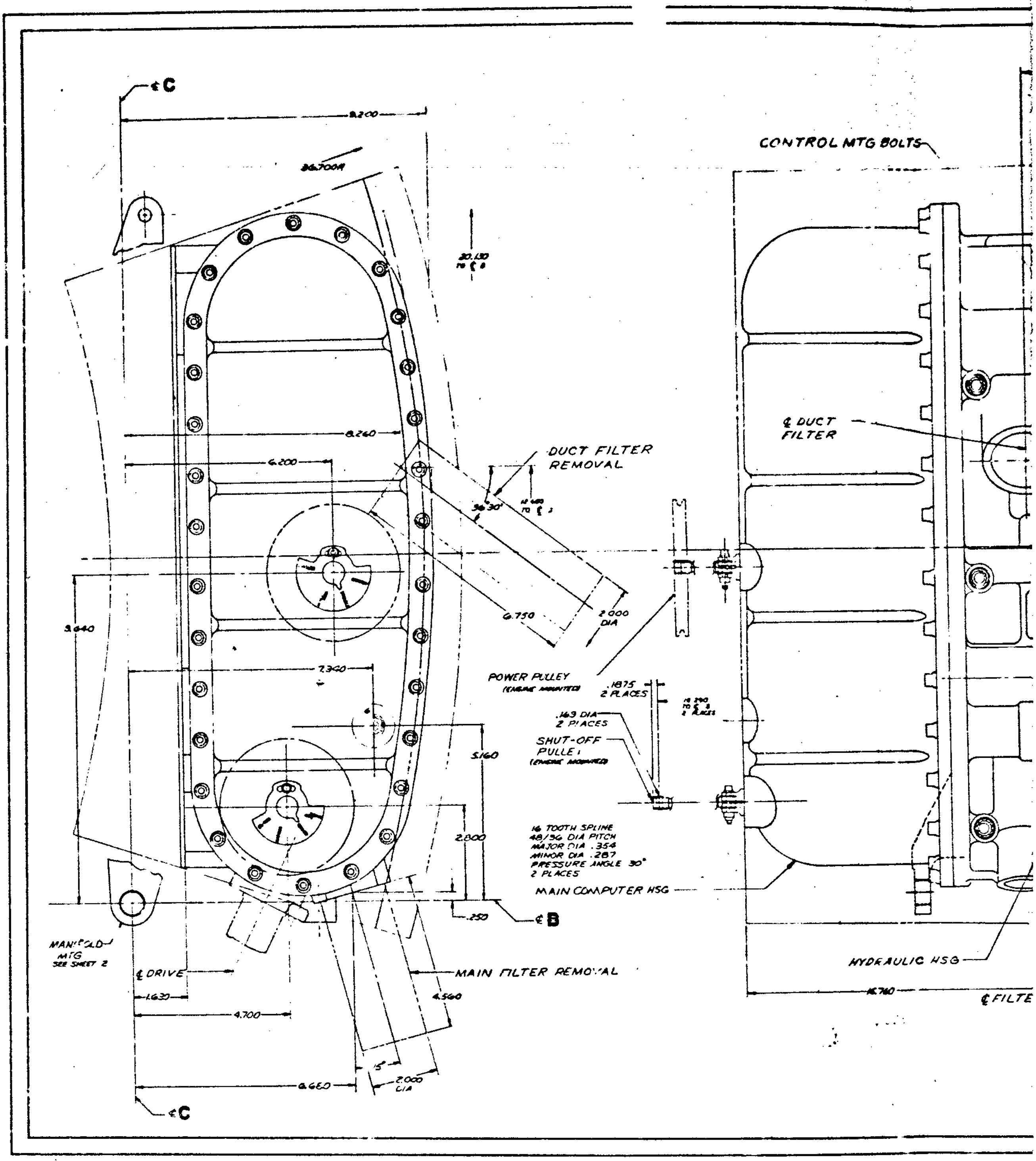
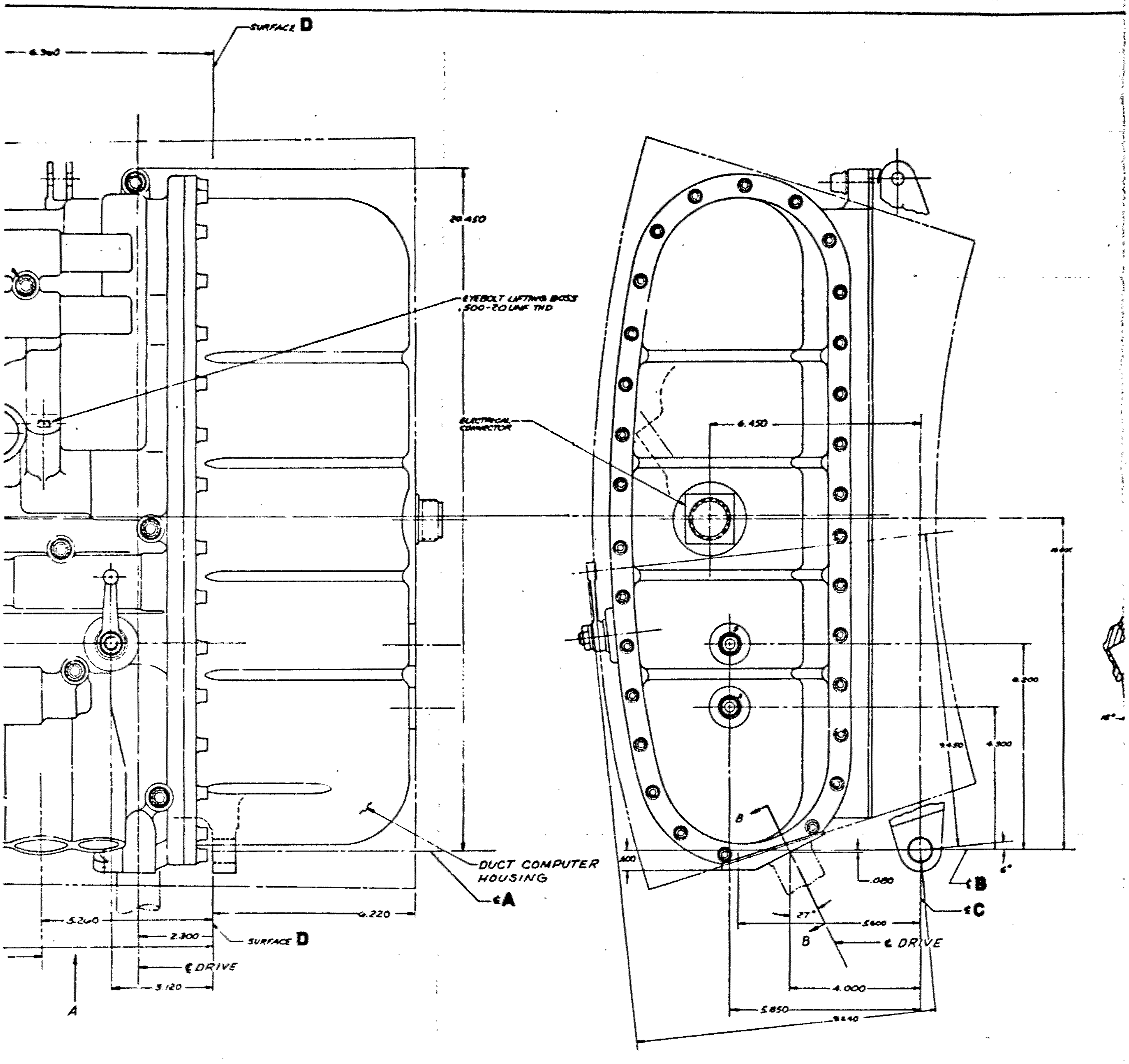
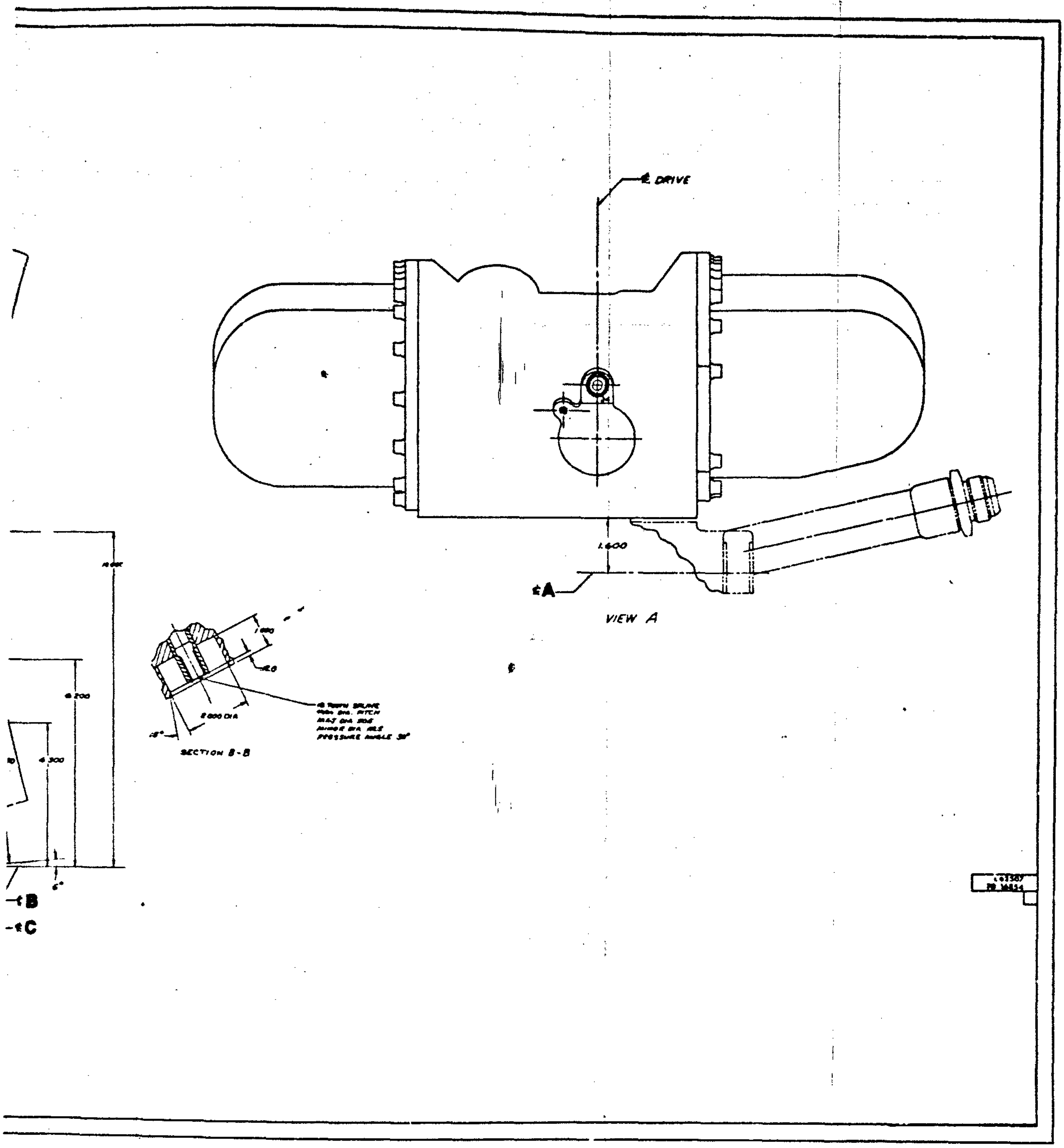


Figure 24. JTF17 Engine Unitized Fuel and Area Control Installation Layout - Hamilton Stand



SURFACE D
6.940
20.450
EYEBOLT LIFTING BOSS .500-20 UNF THD
ELECTRICAL CONNECTOR
6.450
DUCT COMPUTER HOUSING
℄ A
6.220
5.240
2.300
SURFACE D
℄ DRIVE
3.120
A
B
27°
5.600
B
℄ DRIVE
4.000
5.850
.080
℄ B
℄ C
6.200
4.300

2


℄ DRIVE
1.600
℄A
VIEW A
SECTION B-B
2.600 DIA
℄B
℄C

(7) Figure 25

This figure shows the mounting plate used with the Hamilton Standard unitized control.

8. Anticipated Problem Areas

The program for the unitized fuel and area control will encounter problems that will require a significant amount of development effort to eliminate. These problems are the result of the particular implementation of a concept, whether the concept is new or has been utilized in numerous other controls. The design of the control will utilize the experience obtained from previous control development programs. However, the following problem areas may still be encountered:

1. Rotating pilot valve life: Optimum pilot valve diametral clearance and materials will be determined through extensive endurance testing, particularly with hot fuel.
2. Optimum rotating pilot valve damping: A compromise between pilot valve response and servo stability will be determined for each of the various servos in the system.
3. Compensation for temperature effects on servo linkages and fluid viscosity to attain the desired servo accuracies: Hot fuel and ambient testing will empirically correct any errors in the calculated compensation required.
4. Main gas generator and duct heater metering valve pressure regulating valves sensitivity to flow and back pressure: Flow sensitivity which adversely affects metered fuel accuracy will be corrected with regulating valve contour changes resulting from empirical, analytical, and test data.
5. Duct airflow computer and duct nozzle control response and stability: Extensive dynamic bench testing and engine testing will be conducted to obtain satisfactory system performance.
6. Metering valve differential pressure sensor fuel temperature compensation: Fuel flow distribution will be investigated to assure that the fuel temperatures being imposed on the sensor are indicative of the fuel passing through the metering valve.
7. Servo linkage bending: Weight-accuracy compromises will be determined from empirical accuracy measurements.
8. System pressure fluctuations affecting main gas generator, duct metered fuel flow, and duct nozzle stability: Complete system component interaction resulting in system pressure fluctuations will be studied with the complete control system on the engine.

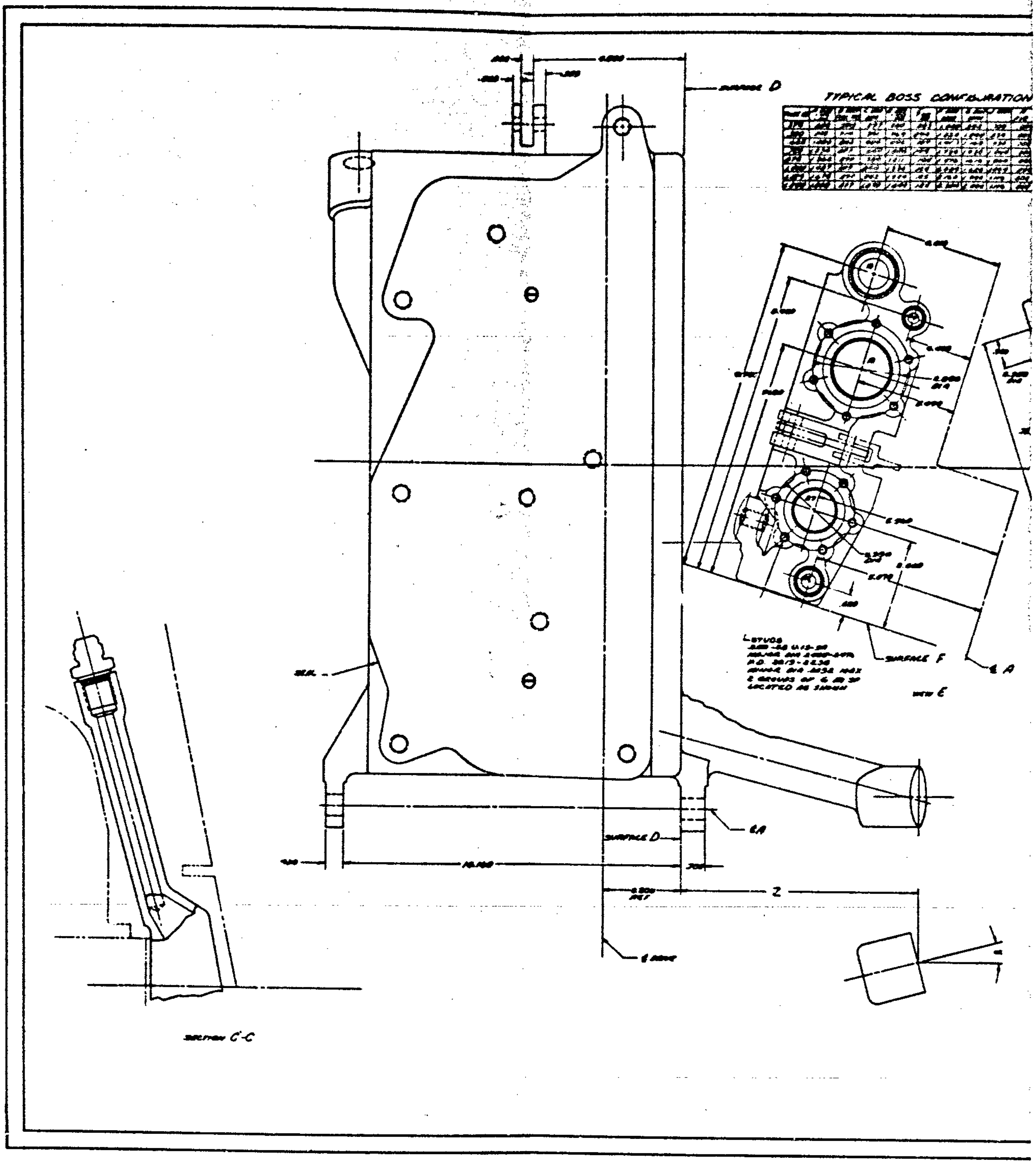
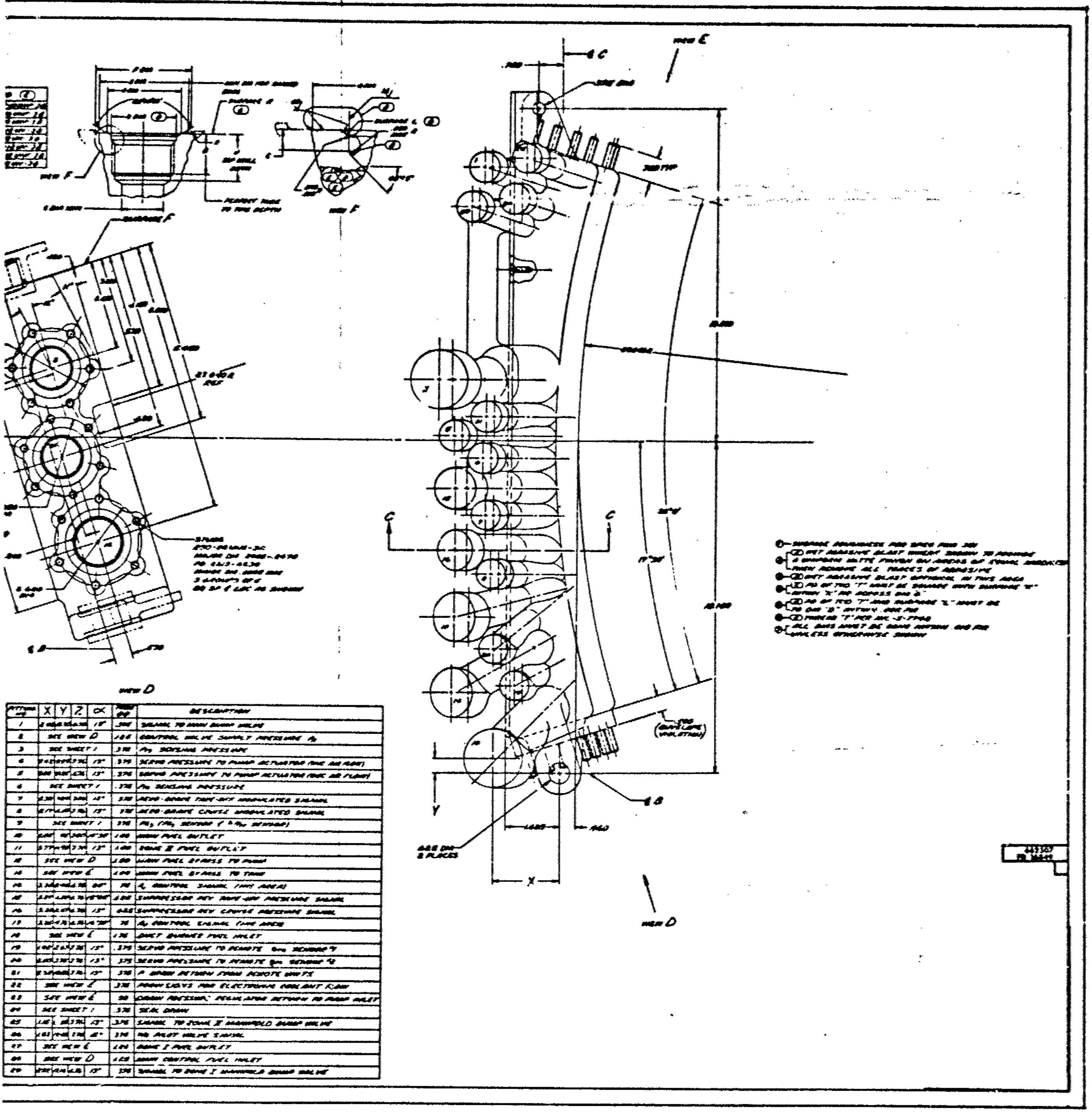


Figure 25. Mounting Plate Installation Layout (Unitized Fuel and Area Control - Hamilton Sta



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E. ENGINE PRESSURE RATIO (EPR) CONTROL

1. Reason for Control

Studies being conducted in Phase II-C to determine suitable power setting parameters for the JTF17 engine have shown engine pressure ratio (EPR) to be an excellent indication of nonaugmented thrust. Refer to Volume III, Report A for discussion of in-flight power setting. Analog simulations of takeoff and cruise power setting requirements have confirmed the usefulness of this parameter for power setting of both nonaugmented and augmented thrust. Although another parameter must be used to determine augmenter-lit thrust output, EPR will be the basic gas generator output indicator and as such it must be maintained within pre-established limits by the flight crew.

Recommended thrust setting procedures for the JTF17 engine, outlined in paragraph B, will call for use of EPR as the only engine parameter in setting nonaugmented thrust, and as one of two engine parameters in setting augmented thrust; augmented thrust will be set by a combination of EPR and total fuel flow. At any flight and engine condition the value of EPR required to set a desired thrust will be available to the flight crew in table and curve form.

If no EPR control is used, manual EPR trim adjustments by the flight crew are necessitated by three factors:

1. The unitized fuel control will be programmed to control fuel flow so as to produce a nominal EPR schedule when power lever is in a full nonaugmented position or above. Any deviations caused by control tolerances of the open loop hydromechanical unitized fuel control may be compensated for by manual adjustments by the flight crew.
2. The nominal EPR schedule used for fuel scheduling purposes represents the EPR required for an engine with fixed accessory loading and fixed compressor aircraft air bleed. Any deviation of these factors causes a change of EPR requirements that must be compensated for by the flight crew.
3. EPR must be adjusted at takeoff to compensate for airport altitude and again at the start of climb.

Use of the EPR control essentially eliminates the requirement for manual adjustments due to control system tolerances and adjustments due to changes in accessory loading, and compressor air bleed (items 1 and 2 above), although adjustments will still be required due to airport altitude at takeoff and climb (item 3 above). Therefore, overall magnitude and frequency of adjustments will be significantly reduced. A vernier knob marked "EPR Adjust" is available for EPR control schedule adjustments.

The airframe-mounted EPR control will be offered to the airlines as optional equipment. It is not required for satisfactory engine operation, but will reduce the flight crew load.

2. Description

The EPR Control is an airframe-mounted, digital electronic, all solid state device that is designed to operate as a closed loop around the unitized fuel control.

Figure 1 shows the general control layout and its external electrical connections. Inputs to the EPR Control will be from engine mounted temperature probes and from a manual vernier labeled "EPR Adjust" available to the flight crew. Output action will be accomplished by energizing an electric motor mounted on the unitized fuel control.

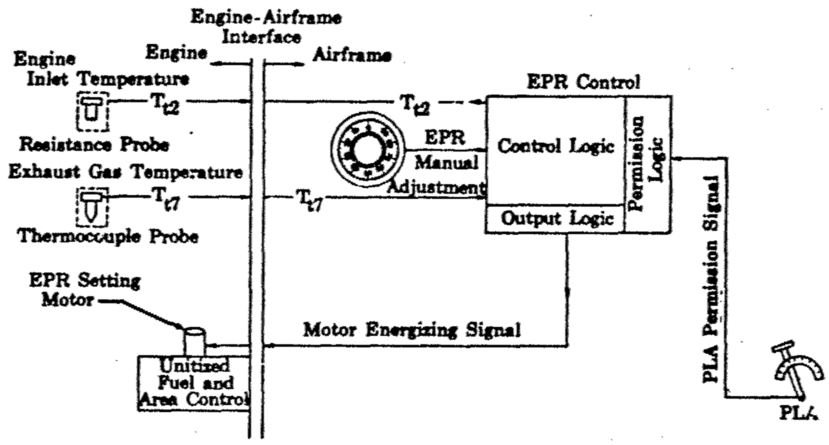

Figure 1. EPR Control Block Diagram FD 17155 BIII

Figure 2 is a more detailed logic diagram.

The EPR control will schedule a level of exhaust gas temperature which will maintain desired EPR within narrow limits. Adjustments to the nominal EPR schedule may be made by the flight crew by adjusting the "EPR Adjust" vernier knob.

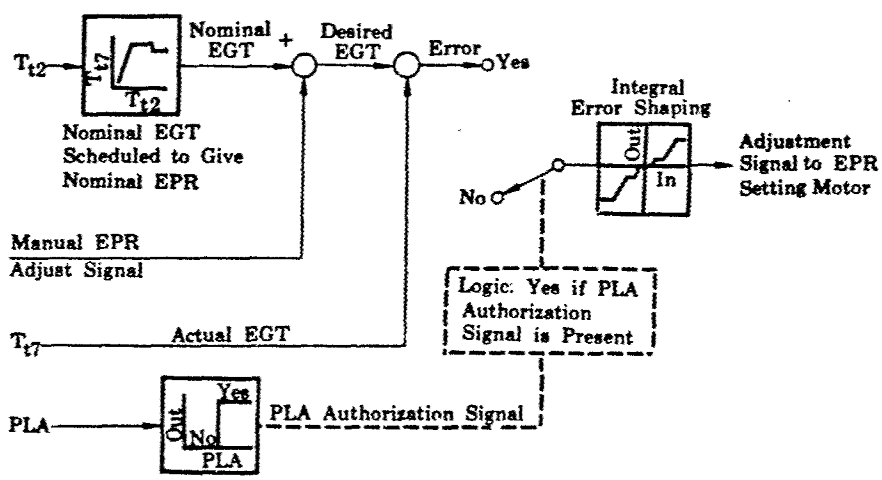

Figure 2. EPR Control Logic Schematic FD 17154 BIII

Figure 3 shows a photograph of a similar device being developed for J58 applications. Figure 4 shows a cutaway view of the same J58 device. Figure 5 is a photograph showing the mechanical construction and illustrating the small size of electronic subassemblies utilized in the J58 program.

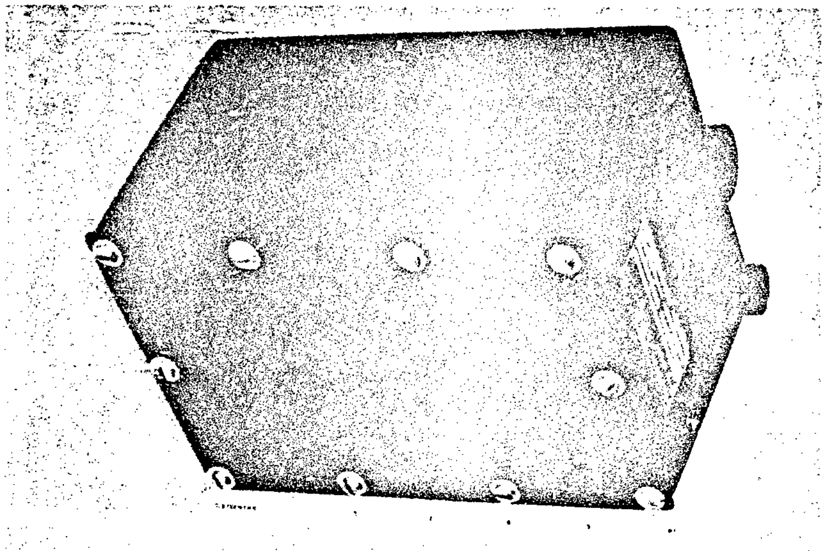

Figure 3. Mockup of J58 Digital EPR Control FE 61306 BIII

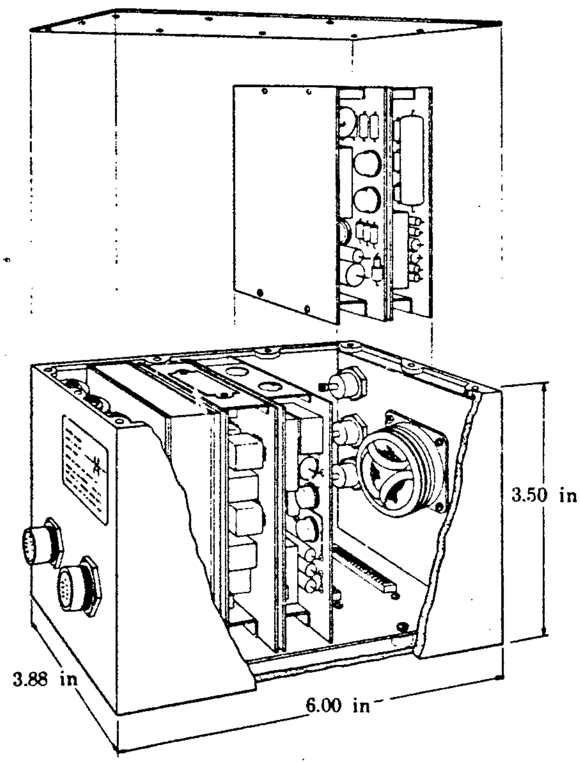

Figure 4. Sketch of J58 Digital EPR Control FD 17156 BII:

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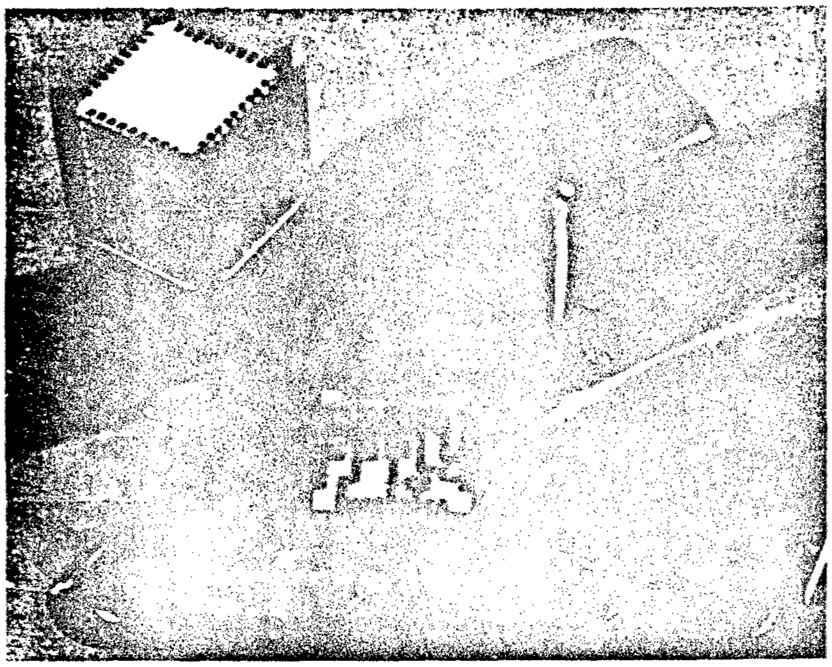

Figure 5. Typical Microcircuit Assembly FD 17153
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A "push-to-check" feature is being studied that will show equipment readiness for use and a manual over-ride switch will be provided.

3. Design Objectives

The design objectives for the EPR control are:

1. Reliability must be equivalent to the reliability of the hydro-mechanical control components.
2. Control must be capable of controlling EPR with sufficient accuracy to significantly reduce the need for flight crew adjustments to the system.
3. Control should provide protection against turbine over-temperature.

4. Design Requirements

1. The EPR Control shall hold the requested EPR schedule within a tolerance of $\pm$ 0.01.
2. The control shall operate satisfactorily at the ambient temperatures, pressures, and vibration levels consistent with the airframe environment. Specifically, the control shall operate satisfactorily in the environment as defined by MIL-STD-810 and MIL-E-5400 (Class II) and as defined by specific P&WA environmental specifications.
3. The power required shall not exceed 20 volt amp.
4. The volume shall not exceed 90 cubic inches.
5. The weight shall not exceed six pounds.

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5. Design Approach and Status

1. The EPR Control has an "authority" (i.e. range) slightly larger than the estimated tolerance band of the open loop hydromechanical unitized fuel control. Over this range it can effectively remove the tolerance errors of the control.
2. The EPR control utilizes exhaust gas temperature (EGT) as the controlling parameter. EGT is uniquely related to EPR and the desired schedule can be defined to a high degree of accuracy on an EGT basis. EGT has advantages as a scheduling parameter in that it automatically schedules changes in allowable EPR to compensate for changes in accessory loading and aircraft compressor bleed air.
3. Airframe mounting was selected because state-of-the-art electronics can be utilized with a minimum of development effort.
4. The solid state digital system was chosen because of its basic computational accuracy, small size, low weight, and reliability.
5. The design is based upon development of a similar control for the J58 application. The following summary shows the P&WA experience with controls of this design:

AIRFRAME MOUNTED EPR CONTROLS

Phase	Vendor	Description	Status
Flyable breadboard	P&WA	Electromechanical (Analog)	Initiated 1963. Seven units constructed. Engine and Flight Evaluation Tests successfully completed in 1964.
Prototype and limited production	Honeywell	Electromechanical (Analog)	Initiated 1965. Three of fifteen units delivered. Undergoing qualification tests and engine evaluation tests.
Flyable breadboard	HSD	Electronic-Solid State (Digital)	Initiated 1965. Three units delivered in 1966. Undergoing engine evaluation.
Prototype and limited production	HSD	Electronic-Solid State (Digital)	In proposal form. Present plans call for purchase of several units in 1966.

The design of this system is being coordinated with control vendors that have had experience with engine controls as well as with development of electronic components. These vendors are Hamilton Standard Division, United Aircraft Corporation, and Eclipse-Pioneer Division, Bendix Corporation. Both of these vendors have had

prior experience in designing and constructing electronic closed loop controls and both have had experience in the electronic and fuel control fields.

P&WA Purchase Specification PPS-J-113, which defined the technical requirements for the EPR Control, has been furnished to both vendors and is available from FRDC upon request. HSD has based a preliminary design of the solid state digital device on a configuration under development for J58 applications. (See figure 3.) Relatively minor changes to the circuit were required to meet JTF17 requirements. Bendix has produced preliminary design layouts based on P&WA requirements.

Technical proposals will be available in Phase II-C. Selection of a vendor will take place in Phase II-C and vendor work will be initiated immediately after start of Phase III.

6. Anticipated Problem Areas

Most of the problem areas have been solved by the extensive J58 engine development program. Remaining problem areas are discussed below:

1. Engine Temperature Probes and Electrical Harnesses
   Engine temperature and electrical harnesses are discussed in detail in Volume III, Report B, Section II of this proposal.

2. Mechanical Construction of Electrical Assemblies
   The mechanical construction of electrical assemblies will be improved by use of high density integrated circuits that require few inter-connecting leads.

7. Test Program

A detailed description of the development test plan and test schedule for the EPR control is included in the Component Test and Certification Plan of this proposal, Volume III, Report E, Section II.

F. GAS GENERATOR FUEL PUMP

1. Description

The gas generator pump is an engine-driven two-stage unit that incorporates a centrifugal boost stage in series with a single high pressure gear stage. The boost element supplies fuel to the high pressure gear stage, the hydraulic pump inlet, the duct manifold quick fill system, and the ignition exciters for cooling. The high pressure stage supplies fuel to the unitized control where it is properly metered before being injected into the gas generator combustor. A small amount of this flow is also used by the unitized control computer section to power hydraulic servos and generate hydraulic signals. A schematic of the pump is shown in figure 1.

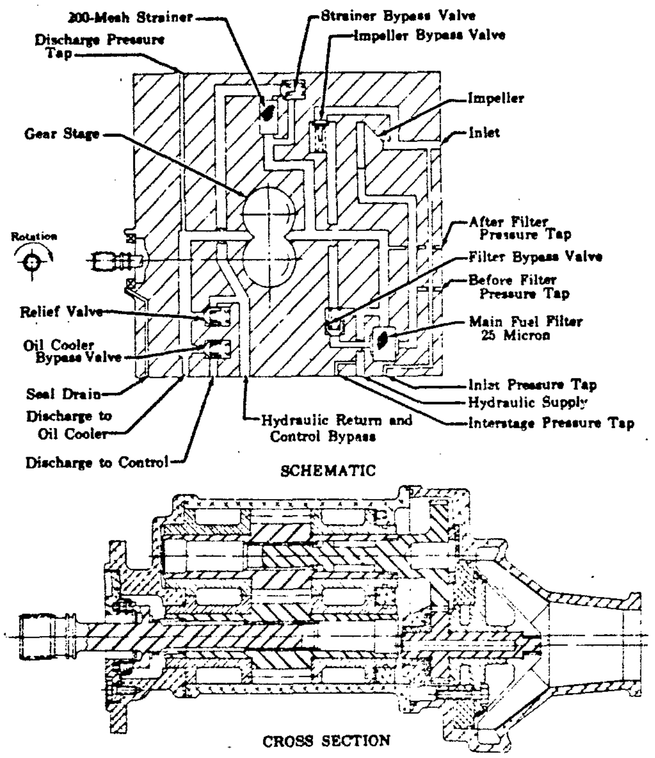

Figure 1. Gas Generator Fuel Pump Representation FD 16887 BIII

A 25-micron filter is incorporated at the boost stage discharge, and a 200-mesh screen is located within the hydraulic and control by-pass return flow path to the gear stage, the unitized control, and the fuel injection nozzles. Bypass valves are located in parallel with each of the filters to provide a flow path if the filters become contaminated. An indicator is incorporated which produces a visual indication if the 25-micron filter pressure drop approaches the bypass condition.

A bypass valve is incorporated in parallel with the boost stage that opens in the event of impeller blockage to provide a low restriction flow path to the high pressure section. This will permit the pump to continue to operate on the main stage alone.

A relief valve is included at the pump discharge which opens to prevent excessive pump discharge pressure in event of a downstream malfunction.

Two fuel outlets are provided on the pump. One is connected to the unitized control through the fuel oil cooler and the other outlet is connected to the control through a flow divider valve.

The flow divider valve directs total pump discharge flow through the fuel-oil cooler until the pressure drop across the cooler reaches 30 psi. All additional flow beyond that required to maintain the 30 psi is bypassed directly to the unitized control. This scheme reduces the size of the cooler and associated plumbing, resulting in a total system weight reduction.

Instrumentation pressure taps are provided at the pump inlet, filter inlet, filter discharge, and gear stage discharge. These pressure taps may be used to obtain signals for cockpit instrumentation.

The pump drive spline is lubricated by oil supplied under pressure from the engine oil system.

The pump design includes a quick disconnect adaptor plate to which all the external fuel connections, except the main inlet, are made. This feature permits the pump to be removed from the engine without disconnecting the associated plumbing, to assure the pump can be replaced on an installed engine in less than 30 minutes.

This particular type of pump has been selected to utilize P&WA's experience with commercial and military jet engines including the high Mach number J58 engine.

A mockup photograph and an installation layout are included as figures 2 and 3.

2. Design Objectives

The design objectives for the gas generator fuel pump are:

1. To require minimum field maintenance and be replaceable on an installed engine within 30 minutes
2. To be overhaulable at existing facilities without elaborate tooling
3. To have a TBO equal to or greater than that of the engine
4. To have a mean-time-between-failure and chargeable premature removal rate consistent with current commercial applications
5. To minimize development and production costs by utilizing previous development experience, economical manufacturing techniques, and materials that are inexpensive yet suitable for the purpose

6. To be of the lightest weight possible consistent with the requirements of maintainability, reliability, durability and safety
7. To incorporate an adequate margin of safety even in event of malfunction or failure
8. To meet the performance requirements of the engine throughout the complete operating range, including emergency conditions
9. To provide growth potential through minor modifications without sacrificing current efficiency.

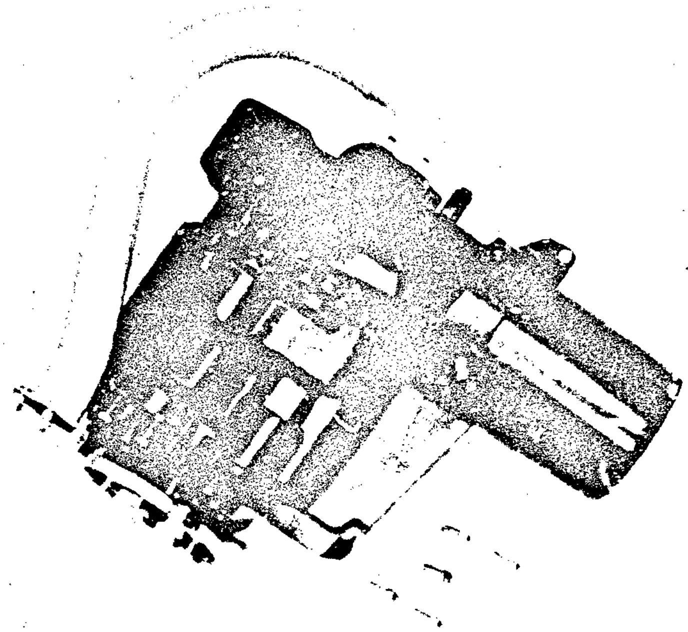

Figure 2. Gas Generator Fuel Pump

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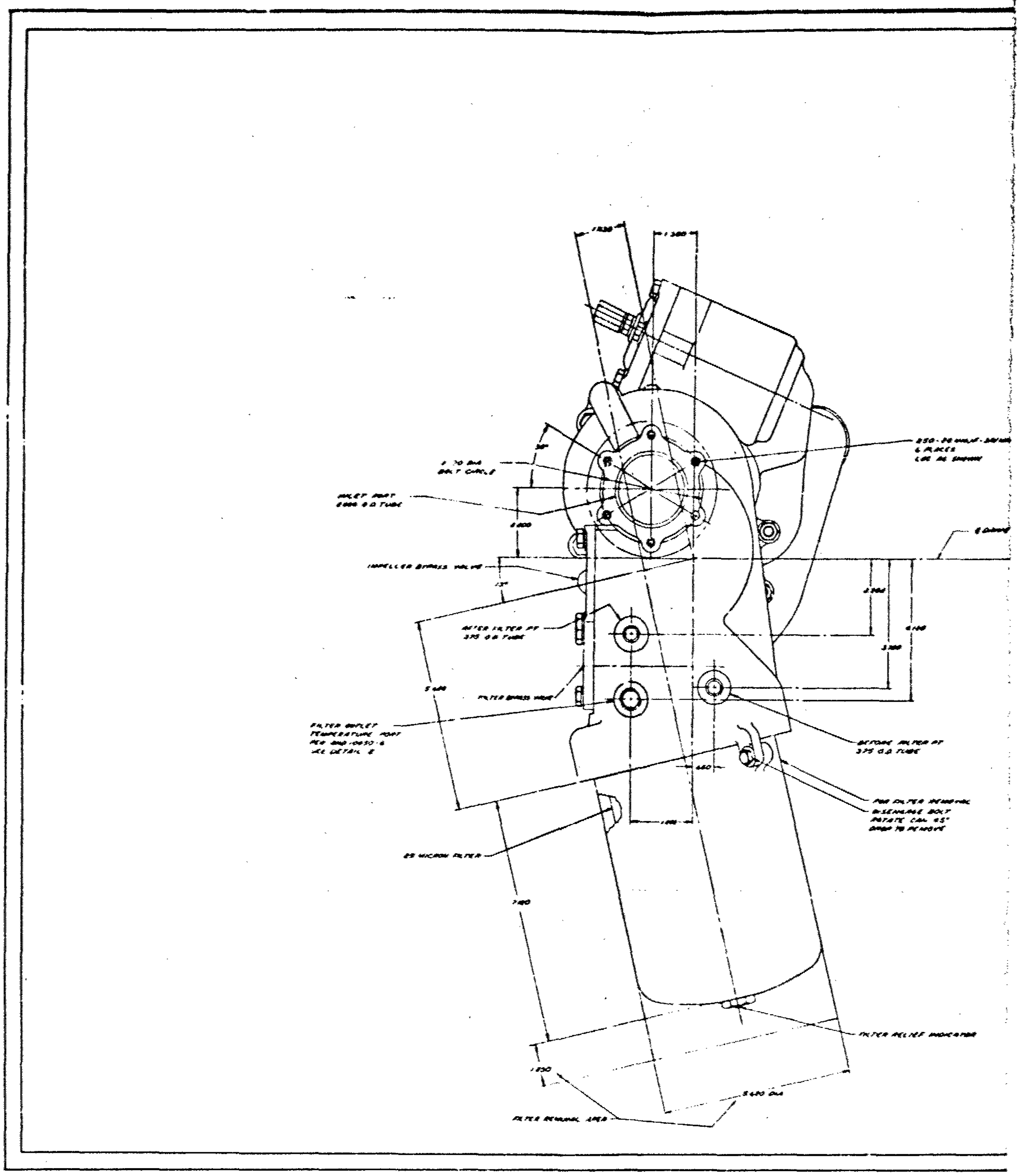
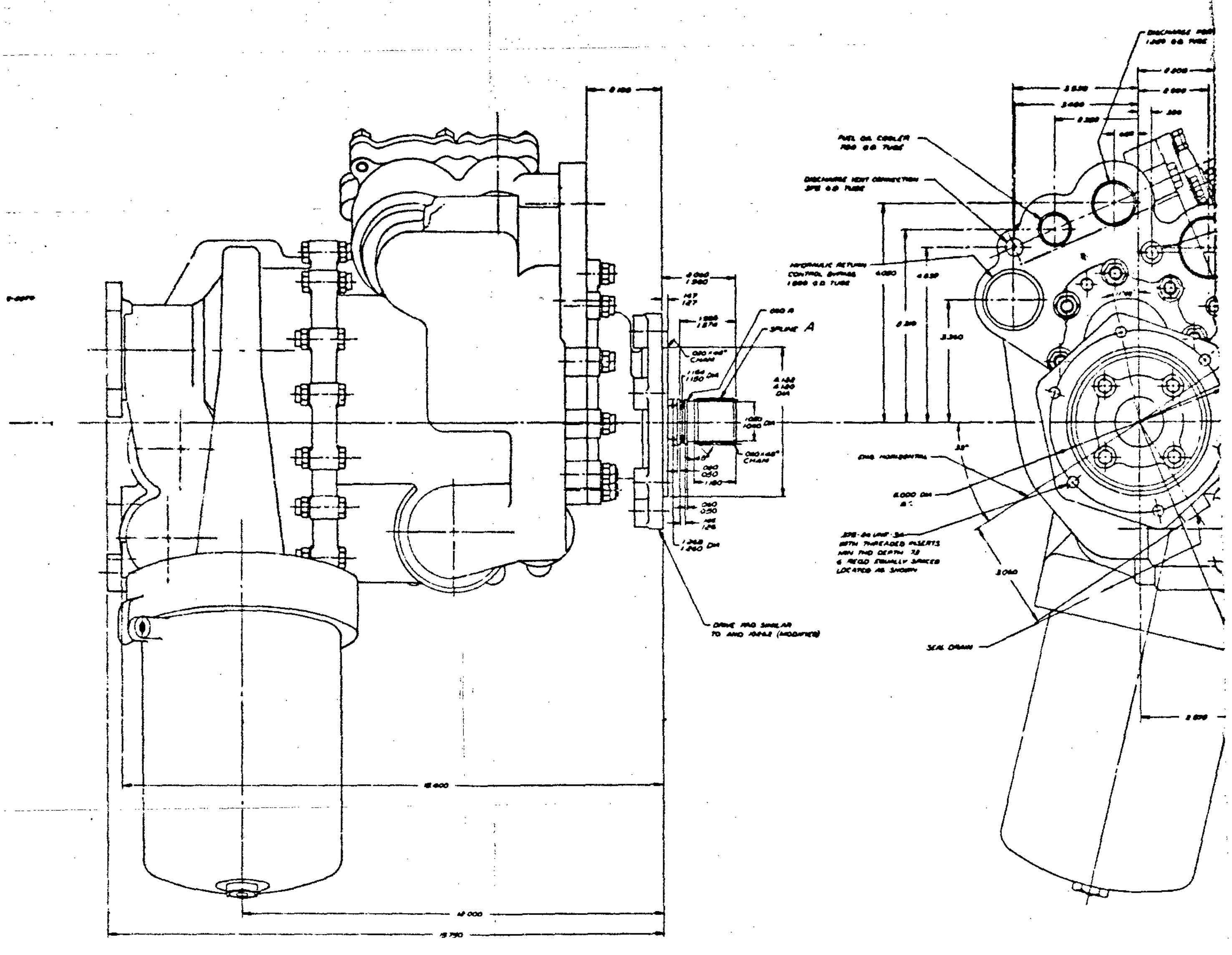


Figure 3. Gas Generator Fuel Pump Installation



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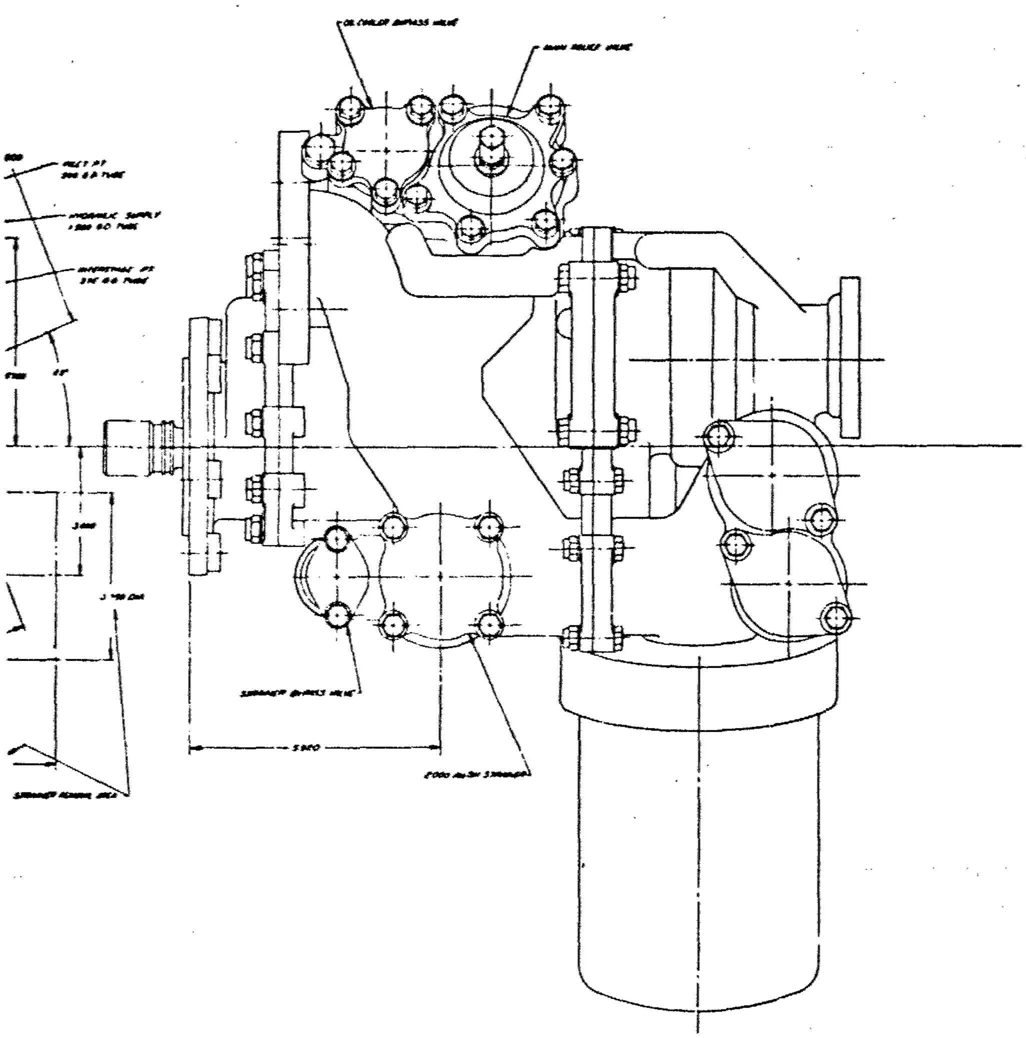
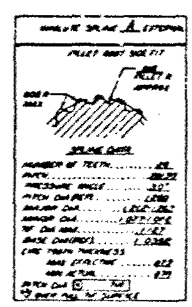

OIL COOLER BYPASS VALVE
MAIN RELIEF VALVE
HYDRAULIC SUPPLY
STRAINER BYPASS VALVE
45°


SPLINE DATA
NUMBER OF TEETH
PRESSURE ANGLE
30°
PITCH DIA
MAJOR DIA
MINOR DIA
BASE DIA
CIRC TOOTH THICKNESS

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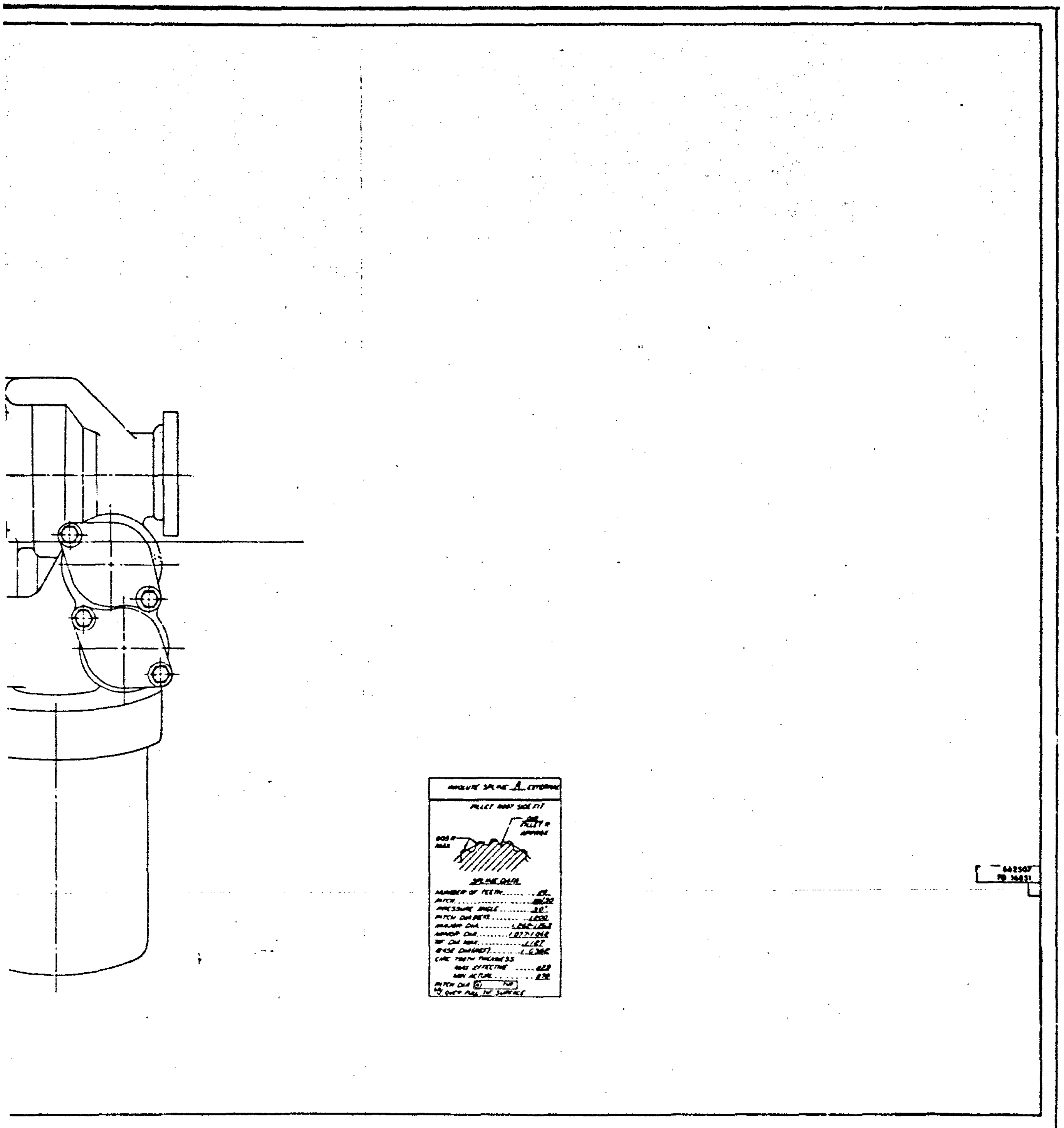

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3. Design Requirements

Procurement Specification J107 describes the technical requirements of the gas generator fuel pump and is available from the FRDC upon request.

Some of the more significant requirements are:

1. Fuel compatibility: The pump is designed for normal operation with PWA 522 (Jet A, A-1) and PWA 533 aviation kerosene
2. Fuel inlet pressure: The pump will operate satisfactorily under normal operating conditions with a minimum fuel inlet pressure of 5.0 psi above the true vapor pressure of the fuel and a maximum pressure of 50 psig. Under emergency conditions, with the aircraft boost pumps inoperative, the pump will operate for at least 10 hours with a vapor liquid ratio of 0.45.
3. Fuel inlet temperature: The pump is capable of operating within a fuel inlet temperature range between that temperature corresponding to 12 centistokes viscosity and a maximum temperature of 310°F.
4. Fuel discharge flow: The pump is capable of delivering fuel flow to 40,000 pph.
5. Ambient temperature: The pump will operate continuously within an ambient air temperature range of -65°F to 575°F, with intermittent duty to 650°F.
6. Fuel leakage: There will be no external fuel leakage from the pump and a maximum overboard drain fuel seepage of 10 drops per minute.
7. Fuel contamination: The pump will operate satisfactorily for a minimum of 80 hours on fuel containing the following contaminant:

Particle Size Microns	Percent of Total
0-5	39 ± 2 by weight
5-10	18 ± 3 by weight
10-20	16 ± 3 by weight
20-40	18 ± 3 by weight
Over 40	9 ± 3 by weight
Through a 200-mesh screen	100 by weight

8. Maximum operating speed: The maximum operating speed will be 4800 rpm.

4. Design Criteria

1. Housings: The major pump housings will consist of high temperature aluminum alloy castings designed for 7000-psi maximum stress. The material has a yield strength of 14,000 psi at 600°F. The aluminum used in current pump housings has a 14,000-psi yield strength at approximately 375°F.

2. Pumping Gears: The high pressure pumping element will use a set of 12-tooth gears. Production and development experience indicated the use of a 12-tooth gear pumping element will provide the best performance and pump life.
3. Gear Bearings: The pumping gear bearings are radially loaded to 650 psi. Present pumps have an average of 750 psi and the normal design limit is 1000 psi.
4. Impeller: The impeller will be of cast construction with back-venting. Back-venting the impeller reduces the thrust bearing loads. The impeller thrust bearing has a design load of 250 psi and incorporates cooling grooves on the journal diameter and the thrust face.
5. Seals: The shaft seal consists of a copper impregnated graphite body. The seal face is positively loaded against a highly machined flat surface on the drive shaft. This seal configuration, which is used on other commercial and high Mach number military applications, lends itself well to misalignment and varying loads.
6. Drive Shaft: The pump drive shaft is designed with a safety factor of 4. To improve shaft life, the drive spline is chrome plated and the splines are designed for 2:1 shear factor. The drive spline design wear factor is 1.3:1 compared to present designs with 1:1 wear factors.

5. Materials Summary

Typical materials used in the gas generator pump are identified in the following table:

Part	Material
Housings	Cast RR 350 Hiduminium
Impeller	Cast RR 350 Hiduminium
Pumping gears	M-4 Tool Steel or AMS 6470 Nitroloy
Pumping gear bearings	Leaded bronze
Shafts	AMS 6470 Nitroloy
Shaft seal	Pure Carbon Co., P59BH copper-impregnated graphite

6. Design Approach

This type of gas generator fuel pump has been selected to best utilize previous P&WA commercial and military engine experience. Similar pumps have been used on all P&WA turbine engines including the J58. The extreme temperature gradients across components of an engine in supersonic flight require emphasis on component housing design and on seal and bolt design to prevent leakage. The materials, techniques, and experience gained in the development of housings, seals, bearings, and pumping elements to satisfy the rigid requirements of the J58 will be used to assure the successful development of the pump.

J58 experience will be used to choose suitable materials and designs to preclude early wear problems in gears, journals, and bearings. For

example, a typical J58 pump development problem was gear "scuffing" at high fuel temperatures. The problem solution required a material change and a revised tooth form, both of which will be incorporated in the JTF17 design. P&WA experience in the sealing and lubrication of high temperature rotating shafts will be used in directing the vendor to a reliable drive shaft and seal design. Forced lubrication of the drive spline will be used to prevent excessive wear.

Chandler Evans Corporation, Thompson Ramo Woolridge, Inc., and PESCO have submitted proposals that include analytical and test data; layouts; descriptions; weight; cost; and reliability and durability estimates for the gas generator pump. These proposals are currently being evaluated with respect to the requirements stated above and also with respect to the more detailed requirements of the procurement specification. This evaluation procedure will continue, and a selection will be made prior to the end of Phase II-C. A contract with the chosen vendor will then be negotiated and work will proceed in accordance with the procurement specification at the Phase III go-ahead.

7. Anticipated Problem Areas

Previous experience indicates that considerable development effort will be required to prevent excessive wear of the drive splines, bearings, and pumping gears. Pressure lubrication of the drive spline and accurate control of drive pad alignment will be used to minimize the drive spline wear problem. Wear-resisting materials such as M-4 Tool Steel and AMS 6470 Nitroloy will be used to achieve maximum pumping gear life. Bearing loading will be minimized to provide longer life.

Boost impeller thrust bearing wear, resulting from variations in thrust loading, has been a problem in the past. Provisions for measuring impeller thrust are being incorporated into experimental and production pumps to establish the optimum thrust balance configuration and to ensure acceptable limits in all manufactured pumps.

8. Development Test Plan

A detailed description of the development test plan and test schedule is included in the Component Test Certification Plan of this proposal, Volume III, Report E, Section II.

G. DUCT HEATER FUEL PUMP

1. Description

The duct heater fuel pump supplies fuel, through the unitized fuel control and the fuel injection system, to the duct heater combustor where it is burned to produce thrust augmentation. A cross section drawing of the HSD unit is shown in figure 1. The pump assembly consists of an inducer boosted centrifugal pumping element that is driven by an axial flow air turbine. Use of this variable speed capability permits operation of the pump at reduced speed for most of the flight regime. The speed is modulated to produce only the pressure rise necessary to provide the duct heater fuel flow required for the specific altitude and Mach number conditions. The principle benefit of this system over a gearbox driven pump, which must be operated at higher than the required speed for most of the flight regime, is the significant reduction in heat added to the fuel by the pump. Secondary benefits are improved reliability and durability resulting from the reduction in operating time at maximum speed and stress.

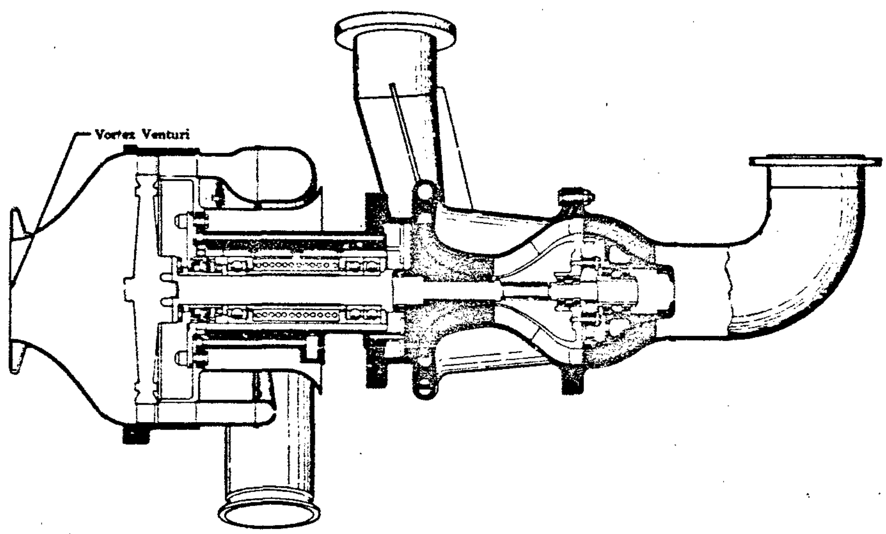


Figure 1. HSD Duct Heater Fuel Pump Cross Section — FD 13508A BIII

Turbine drive air is supplied from the compressor discharge bleed manifold and is regulated by a duct pump controller. This controller varies pump speed as required to produce only the output pressure demanded by the fuel control. The pump controller is described as part of the unitized control in paragraph D.

Overspeed protection is provided by a vortex venturi at the turbine discharge. This device, which does not require moving parts or a pump speed sensor, aerodynamically limits pump overspeed by creating a back pressure at the turbine discharge if an overspeed condition develops, thereby reducing the available turbine horsepower. Increased turbine discharge swirl angle associated with overspeed initiates a vortex that

produces an aerodynamic restriction to turbine discharge air flow. Tests of a substantially identical vortex venturi used on the J58 engine afterburner pump have demonstrated positive overspeed protection when all load is suddenly removed from the turbine.

The centrifugal pumping element is driven directly by the turbine through an interconnecting shaft. Fuel is force fed into the impeller by an inducer located upstream in the fuel inlet housing. The inducer is driven at one-sixth of turbine speed by a planetary-geared drive. The low speed inducer provides excellent pumping characteristics at very low fuel inlet pressure levels.

Fuel is used to lubricate and cool and bearings, seals, and inducer speed reduction gears. This feature eliminates the need for an external oil supply and scavenge system, and also precludes the possibility of depleting or diluting the engine oil supply in the event of a turbine end or impeller end shaft seal failure.

An alternative design proposed by Thompson Ramo Wooldridge is a similar type of pump except that the air turbine has been replaced by a Roots air motor and the upstream inducer is driven at one-half of the impeller speed. Aerodynamic overspeed limiting is utilized in this design also; however, it is necessary to generate the vortex discharge flow by exhausting the air tangentially into the vortex chamber.

A mockup photograph and an installation layout of the Hamilton Standard turbopump are included in figures 2 and 3.

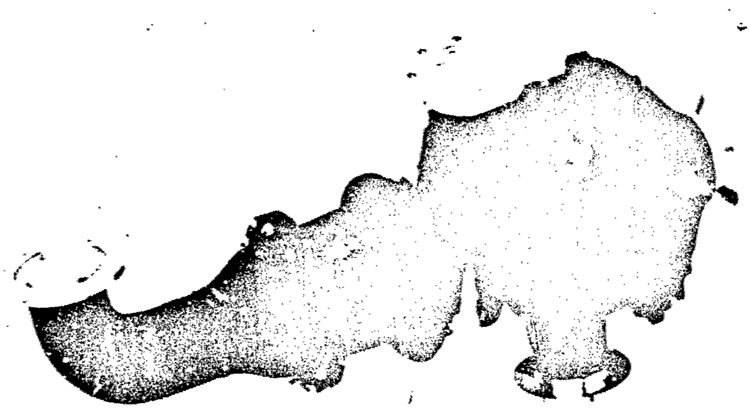


Figure 2. HSD Duct Heater Fuel Pump Mockup FE 55706 BIII

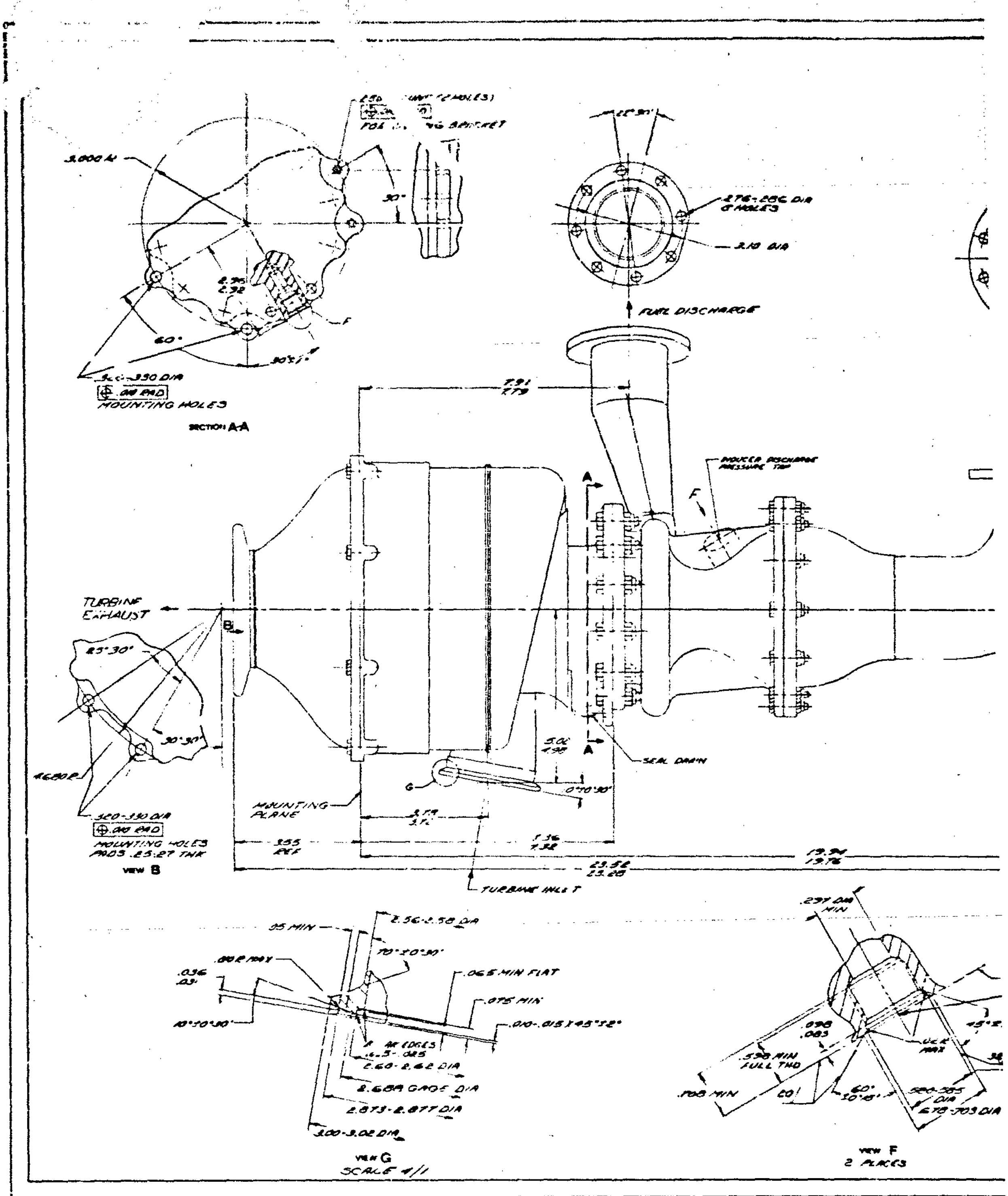
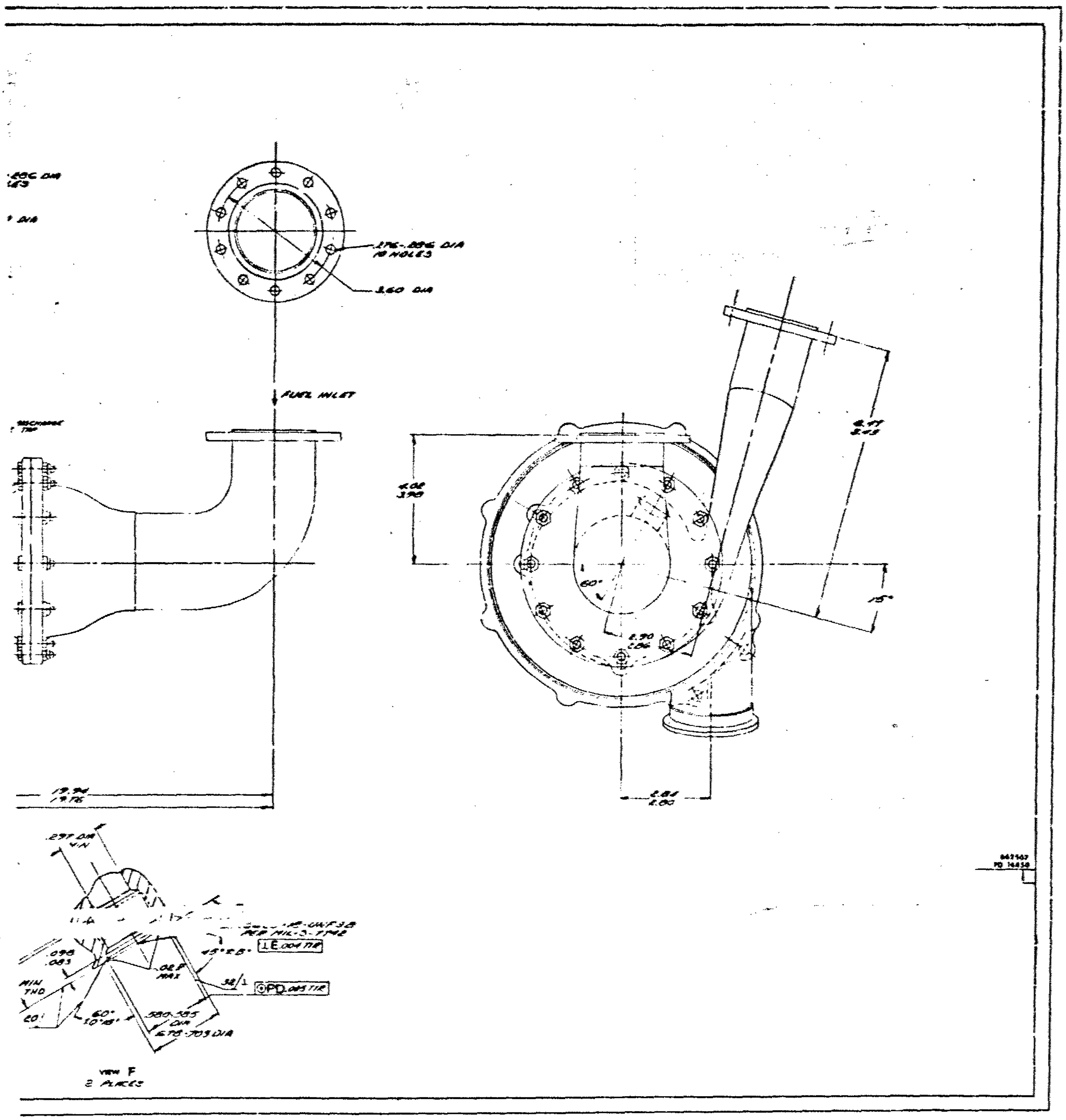


Figure 3. HSD Duct Heater Fuel Pump Installation Layout



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2. Design Objectives

The design objectives for the duct heater fuel pump are:

1. To require minimum field maintenance and be replaceable on an installed engine within 30 minutes
2. To be overhaulable without difficulty at existing facilities with standard tooling
3. To have a TBO equal to or greater than the engine
4. To have a mean-time-between-failure and chargeable premature removal rate consistent with current commercial applications
5. To minimize development and production costs by utilizing previous development experience, economical manufacturing techniques, and materials which are inexpensive yet suitable for the purpose
6. To be of the lightest weight possible consistent with the requirements of maintainability, reliability, durability, and safety
7. To incorporate an adequate margin of safety even in event of malfunction or failure
8. To meet the performance requirements of the engine throughout the complete operating range including emergency conditions
9. To provide growth potential through minor modifications without sacrificing current efficiency.

3. Design Requirements

The detailed design requirements for the duct heater fuel pump are defined in purchase specification PWA-PPS-J110, which is available from the FRDC on request.

Some of the more significant requirements are:

1. Fuel compatibility: The pump is designed for operation with PWA 522 (Jet A, A-1) and PWA 533 fuel.
2. Fuel inlet pressure: The pump will operate satisfactorily under normal operating conditions with a minimum fuel inlet pressure of 5.0 psi above the true vapor pressure of the fuel and a maximum pressure of 50 psig. Under emergency conditions, with the aircraft boost pumps inoperative, the pump is designed to operate for at least 10 hours with an inlet fuel vapor-liquid ratio of 0.45.
3. Fuel inlet temperature: The pump is capable of operating within a fuel inlet temperature range between that temperature corresponding to 12 centistokes viscosity and a maximum of 310°F.
4. Fuel discharge pressure: The pump is capable of producing fuel discharge pressure levels throughout the flow range as scheduled by the pump controller up to a maximum of 1150 psig.
5. Fuel discharge flow: The pump is capable of delivering cooling, metered, and servo flow within the range of 3000 to 97,000 pph.
6. Ambient temperature: The pump will operate continuously within an ambient air temperature range of -65°F to 575°F with intermittent duty to 650°F.

7. Fuel leakage: There will be no external leakage from the pump and a maximum overboard drain seepage of 10 drops per minute.
8. Fuel contamination: The pump will operate satisfactorily for 80 hours on fuel containing the following contaminant:

Particle Size Microns	Percent of Total
0-5	39 ± 2 by weight
5-10	18 ± 3 by weight
10-20	16 ± 3 by weight
20-40	18 ± 3 by weight
Over 40	9 ± 3 by weight
Through a 200-mesh screen	100 by weight

9. Air contamination: The drive turbine will operate satisfactorily when supplied with air containing three parts per million of engine lubricating oil and 0.065 grams of "fine Arizona road dust" per pound of air.
10. Overspeed protection: The drive turbine is positively limited at no load conditions to 150% of maximum normal rpm.
11. Turbine burst speed: The drive turbine burst speed at maximum turbine inlet temperature is at least 300% of maximum normal rpm.

4. Design Criteria

1. Housings: The major pump housings consist of castings which are designed to operate at 30% of the material yield strength. The housings are designed for 10^8 cycles at 10% of the maximum operating stress level. According to current standards this results in essentially infinite life.
2. Seals: Metallic static seals are used throughout. The carbon shaft seal is composed of silver-impregnated carbon backed with a tool steel mating ring. The seal face is sized for 14 psi maximum face pressure to achieve at least 6000-hour durability.
3. Bolts: All bolts have rolled rather than machined threads to reduce stress concentration. The number of bolts and torque limits at each location is established by the anticipated cyclic and vibratory stresses to provide ample margin for stress corrosion.
4. Turbine: The turbine disk and integral blades are machined from a single forging to assure maximum integrity. The design burst speed at maximum turbine temperature is 300% of maximum normal rpm. The turbine low cycle fatigue design goal is 100,000 cycles.
5. Inducer and Impeller: The inducer and impeller rotors both are investment castings. The inducer is driven at one-sixth of impeller speed to assure cavitation-free operation.
6. Bearings: The shaft bearings are selected to meet the anticipated radial and thrust loading. A factor of 0.1 is applied to the bearing manufacturers normal allowable loading data to compensate for the reduced lubricity of the fuel.
7. Gears: The gear teeth are designed for a bending stress not to exceed 50% of yield strength.
8. Growth Potential: A 10% flow growth has been designed into the pump.

5. Materials Summary

The materials used are the same as those which have demonstrated reliable service in the J58 afterburner turbopump and are presented below.

Part	Material
Housings	AMS 5362 Corrosion and Heat Resistant Steel
Shafts	AMS 5616 Greek Ascoloy
Inducer and impeller	AMS 5353 Corrosion and Heat Resistant Steel
Turbine	PWA 1005A Waspaloy
Bearings	M50 Tool Steel
Gears	AMS 6260 Wrought Alloy Steels
Bolts	AMS 5735 Corrosion and Heat Resistant Steel
Static seals	AMS 5508 Greek Ascoloy
Shaft seals	National Carbon Co. CDJ83 Carbon with M2 Tool Steel mating ring

6. Design Approach

The proposed duct fuel turbopump is a scaled-up version of the J58 afterburner pump with the addition of a low speed inducer to assure satisfactory operation at very low pump inlet pressure levels.

The fuel and air connections have been located to offer maximum accessibility within the available envelope. A quick disconnect is provided for the turbine air supply to permit replacement of the unit on an installed engine in less than 30 minutes.

The maximum normal turbine rotor speed is 27,500 rpm. The vortex venturi overspeed device limits turbine overspeed to 41,000 rpm under no load conditions and maximum turbine inlet temperature. The turbine is designed for a burst speed in excess of 85,000 rpm. A spin test will be performed on each turbine wheel to assure its structural integrity.

The pump assembly incorporates seal configurations that were selected during the J58 engine development program and have since proved to be highly reliable.

In the event of malfunction or complete pump failure, the most serious engine effect will be loss of augmentor thrust.

Proposals received from [illegible] and Thompson Ramo Wooldridge, Inc. will be evaluated with respect to the requirements stated above and the more detailed requirements of the purchase specification. This evaluation procedure will continue throughout Phase II-C of the SST program. Also, vendor coordination will continue as required to assure that any changes or revisions resulting from the Phase II-C testing are incorporated into the basic design. The duct pump vendor will be selected prior to the end of Phase II-C, and work will continue at the start of Phase III without interruption.

7. Anticipated Problem Area:

The planetary speed reduction gear train used to reduce inducer speed is the only new feature within the pump package. Reasonable design margins have been allowed. Early subassembly bench tests are scheduled at the vendor facility to determine and correct problem areas.

Previous experience with the J58 turbopump indicates that carbon shaft seal leakage resulting from seal face wear during extended operation will probably be encountered; therefore, an improved configuration with increased design margin is being incorporated. A seal face pressure loading of 26 psi was used in the J58 pump design while the SST pump shaft seal loading has been reduced to 14 psi.

8. Development Test Plan

A detail description of the development test plan and test schedule is included in Volume III, Report E, Section II.

H. HYDRAULIC PUMP

1. Description

The hydraulic pump is an engine-drive, reciprocating multiple piston fuel pump that is utilized to provide the engine hydraulic system with fuel at the required flow rates with a pressure rise across the pump of 1500 psi. The pump design is based on the design of pumps successfully qualified and used on the J58 and TF30 engines.

Integral and proportional servo valves control the pump cam plate to maintain the constant 1500 psi pump discharge pressure. The variation in hydraulic fuel flow necessary to position and control the duct nozzle area and the reverser-suppressor is met by varying the stroke of the pump pistons.

The Bendix Corporation, Utica Division design, as a representative design, is described below and is illustrated in figure 1. Two rotors that are driven and supported by a common shaft, each having nine equally spaced pistons, are reciprocated by a common nonrotating cam plate, through piston shoes. Auxiliary cam plates, which are loaded by rotor springs, hold the piston shoes against the cam plate at all times, assisting return of the pistons during the suction strokes. The geometry of the spherical cam plate face and convergent piston axes significantly reduces the side loading applied to the pistons when they are in the extreme retracted position. This design feature also takes advantage of centrifugal force to help retract the pistons and minimizes the pump volume by reducing the diameter of the valving interface.

The cam plate angle controls the displacement capacity of the pump. The cam plate is supported by two trunnion bearings on an axis that is located on a diameter of the cam plate. A shimmed stop screw limits the maximum cam plate angle to provide the desired maximum piston displacement.

Variable delivery at constant pressure is provided by controlling the cam plate angle with two concentric actuator pistons that act in opposition to a return spring. The spring drives the cam plate toward full stroke and provides the required rapid response to demands for increased fuel flow rate. The pistons respond to pump output pressure level as sensed by their respective control valves. The two control valves provide integral and proportional control for stable operation and fast response throughout the required operating regime of this application.

Each of the sensing control valves modulates the pressure to its actuator piston. The physical arrangement of the actuator pistons provides a "summing" action without structural linkages.

The inner piston is controlled in virtually an integral action: its control valve is nulled by only the balance of an adjustable reference spring and the reaction of pump delivery pressure on the end of the valve.

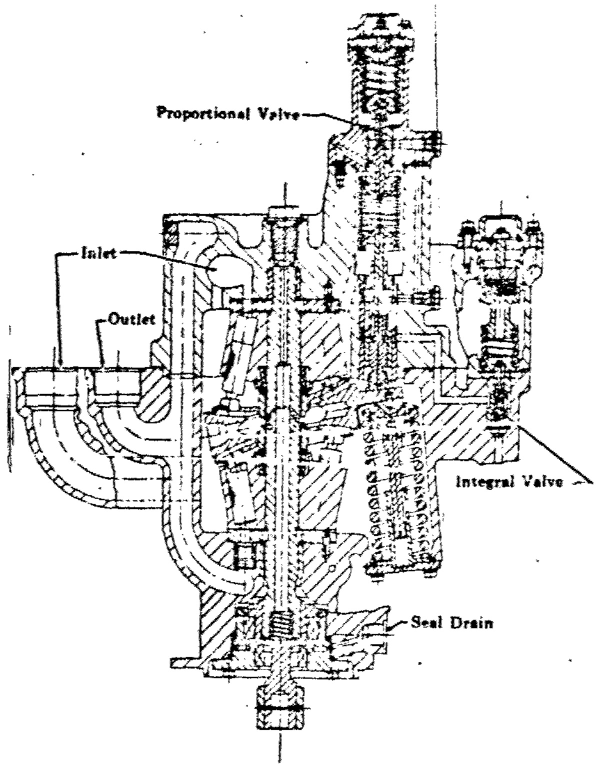

Figure 1. Bendix Hydraulic Fuel Pump Cross Section FD 13543A BIII

The outer actuator piston is controlled in a similar manner, but in a proportional action: its control valve senses piston position through a feedback spring as well as delivery pressure and a reference spring force.

The combined action of these servo-circuits is proportional in any transient.

Sealing and valving of the main flows of the pump are provided by ported insert plates located at the face of each rotor.

Hydraulic pump discharge fuel flows through an integral, 10-micron full-flow filter, with a differential pressure actuated bypass, which relieves if the pressure drop through the filter exceeds 20 psi.

The external drive spline is forced-oil-lubricated by the engine oil lubrication system.

A photograph of the pump mockup is shown in figure 2, and the installation drawing is shown in figure 3.

2. Design Objectives for the Hydraulic Fuel Pump

The following is a list of the design objectives for the hydraulic fuel pump:

1. To require minimum field maintenance
2. To be replaceable on an installed engine within 30 minutes
3. To be capable of overhaul without difficulty at existing facilities without special tooling
4. To have a TBO equal to or greater than the engine TBO
5. To have a mean-time-between-failure and chargeable premature removal rate consistent with current commercial applications
6. To minimize development and production costs by utilization of previous development experience, economical manufacturing techniques, and materials that are inexpensive yet suitable for the purpose
7. To be of the lightest weight possible consistent with the requirements of maintainability, reliability, durability, and safety
8. To incorporate an adequate margin of safety in the event of malfunction or failure
9. To meet the performance requirements of the engine throughout the complete operating range including emergency conditions
10. To provide growth potential through minor modifications without sacrificing efficiency.

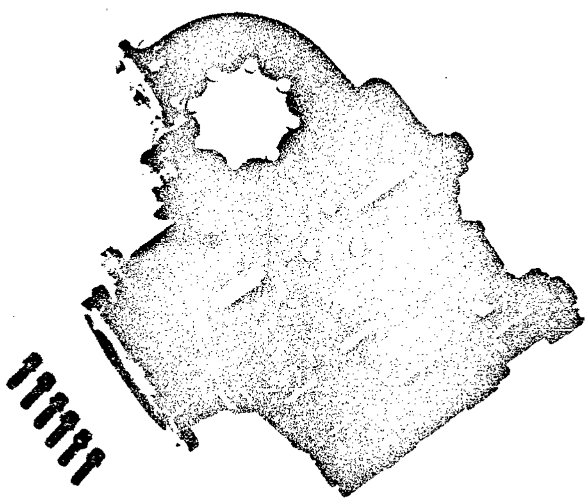

Figure 2. Bendix Hydraulic Fuel Pump Mockup

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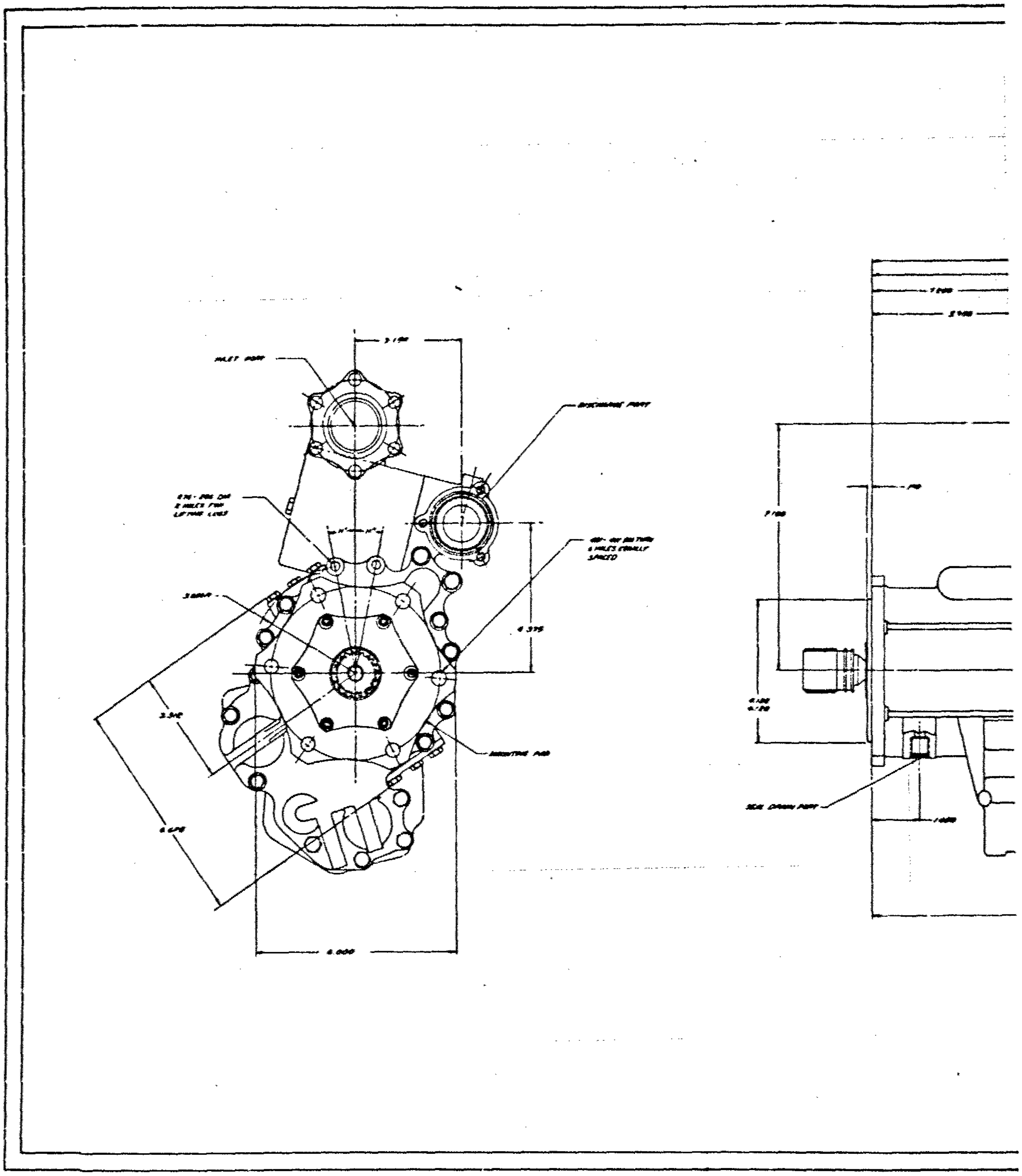
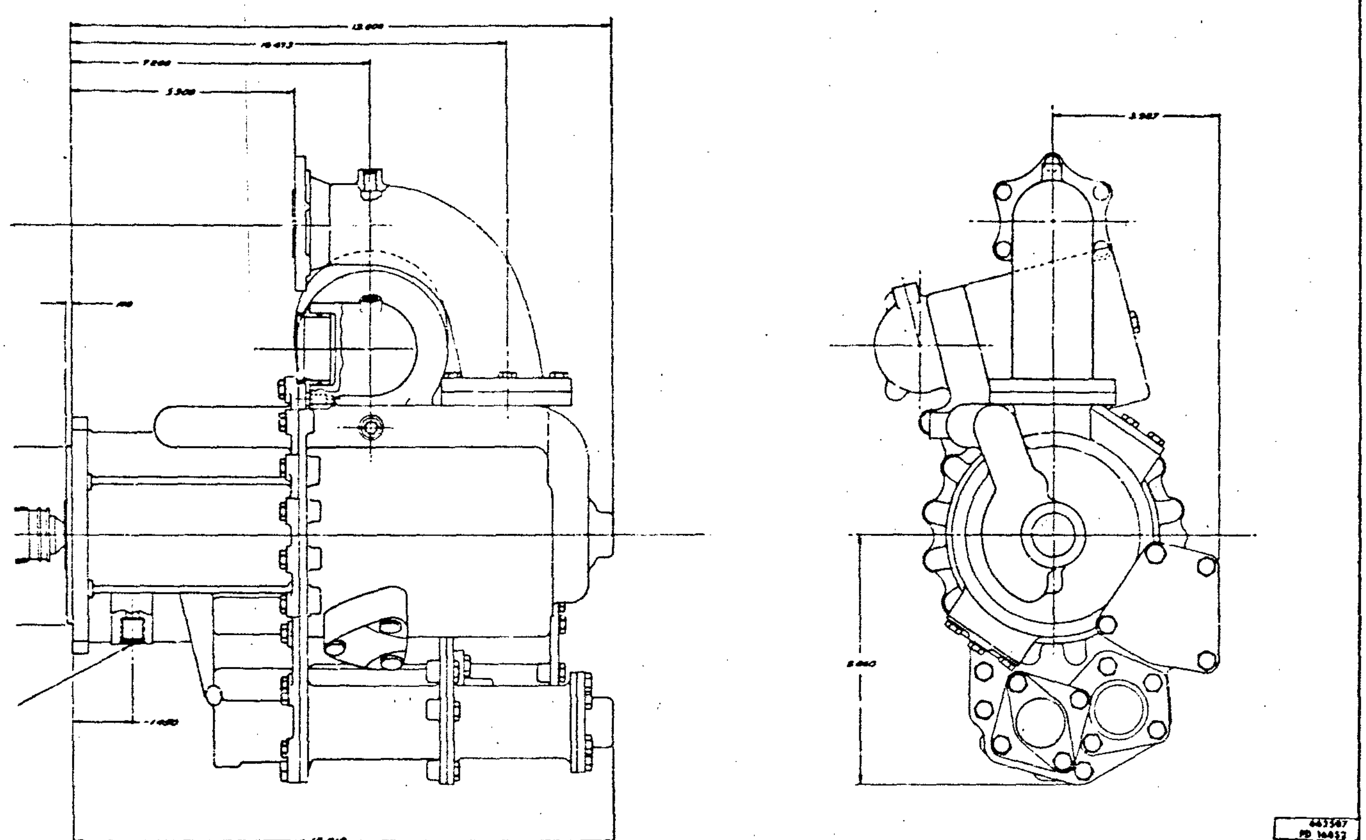


Figure 3. Bendix Hydraulic Fuel Pump Installation



3. Design Requirements

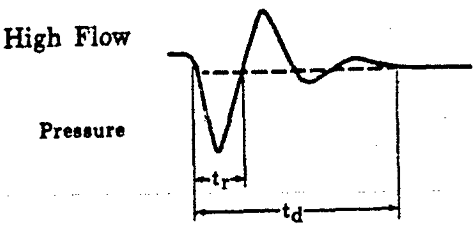
Procurement Specification J-109 describes the technical requirements of the hydraulic pump and is available from the FRDC upon request.

1. Materials and processes: Materials and processes used in the manufacture of the pump will be of high quality and suitable for the purpose.
2. Fuel compatibility: The pump is designed for normal operation with PWA 522 (Jet A, A-1) and PWA 533 aviation kerosene.
3. Fuel inlet pressure: The pump will operate satisfactorily under normal operating conditions with a minimum fuel inlet pressure of 35 psi above the true vapor pressure of the fuel and a maximum fuel inlet pressure of 225 psia. The pump will operate satisfactorily for a limited time with a full inlet pressure of 10 psi above the true vapor pressure of the fuel.
4. Fuel inlet temperature: The pump is capable of operating with a fuel inlet temperature range between that temperature corresponding to 12 centistokes viscosity and a maximum temperature of 325°F.
5. Fuel discharge pressure: The pump will be capable of maintaining a nominal discharge pressure above inlet pressure of 1500 psi throughout the required flow range.
6. Pump discharge flow and speed: The pump will be capable of delivering 55 gpm flow at the maximum rated speed of 4600 rpm.
7. Ambient temperature: The pump will operate continuously within an ambient air temperature range of -65°F to 575°F with intermittent duty to 650°F.
8. Fuel leakage: There will be no external leakage from the pump and the maximum overboard drain seepage will not exceed 10 drops per minute.
9. Fuel contamination: The pump will operate satisfactorily for 80 hours on fuel containing the following contaminant:

Particle Size Microns	Percent of Total
0 - 5	39 ± 2 by weight
5 - 10	18 ± 3 by weight
10 - 20	16 ± 3 by weight
20 - 40	18 ± 3 by weight
Over 40	9 ± 3 by weight
Through a 200-mesh screen	100 by weight

10. Fuel filtration: The pump will incorporate a 10-micron nominal filter at the discharge. The filter element will be accessible for easy and rapid replacement.
11. Response characteristics: The step-input response characteristics will meet the limits defined by figure 4. At steady-state condition, the peak-to-peak discharge pressure fluctuations will not exceed 80 psi at 4450 rpm.
12. Temperature rise: At a flow of 15 gpm, 4600 rpm, and with a discharge pressure rise of 1500 psi, the temperature rise across the pump will not exceed 20°F.

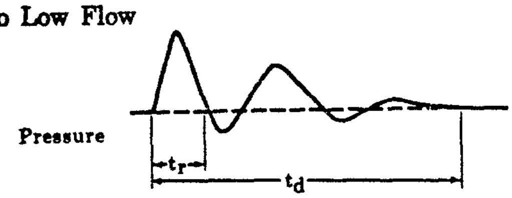
Low to High Flow



LIMITS:

Maximum Pressure Surge - 700 psi
t_r - Time Response - 0.060 sec
t_d - Total Damping Time - 0.50 sec

High to Low Flow



Note: Above response limits, assume a load volume of 90-100 cu in., a step change in flow of 20 gpm, and a nominal pressure rating of 1650 psig

Figure 4. Hydraulic Pump Step-Input Response Limits — FD 16821 BIII

4. Design Criteria

1. Bearings: The thrust, pintle, and radial bearings are designed for a minimum life of 7800 hours. This life is predicted by the standard "Anti Friction Bearing Manufacturers Association" method and the calculated value was divided by a factor of 10 to ensure a high degree of reliability.
2. Housings: The housings are designed for a minimum safety factor of 3 in shear and 5 in tensile stress at maximum temperature.
3. Rotating group: The rotating group is designed for a minimum safety factor of 2.5.
4. Bolts: Bolts are designed for a minimum safety factor of 3.
5. Springs: Springs are designed for a minimum safety factor of 2.

5. Material Summary

The materials used, in general, are based on successful performance in current applications and are identified below.

Part	Vendor	Material
Housing	Bendix	AMS 5355A Corrosion and Heat Resistant Steel
	Vickers	
	Pesco	RR-350 Hidumunium (aluminum alloy)
Shaft	Bendix	AMS 6485 Low Alloy Steel
	Pesco	M-2 Tool Steel
	Vickers	AMS 5616 Greek Ascoloy

Part	Vendor	Material
Piston	Bendix	AMS 6485 Low Alloy Steel
	Pesco	Tungsten Carbide
	Vickers	H-13 Steel
Piston shoe	Bendix	Berylco-10-Silver Plated
	Vickers	AMS 6470 Nitroloy - Silver Plated
	Pesco	Tungsten Carbide
Port Plate	Bendix	E-135 Nitroloy
	Pesco	Tungsten Carbide
	Vickers	M-2 Tool Steel
Piston Block	Vickers	H-13 Tool Steel
	Pesco	Tungsten Carbide
	Bendix	Lesco BG 42
Spring	All vendors	AMS 5673 Corrosion and Heat Resistant steel wire
Valve	Pesco	M-4 Tool Steel
	Vickers	M-2 Tool Steel
	Bendix	007-570 (D-2)
Shaft seals	All vendors	P-5AG Carbon - Silver Plated

6. Design Approach

Proposals have been received from three vendors, Bendix Utica, Pesco, and Vickers, in response to P&WA preliminary procurement specifications for a hydraulic pump for the JTF17 engine. All three of these vendors are experienced in the design, development, and operation of high pressure hydraulic pumps that utilize engine fuel as the working fluid. Pumps have been designed and developed by these vendors for the J58 and TF30 engines. The pump vendor will be selected by the end of Phase II-C and directed to proceed in accordance with the requirements defined in the P&WA procurement specifications. P&WA will then utilize the vendor coordination procedures established through successful experience that includes engineering guidance, parallel development programs, scheduled conferences, and adherence to the engineering change system defined in Volume V, Section III to design and develop the pump to provide the desired performance, reliability, and durability.

The design approach taken by the vendors is to utilize existing qualified designs as much as possible and minimize unit loadings to achieve the required pump life.

Existing fuel hydraulic pumps operating on the J58 and TF30 engines are qualified for application at 2500 and 3000 psi pressure rise, respectively. The hydraulic pump for the JTF17 engine will operate with a pressure rise of 1500 psi. Both the J58 and TF30 pumps are qualified at 350°F fuel inlet temperature; the pump for the JTF17 engine will operate at 325°F fuel inlet temperature.

The reduced operating pressures and temperatures noted above will permit development of the fuel hydraulic pump for the JTF17 engine to a level of reliability and durability equivalent to that of the piston-type main fuel pumps and the aircraft hydraulic pumps currently in airline service.

7. Development Test Plan

A detail description of the development test plan and test schedule is included in the Component Test Certification Plan of this proposal, Volume III, Report E, Section II.

I. IGNITION SYSTEM

1. Description

The JTF17 ignition system is composed of two fuel-cooled 400-cps alternating current powered exciters and four shunted surface gap igniters electrically connected to the exciters by flexibly shielded low tension electrical cables. Each of the exciters contains two capacitance discharge type independent electrical circuits. Each circuit produces a 4-joule, 3000-volt electrical output to fire the igniter. One circuit in each exciter fires a gas generator igniter while an associated circuit fires a duct heater igniter. To extend the useful service life of the igniters, an inductance type voltage booster is incorporated into each circuit just prior to the exciter discharge. The exciter circuits are shown in figure 1. The igniter electrodes and the igniter gap shunt material will erode to some degree after extended service, causing an air gap to be created between the electrodes and the shunting material. An increased voltage is then required to ionize the air in the gap to fire the igniter. Under this condition, the exciter voltage output is increased by the voltage booster as required to fire the igniter up to 6000 volts maximum.

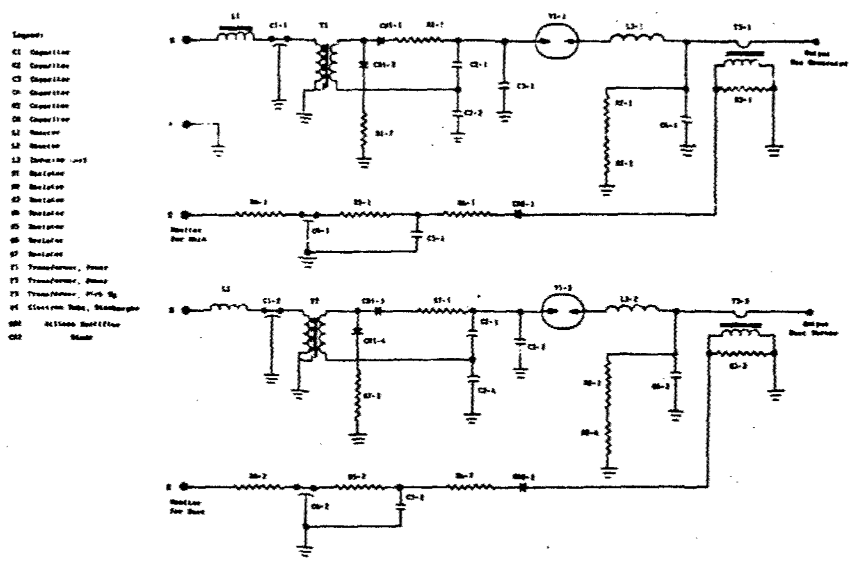


Figure 1. GLA Ignition Exciter Wiring Diagram FD 16859 BIII

Also included in each exciter circuit is a voltage signal generator that produces a signal voltage whenever the exciter delivers an output electrical discharge to the igniter. The electrical signal indicates that the exciter is transmitting spark electrical energy to the igniter and is provided for monitor and checkout purposes. A separate aircraft switch is required to checkout the duct heater ignition system at altitude conditions prior to descent. A power lever switch in the unitized control limits such checkouts to minimum nonaugmented operation. Gas generator ignition system checkout at altitude is accomplished using the aircraft ignition switches.

Two vendors have proposed an ignition system for use on the JTF17 engine. These vendors are General Laboratories Associates and Bendix, Scintilla Division.

The General Laboratories ignition system is shown on the mockup engine in figure 2. Detail features of the components of the ignition system are shown in figure 3 for the exciters and figures 4 and 5 for the electrical leads. A cross section partial engine schematic, figure 6, shows the engine installation requirements of the gas generator igniter (figure 7) and duct heater igniter (figure 8).

A basic airframe-to-engine ignition system electrical schematic is shown in figure 9. This illustrates the type of system that will be used to provide the flexibility of ignition system operation required to select dual ignition for starting and single ignition during operating periods when ignition "on" may be desirable, such as takeoff and climb or descending through turbulent weather conditions. This system also permits alternating operation of the two gas generator units to prolong service life under such conditions.

An automatic restart switch will be offered as optional equipment for the gas generator ignition system. This switch is designed to automatically turn on the ignition system if a flameout occurs as a result of compressor airflow interruption induced by turbulence, ice, or foreign object ingestion. The automatic restart switch provides the same protection to the engine as provided by manually selected continuous ignition. This switch eliminates any need for operating the ignition system on a continuous basis during the critical periods described above. The reduction in use increases the service life of the ignition system with consequent savings in overhaul costs.

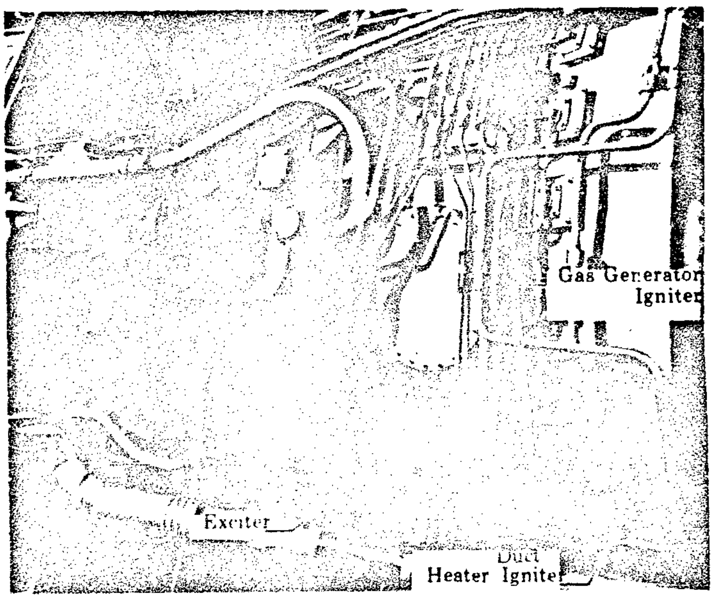


Figure 2. Ignition Installation

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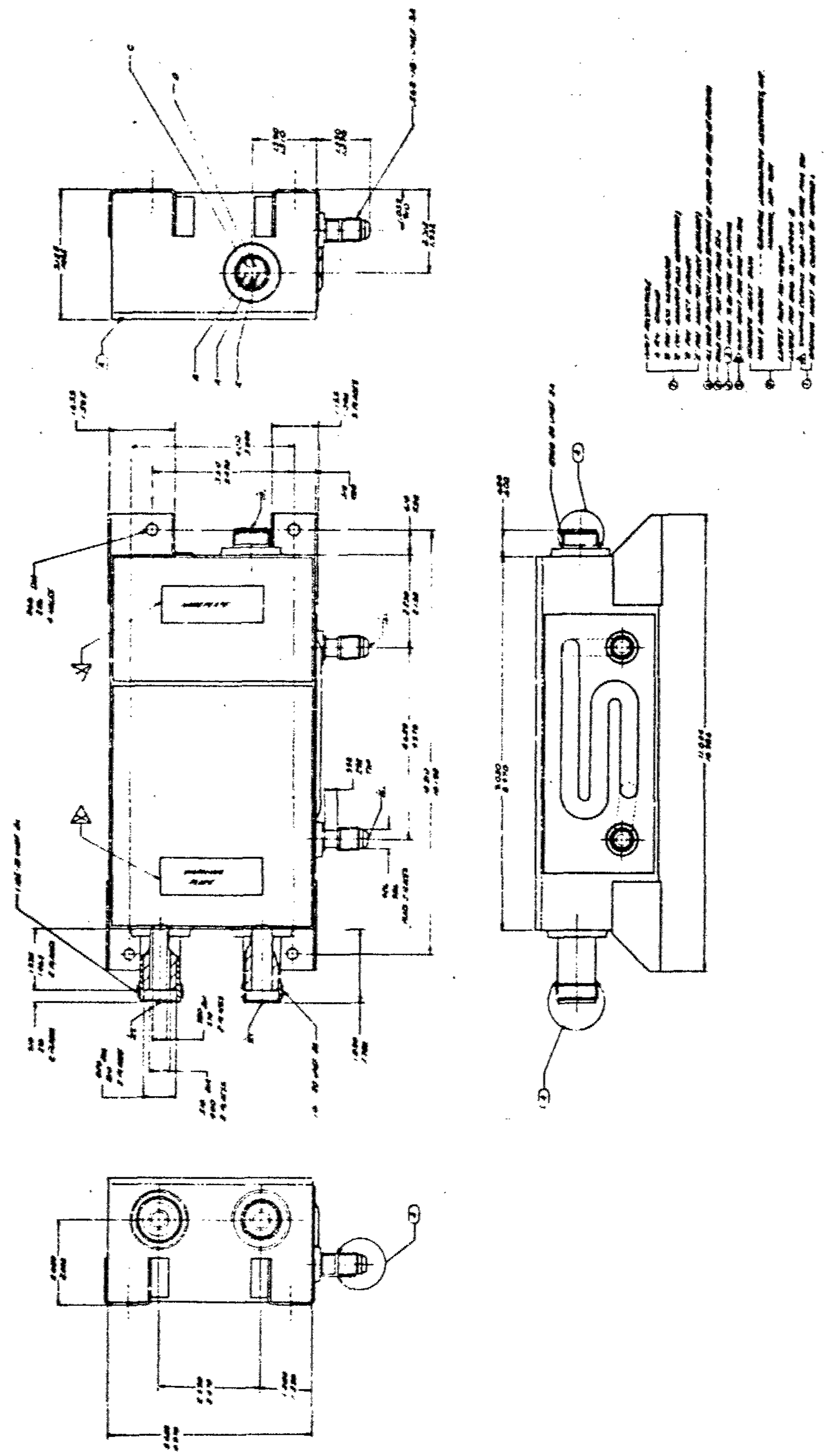

Figure 3. Ignition Exciter Installation

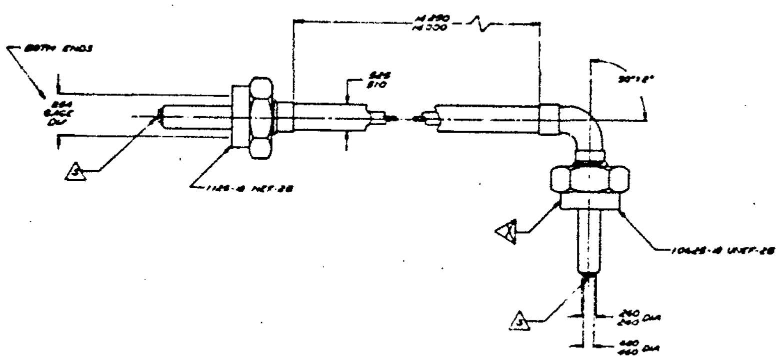

Figure 4. Gas Generator Burner Igniter Lead Installation

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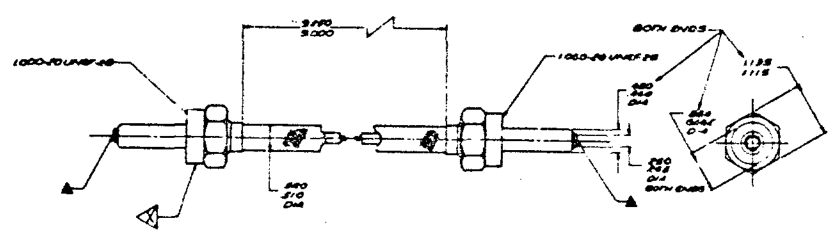

Figure 5. Duct Heater Igniter Lead Installation

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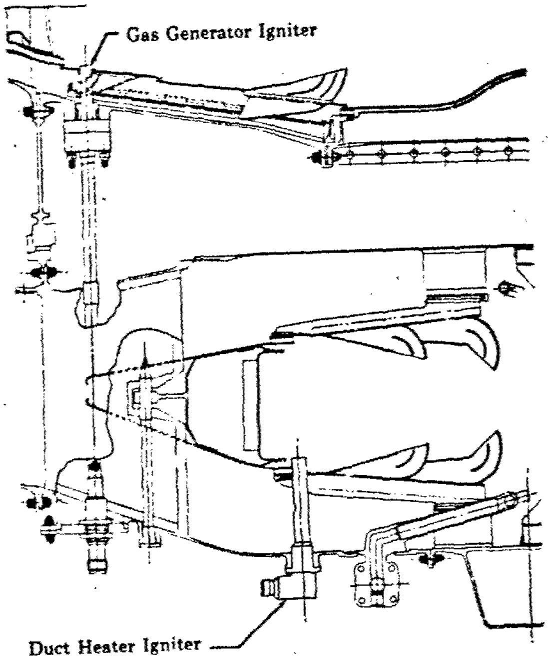

Figure 6. Gas Generator and Duct Heater Igniter Engine Installation

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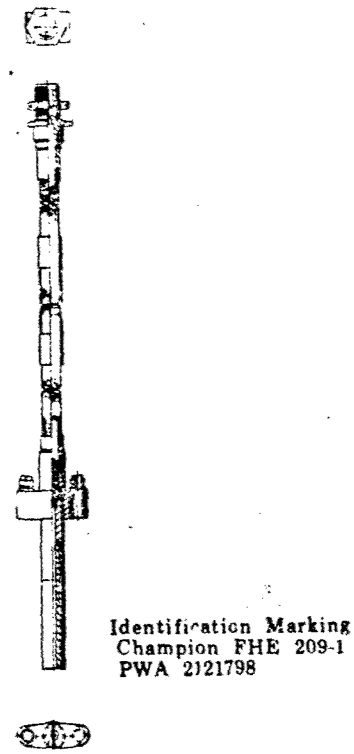

Figure 7. Gas Generator Igniter Assembly

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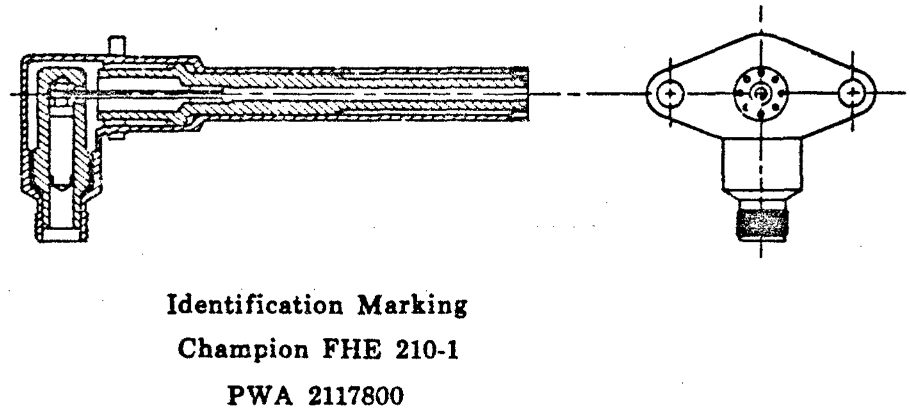

Figure 8. Duct Heater Igniter Assembly FD 17159 BIII

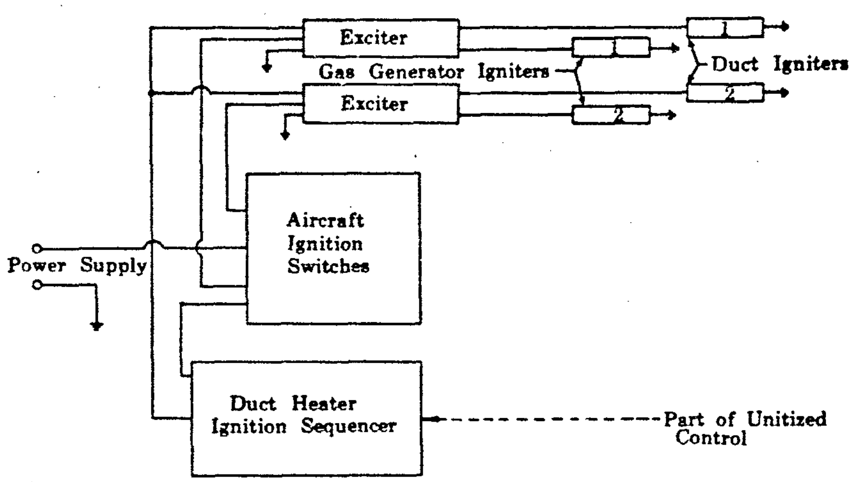

Figure 9. Ignition System Schematic FD 17157 BIII

2. Design Objectives

The design objectives of the ignition system are to define a configuration that will reliably produce ignition of the fuel in both the gas generator and the duct heater burner sections under all operating conditions of the engine throughout the flight envelope.

The system TBO goals have been established to coincide with those of the engine with the exception that the igniter may require more frequent replacement depending on service use. Particular emphasis has been given to (1) meeting ground and flight safety requirements, (2) maintainability goals of easy, rapid removal from the engine, and (3) minimum overhaul cost. The system is light weight, economical, and compatible with the associated engine components.

3. Design Requirements

Procurement specification J108 describes the design requirements of the ignition exciters and electrical cables, and procurement specification J117 is provided for the ignition igniters. Both specifications are available from FRDC on request. These requirements are based on the experience accumulated in commercial and military applications of spark ignition systems.

Detail design requirements are as follows:

1. The duty cycle of the gas generator system will be 10 minutes on, 20 minutes off. The fan duct igniter normally operates for 1.25 seconds, but for design and substantiating test purposes a duty cycle of 2 minutes on, 3 minutes off, 2 minutes on, and 23 minutes off has been established.
2. A maximum of 500 pph of fuel flow will be used for cooling the exciters. The maximum fuel temperature will be 310°F.
3. The ignition system weight will be less than 26 pounds.
4. The internally pressurized (one atmosphere), hermetically sealed exciter housing must operate satisfactorily at 80,000 feet altitude. The housing is fabricated of stainless steel with fusion welded joints and has hermetically sealed electrical connections.
5. The system will be designed to prevent the possibility of fuel leakage into the electrical system to an absolute minimum.
6. Each exciter circuit will include a spark signal generator for use as a ground and flight monitor.
7. The system will function satisfactorily when subjected to ambient pressure equal to sea level through 80,000 feet with steady state ambient temperatures from -65°F to 575°F, intermittent operation to 650°F, and engine induced vibratory fields applied to the system.
8. Radio interference will be less than the limits specified in MIL-E-5007B. The reduced limits of the broadband peak noise as shown in figure 10 apply.
9. The gas generator spark repetition rate will be a minimum of 1.0 spark per second at the minimum aircraft voltage supplied to the exciters. The duct heater rate will be 5 sparks per second at the minimum aircraft voltage supplied to the exciters.
10. The exciter output voltage will be 2800 to 3200 volts, with boost capability to 6000 volts. The energy level at the igniter gap will be at least 1 joule. Input voltage will be 230 volts for the Lockheed application and 200 volts for the Boeing application at 400 cps.

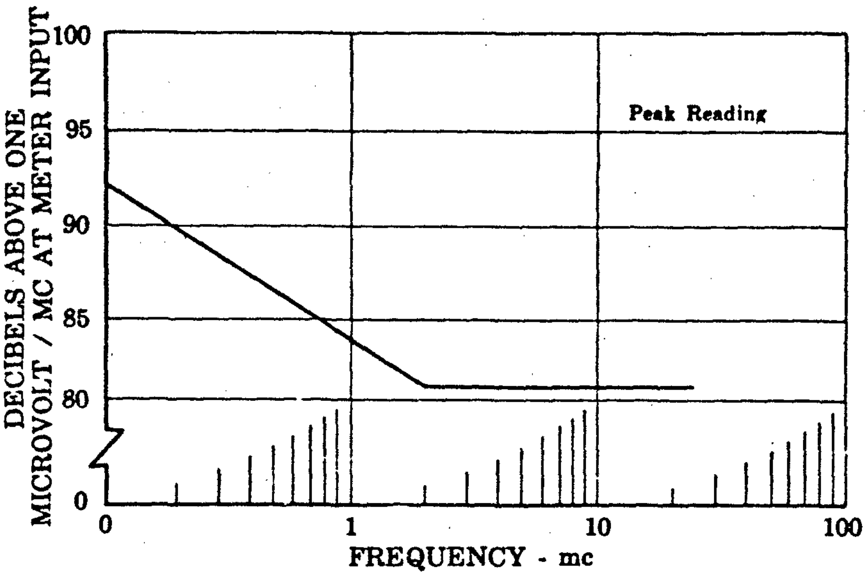

Figure 10. Radio Broadband Interference Limits FD 17158 BIII

4. Design Criteria

Design criteria for the ignition system details are listed below:

1. Housing: The case thickness is established by radio noise suppression requirements that provide a case wall stress safety factor of four.
2. Electrical input connector: The input connector has an electrical voltage safety factor of 4, i.e., the rated voltage is four times the maximum operating level.
3. Inductor: The inductor coil has a temperature safety factor of 1.1.
4. Capacitor: The filter capacitor has a voltage safety factor of 2.9; the charging capacitors have a voltage safety factor of 1.7; and the tank capacitor has a voltage safety factor of 1.1.
5. Transformer: The transformer has a temperature safety factor of 1.1.
6. Rectifiers: The gas generator rectifiers have an electrical power stress safety factor of 13, and the duct rectifiers have a safety factor of 2.7
7. Resistors: The resistors have an electrical power safety factor of 2.7.
8. Discharge tube: The discharge tube is life limited by gap erosion which is affected by the number of sparks and can be expressed by ignition "on" time. Based upon the normal time "on" per engine hour, average spark rates, and engine TBO, the discharge tube will have a life safety factor of 3.3.

Note: Safety factor as used in this section is based on the rated value of the particular item where satisfactory continuous operation has been demonstrated.

5. Materials Summary

The materials used in the exciter are as follows:

1. Housing: AISI 430 corrosion and heat resistant steel
2. Wire: Copper with teflon insulation
3. Inductor: MIL-W-583 type ML magnetic wire
4. Transformer: Standard type with mica and asbestos paper insulation
5. Capacitor: Aluminum foil with mica insulation
6. Discharge Tube: Glass tube enclosed tungsten tipped spark gap, which is filled with a dry inert gas
7. Connector: 85% alumina insulation in an AMS 5610 case.

6. Design Approach

The ignition system exciter and electrical cable vendor and the igniter vendor will be selected by the end of the Phase II-C program. These vendors will be directed, at the beginning of Phase III, to proceed in accordance with the requirements defined in the P&WA procurement specifications. P&WA will then utilize the vendor coordination procedure established through previous experience that includes engineering guidance, parallel development programs, and scheduled conferences to control the vendor activities toward successful compliance to the SST JTF17 engine system requirements.

The design approach of the two vendors is similar. Both designs utilize capacitance discharge circuits of the same basic design as currently used on P&WA commercial engines. Both designs provide fuel cooling through the use of a fuel cooling coil to provide separation of fuel from the electrical components. Both vendors have accomplished heat rejection simulation tests under maximum SST environmental conditions, which indicated that the internal electrical components of the fuel-cooled exciter units operate at least 10% below the rated temperature limits.

The 3KV, 4-joule ignition system was selected to achieve the lightest weight and the coolest operation compatible with the SST altitude and ambient temperatures. The shunted igniter's ability to fire is virtually unaffected by engine burner pressure or contamination of the electrodes by fuel or carbon type deposits. The low tension system was chosen because of the ability of the leads and connectors to transmit the exciter discharge voltage at the high altitude and temperature conditions of the SST without complex pressurization and sealing methods.

This system design utilizes previous experience in establishing design requirements for use of a nonradioactive exciter discharge tube, an alternating current as a power source to eliminate the vibrators required on direct current powered systems, and a fusion welded steel exciter box to improve sealing of the unit. The spark time duration of a low tension ignition system is shorter than that of a high tension system and does not require the use of the high frequency transformer coil in the discharge circuit. This results in a higher peak power in the first pulse of the spark, which increases the effectiveness of the spark in igniting the fuel-air mixture of the burner.

The ignition leads, which represent a resistance loss in the discharge circuits, are designed for the shortest possible length to provide maximum energy at the igniter gap.

The thermal design of the exciter takes advantage of the fact that the electrical components can be divided into two groups which are established by their ambient temperature limits: (1) those details that are capable of operation in the SST environment, and (2) those that require a lower ambient temperature. Those electrical components that require the lower ambient temperature will be located in the exciter box in direct contact with the fuel-cooled wall. The components that do not require any special thermal consideration can be located internally in the exciter away from the wall.

The system will perform satisfactorily when subjected to variations in voltage, vibration, humidity, sand and dust, acceleration, and impact as required by the engine specification.

Subcomponent design features are as follows:

1. Case: The exciter case is a stainless steel welded assembly including integral external mounting brackets, input and discharge electrical connectors, and subcomponent mounting partitions. The use of weld studs and welded partitions eliminates the need for fasteners through the case wall.
2. Filter: The radio noise filter is an inductance type that utilizes a reactor with a feedthrough capacitor.
3. Power transformer: The leakage reactance type power transformer used in both the gas generator and duct burner circuits will be molded in a module of resilient silicone material and anchored to the cooled wall of the exciter box.
4. Rectifiers: The rectifiers used will be the full wave, solid state type.
5. Capacitors: The capacitors used will be made of impregnated reconstituted mica and aluminum foil molded into a sealed stainless steel case. The capacitor will be designed for operation at a voltage in excess of the actual operating voltage to provide reliability.
6. Discharge tube: The discharge tube is the dry gas-filled type similar to the long life discharge tubes now used in ignition exciters on the JT8D engine.
7. Resistors: The current limiting resistors and the open circuit safety resistors used will be the wire wound vitreous enamel-coated type.

7. Anticipated Problems

Three areas of the system that require concentrated development effort are the igniters, electrical cable insulation and connectors, and hermetic sealing of the exciters.

Previous experience has indicated that the shunted gap igniters used in conjunction with low tension systems are susceptible to electrode

erosion plus erosion and oxidation of the semiconducting material. Improved semiconducting material with greater resistance to oxidation and erosion, electrode materials with increased erosion resistance, and improved tip cooling techniques will be developed to increase igniter life.

The sealing of the exciter to maintain an atmosphere of internal gas pressure to provide electrical insulation within the exciter is essential for SST high altitude ignition system operation. A welded stainless steel package will be used that has a minimum of external connections passing through the housing walls. Fabrication and checkout procedures will be developed to minimize the possibility of system malfunction due to loss of internal pressure.

8. Development Test Plan

A detail description of the development test plan and test schedule for these components is included in the Component Test and Certification Plan, Volume III, Report E, Section II.

J. FUEL MANIFOLD DRAIN VALVES

1. Description

Fuel manifold overboard drain valves are installed in the gas generator fuel manifold and in each of the duct heater fuel manifolds of the JTF17 engine. These valves open after fuel shutoff to drain residual fuel from the manifolds and nozzles and thereby prevent internal coking. The three valves, which are all of common P&WA design, are similar to a J58 design. They are automatically actuated by hydraulic signals from the unitized fuel control. The valve assembly consists of a sliding gate valve actuated by a hydraulic piston. A cross section drawing and an external photograph of the valve are shown in figures 1 and 2.

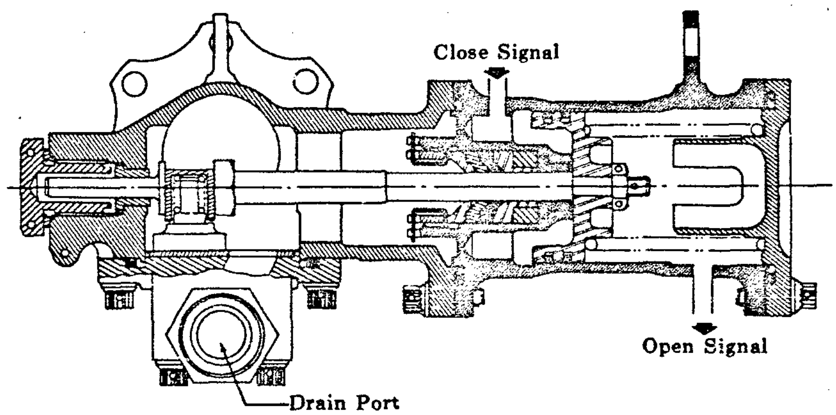

Figure 1. Fuel Manifold Drain Valve Cross Section FD 14829 BIII

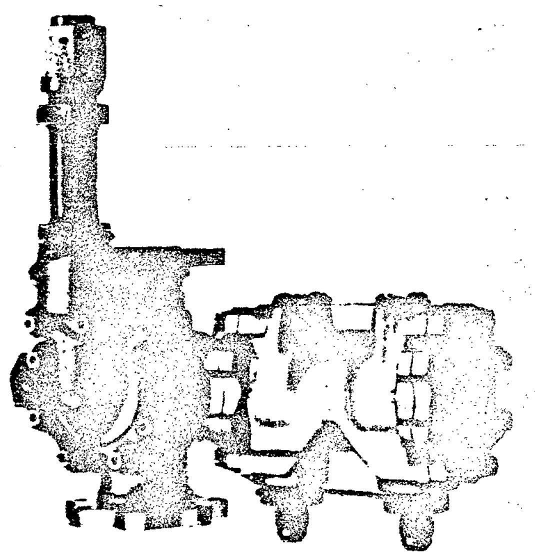

Figure 2. J58 Engine Fuel Manifold Drain Valve FD 16737 BIII

2. Design Objectives

The design objectives for the fuel manifold drain valves are:

1. To be replaceable on an installed engine within 30 minutes
2. To be quickly and easily overhaulable at existing facilities
3. To have a TBO equal to or greater than the engine
4. To have a mean-time-between-failure and chargeable premature removal rate consistent with current commercial applications
5. To minimize development and production costs by utilizing previous development experience, economical manufacturing techniques, and materials that are inexpensive yet suitable for the purpose
6. To be of the lightest weight possible consistent with the requirements of maintainability, reliability, durability, and safety
7. To provide growth potential through minor modifications.

3. Design Requirements

The detail design requirements for the valve include:

1. Gate leakage: Overboard fuel leakage will be less than 10 drops per minute at 0 to 700 psig fuel pressure with fuel temperature from a minimum temperature corresponding to 12 centistokes viscosity to 400°F.
2. Shaft seal leakage: The shaft seal leakage in the valve open position will be less than 10 drops per minute at the same pressure and temperature conditions as in paragraph 2 above.
3. Proof pressure: The valve must withstand an internal pressure of 1750 psig without permanent deformation.
4. Durability: The valve must function on the engine without significant degradation for the full period of engine TBO.
5. Ambient air temperature: The valve must operate satisfactorily within an ambient air temperature range from -65°F to 650°F.
6. Vibration and pressure fluctuations: The valve must function normally when exposed to engine induced vibratory forces equivalent to 10 G and fuel pressure fluctuations of 10% of operating pressure.
7. Fuel contamination: The valve must operate satisfactorily for 80 hours with fuel containing the following contaminant:

Particle Size, Microns	Percent of Total
0 - 5	39 ± 2 by weight
5 - 10	18 ± 3 by weight
10 - 20	16 ± 3 by weight
20 - 40	18 ± 3 by weight
Over 40	9 ± 3 by weight
Through a 200-mesh screen	100 by weight

8. Fuel compatibility: The valve is designed for operation with PWA 522 (Jet A, A1) and PWA 533 aviation kerosene.
9. Fuel inlet temperature: The valve is capable of operating within a fuel inlet temperature range between that temperature corresponding to 12 centistokes viscosity and a maximum of 400°F.

4. Design Criteria

1. Housings: The minimum housing operating stress safety factor is 4.
2. Seals: Metallic static seals are used throughout. The actuator shaft seal packing consists of Inconel X filled asbestos.
3. Bolts: All bolts have rolled threads. The number of bolts and torque limits at each location is established with respect to the anticipated cyclic and vibratory stresses and to provide ample margin for stress corrosion. The bolt stress safety factor is 2.
4. Piston, shaft, and bearings: The piston, shaft, and shaft bearings were selected and sized to utilize previous experience resulting from the development of high temperature drain valves and actuators for the J58 engine. The bearing stress safety factor is 3.
5. Gate and seat: The gate and seat are both composed of PWA 771 material with large contact areas to assure long life wear characteristics. The gate-to-seat bearing stress safety factor is 4.

5. Materials Summary

The following materials are used in the drain valve assembly:

Part	Material
Housing, cover, bolts, plug and piston	AMS 5616 Greek Ascoloy
Rod	AMS 5667 Inconel X
Gate and seat	PWA 771 Corrosion and Heat Resistant Alloy
Bushing	PWA 790C1 Cast Iron
Piston rings	AMS 7310 Cast Iron
Static seals	AMS 5542 Inconel X
Shaft seals	Inconel X Filled Asbestos

6. Design Approach

The basic drain valve configuration was qualified in the J58 program at more severe conditions than expected in the SST application. The only revision is a minor housing redesign to enlarge the flow passages.

7. Anticipated Problem Areas

The drain valves are of simple design, with prior development experience. P&WA anticipates that a minimal development effort will be required. The only foreseeable problem area is gate valve

leakage. An accelerated cyclic endurance test will be accomplished early in the de 'opment program to determine if a problem exists. If required, alternate valve gate and seat materials and configurations will be evaluated.

8. Test Programs

A detail description of the development test plan and test schedule is included in the Component Test and Certification Plan of this proposal, Volume III, Report E, Section II.

K. MISCELLANEOUS COMPONENTS INCLUDING POSITION SWITCHES, EXHAUST NOZZLE POSITION INDICATOR, ELECTRICAL HARNESSES, EXHAUST GAS TEMPERATURE MEASUREMENT SYSTEM, ACTUATORS, AND FUEL OIL COOLERS

The compressor bleed position switches, the reverser-suppressor position switches, the compressor inlet guide vane and the aerodynamic brake position switches, the Boeing secondary air control position switches, the exhaust gas temperature measurement system, the exhaust nozzle position indicator, and the electrical harnesses are described in Volume III, Report B, Section II, paragraph J.

The fuel oil-coolers and the oil bypass valves are described in Volume III, Report B, Section IV.

The reverser-suppressor actuators, the duct heater exhaust nozzle actuators, the compressor bleed actuators, the compressor inlet guide vane and aerodynamic brake actuators, and the Boeing secondary air control actuators are described in Volume III, Report B, Section II, paragraphs F, E, and A respectively.

L. PHASE II-C BENCH AND ENGINE COMPONENT TEST RESULTS

1. Description of Initial Experimental Engine Control Type System Components

Operation of the initial experimental JTF17 engines is being accomplished using existing flight-type control system components to expedite the test program. These components sense the same engine parameters and function in the same basic mode as the final controls that will be used on prototype and production JTF17 engines.

The components used in this initial control system are basic J58 and TF30 units modified as necessary to provide the same functions and mode of control that will be utilized on the prototype and production JTF17 engines. J58 pumps of the same types that will be used on the final JTF17 engines are also being utilized. These are flight-type hardware and are engine-mounted.

In addition to the automatic control system, control components that provide separate manual control of the individual controlling functions are being used. These components provide the flexibility necessary for the operation of the initial experimental engines and include gas generator fuel flow, duct burner fuel flow, duct exhaust nozzle position, compressor bleed position, and high compressor vane position.

The decision to use modified J58 and TF30 controls for the Phase II testing was made to permit operating the engines with flight-type controls sensing the correct parameters and utilizing the desired control mode early in the program. Because of the long lead times involved in analyzing, developing, manufacturing, and bench testing components such as these, it was impractical and uneconomical to attempt to provide prototype design hardware for the initial engine tests. Engine testing with the modified J58 components is providing data confirming that the selected control modes are satisfactory and is supplying information to be used in establishing firm schedules and parameter levels. This approach reduces the cost and cycle time required to obtain satisfactory prototype and production components, because the initial prototype design will be based on engine tested concepts and operating conditions rather than analytical information.

The control system functions are being obtained with the following components:

1. Gas generator fuel supply: Chandler Evans main fuel pump from the J58 engine
2. Duct heater fuel supply: Hamilton Standard turbopump and controller from the J58 engine with the inducer on the pump impeller removed to increase capacity
3. Gas generator fuel scheduling: Bendix CJ-Q1 fuel control. This is a J58 control modified to incorporate new cams to provide required JTF17 schedules, to provide a hydraulic signal to schedule for duct pressure ratio, and to provide signal-to-control inlet guide vane positions as a function of compressor speed and inlet temperature
4. Duct heater fuel scheduling: Bendix AA-M1 fuel and nozzle control or Hamilton Standard JFC-51 control. The Bendix AA-M1 is a TF30 control modified to schedule fuel flow

as a function of power lever angle, compressor inlet temperature and gas generator burner pressure, and airflow as a function of duct pressure ratio. The JFC-51 is a J58 control, unmodified

5. Duct heater exhaust nozzle positioning: Automatic control obtained with the modified Bendix AA-M1 control, and manual control obtained with the unmodified Hamilton Standard exhaust nozzle control system from the J58 engine
6. Gas generator manifold dump valve: Hamilton Standard windmill bypass and dump valve from the J58 engine, unmodified
7. Duct heater manifold dump valves: P&WA manifold dump valves from the J58 engine, unmodified
8. Compressor bleed control: Automatic operation obtained with the Bendix CJ-Q1 fuel control and manual control obtained with test stand mounted solenoid valves
9. High compressor vane position: Automatic control obtained with the Bendix CJ-Q1 fuel control, and manual control obtained with the Hamilton Standard exhaust nozzle control system from the J58 engine
10. Hydraulic pressure: Vickers hydraulic pump from the J58 engine, unmodified
11. Ignition system: GLA exciter and leads from the JT12 engine with igniters designed for the JTF17 engine.

The primary objective of the test program for the components used on the initial experimental engines is to provide a means of controlling the engine and obtaining basic control design, schedule, and parametric data during initial engine development testing. Bench tests of these components are being conducted in support of this objective, that is, to provide control system components as required to support the engine test program.

2. Phase II-C Test Results as of 1 August 1966

a. Initial Experimental Engine Controls

The initial experimental engine control system described above has provided completely satisfactory engine operation to date in Phase II-C. The complete flexibility of manual modes of engine control available by use of the existing engine control components greatly expedited the experimental engine test program. The first experimental engine was successfully started on the third attempt during the first day of testing by manual adjustment of the gas generator starting fuel flow schedule. All engine starts since have been successful using the same starting schedule. The ignition system has never failed to light either the gas generator or the duct heater to date. This control system has successfully provided the desired engine control during 73 hours of experimental engine operation.

All control components are presently available and ready for installation on the experimental engines during Phase II-C to provide a fully integrated control system utilizing the proposed JTF17 control mode and control parameters.

(1) Gas Generator Controls

The Bendix main fuel controls were equipped with a variable start schedule for initial engine tests. Different starting and acceleration schedules were selected by providing false compressor inlet temperature signals to the control. During the early runs of the first experimental engine, the desired starting schedule was determined and approximate steady-state fuel flow requirements established.

All of the control system components performed as desired. Compressor stator vanes and the duct exhaust nozzle were controlled manually through the use of the J58 exhaust nozzle control system.

Compressor stator data obtained during compressor rig testing and the initial engine runs were factored into the control schedules for the next series of engine tests during which the vane system was satisfactorily operated automatically.

A total of 283 bench hours were accumulated in accomplishing the aforementioned tests and in support of the initial experimental engines. Engine hours totaled 73 hours.

(2) Duct Fuel Controls

Three J58 JFL-51 fuel controls were calibrated to function as single-zone duct burner fuel controls with 259 hours of bench time devoted to this task.

A TF30 fuel and nozzle control was modified to a two-zone duct burner fuel control and exhaust nozzle control. Bench testing of this unit has accumulated 252 hours.

All Phase II-C experimental engine operation has been with J58 afterburner controls and manual splitting between zones. This testing has been successful and has demonstrated that this concept of controlling duct fuel flow and nozzle area is satisfactory.

(3) Gas Generator Pump

Tests were run on a CECO J58 main fuel pump to investigate the feasibility of using boost stage discharge flow to supply the manifold rapid fill system. It was found that as much as 30,000 pph was available without seriously depressing boost pressure to the gear stage. The above tests and support of the initial experimental engines has utilized 201 hours of bench testing. All Phase II-C engine testing to date has been accomplished with this pump without difficulty.

(4) Duct Heater Fuel Pump

Bench tests were made to investigate the use of a J58 turbopump on the Phase II-C engines. It was found that pump capacity could be increased 30% to meet the JTF17 requirements by removal of the inducer

and maintaining a higher inlet pressure. Three pumps were thus modified and used during the Phase II-C engine testing. A total of 517 hours was accumulated on the duct heater fuel pumps in support of initial experimental engines and bench tests. Engine operation has been satisfactory, and no problems with the pump have been encountered.

(5) Hydraulic Pump

A Vickers hydraulic pump presently used on the J58 engine was bench tested to determine pump performance with pump discharge pressure regulator reset from 2550 to 1500 psig, which the JTF17 will require, and the performance was found to be satisfactory. Two J58 pumps have been used to support the initial experimental engines. A third pump is available for use as a spare or on the third engine. Hydraulic pump test and support of initial experimental engines accumulated 10 hours of bench time without problems.

(6) Ignition System

Ignition testing at the FRDC in this period consisted primarily of providing an ignition system for the various JTF17 annular combustor development rigs for the JTF17 initial experimental engine program. Those rigs were the 7-inch duct heater segments, the 120° gas generator combustor segment, the full-scale duct heater rig, and the annular combustor JT4 rig for the JTF17 and associated programs. Ignition exciters used for testing were GLA 4-joule low tension (3000V) components using 28 vdc input power. Spark igniters were procured from both Champion Spark Plug and Bendix Scintilla and are of the shunted surface gap type similar to these proposed for the JTF17 prototype. The sparking tip and its location in the combustor are identical to that proposed for the JTF17. These tests indicated that a 4-joule low tension ignition system is entirely adequate for operation within the SST flight envelope. Figure 1 shows several points within the flight envelope where successful ignition was achieved in tests of the full-scale duct heater rig. There have been no instances in which ignition could not be obtained when the other related parameters were within the SST flight envelope.

During this period, GLA has been performing development testing of ignition high tension leads and of exciter components to increase the life and performance of these components at extreme environmental conditions. Bendix Scintilla has been doing similar work on electrical cable connectors.

(7) Miscellaneous Items

(a) Duct Burner Fuel Manifold Quick-Fill System

A system utilizing J58 hardware was assembled to supply gas generator fuel pump interstage fuel to the duct burner manifold to fill the manifold within the specified time during normal low-flow lightoffs. Extensive bench testing was completed to develop the valving and timing necessary to satisfy the needs of a quick-fill system. Engine tests were run on the JTF17 quick-fill system on two J58 engines where it was shown that smooth

augmentor lights at 3000 pph metered flow could be consistently made in 1.2 seconds. This time compares to 7 seconds without the quick-fill system. Breadboard hardware is now ready for initial experimental engine use.

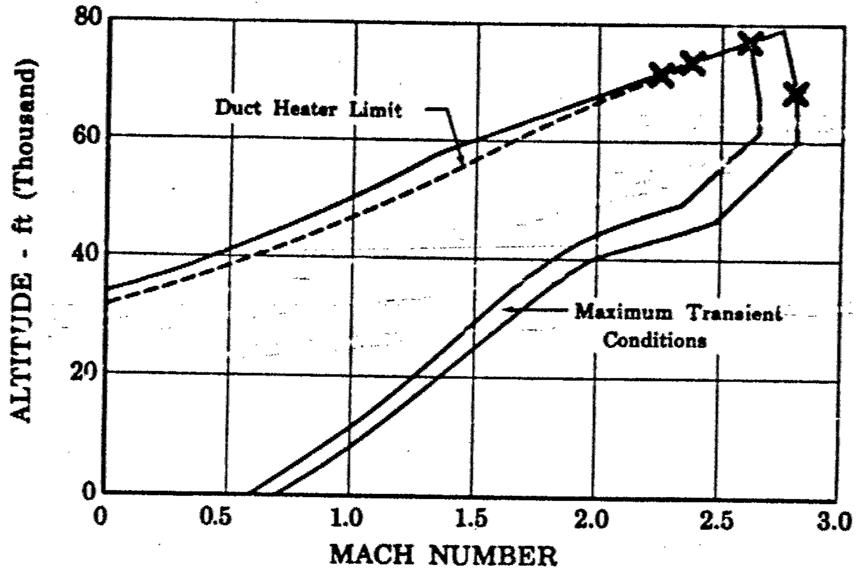

Figure 1. Demonstrated Lights of the Full-Scale Duct Heater Test Rig with 4-Joule Ignition System FD 17160 BIII

(b) Automatic Restart Switch

Five automatic restart switches were obtained, and one was bench tested for possible use on the initial experimental engines. A total of 4 hours of bench testing was accumulated on this unit.

b. Unitized Fuel and Area Control (Critical Component Testing)

Testing accomplished during Phase II-C on certain critical components proposed for the unitized fuel and area control are summarized below.

(1) Hamilton Standard Division Tests

(a) Fluorasilicone Seal Development

Tests of an elastomeric seal compound developed by Hamilton Standard Materials Engineering for use in jet fuel applications at fuel and ambient temperatures from -65 to +450°F and pressures up to 1500 psig were conducted. Of 48 seals evaluated at Hamilton Standard, 24 seals were fuel-to-fuel and the other 24 were fuel-to-air. All seals were static applications. A total of 1000 hours of testing was completed. Test results indicated the seals to be acceptable for the supersonic transport application.

(b) Quick-Disconnect Manifold Seal Temperature Test

Tests were conducted to evaluate a quick-disconnect concept sealing configuration to substantiate previously developed elastomeric seals under the supersonic transport application temperature and pressure conditions and to provide initial data on minimum flow requirements to ensure adequate cooling of the seals. Three seals were tested in each of two quick-disconnect fixture tests. One fixture was constructed of steel, the other of aluminum. Seal and skin temperatures were recorded during the mission cycle testing, which varied ambient temperature from -65°F to 600°F, fuel temperature from -58°F to 425°F, and pressure from 50 to 1000 psig. A total of 300 hours of test time was accumulated, and leakage remained zero throughout the test.

(c) Compatibility Tests of Aluminum Alloy RR-350 and Commercial Kerosene

Six test panels of the proposed JTF17 control housing material (RR-350) were tested with jet fuel at a fuel temperature of 350°F. No problems were noted during the 500-hour test which confirmed the compatibility of the alloy with commercial kerosene.

(2) Bendix Product Aerospace Division Tests

(a) Housing Material Evaluation

Bendix has conducted heat absorption tests using an aluminum alloy, a variable cross section test fixture, and simulated supersonic transport fuel and ambient temperatures to substantiate the choice of housing material. Bendix concluded the aluminum alloy will withstand the environment to be encountered in the supersonic transport.

(b) High Temperature Seal Tests

Bendix has conducted evaluation programs on a fluoracarbon material for use as a high temperature elastomeric seal. The tests indicated that the material would withstand supersonic transport environment satisfactorily. A number of these seals are being used in the initial experimental JTF17 gas generator control. Seal construction consists of the fluoracarbon compound formed in a cup shape and a spiral spring inside the cup to ensure consistent shape throughout the environments of pressure and temperature.

(3) FRDC Tests

(a) Duct Heater Pressure Ratio Computer Test Results

Bendix Products Aerospace Division and Hamilton Standard both fabricated duct airflow computers to check the design concepts being used in the unitized fuel and area control. The units were initially calibrated at the vendor's plant then forwarded to the FRDC for further testing. Both of the vendor units calibrated to within the initial accuracy requirements of ± 2 percent. See figures 2 and 3.

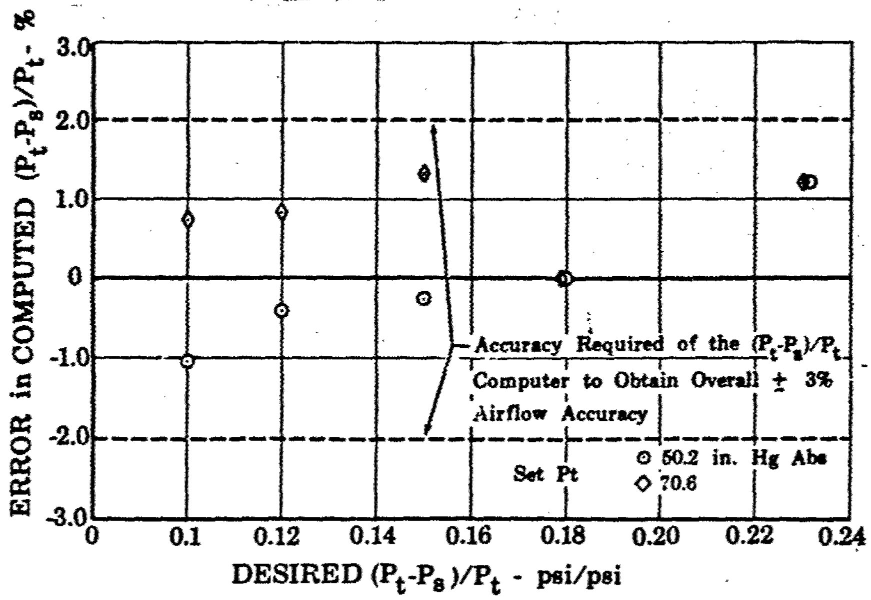

Figure 2. Test Results from Hamilton Standard Breadboard Duct Airflow Computer FD 17163 BIII

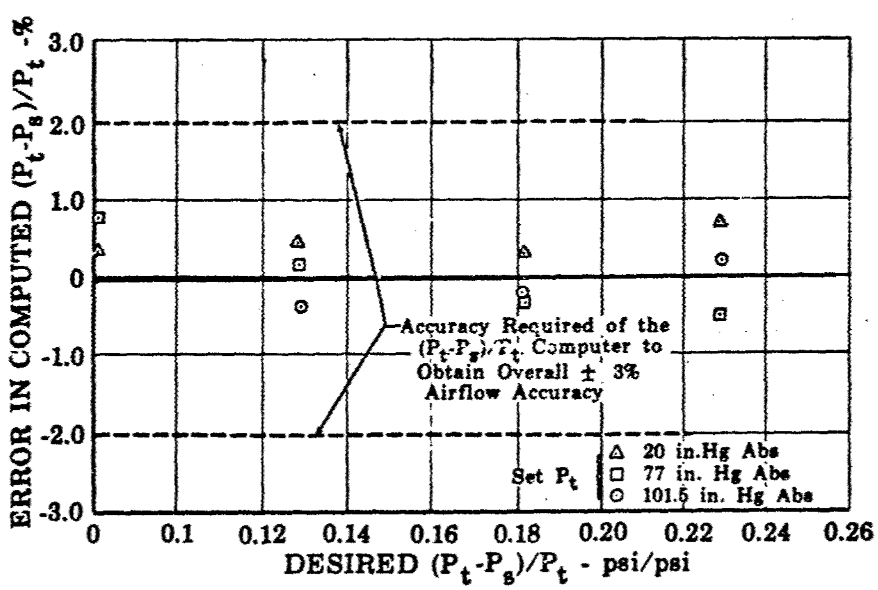

Figure 3. Test Results from Bendix Products Breadboard Duct Airflow Computer FD 17162 BIII

The concept of the Bendix unit was incorporated into the modified TF30 fuel and area control that is being used on the initial experimental JTF17 engines.

The Hamilton Standard unit was incorporated into part of a modified J58 main control that, when coupled with a J58 exhaust nozzle control, will be used to control the JTF17 duct nozzle.

The total run time on the HSD pressure ratio unit at the FRDC was 22.5 hours on a test bench. Approximately 12 hours of bench time were accumulated on the Bendix pressure unit.

(4) Duct Airflow Sensing Probes

It is necessary that total and static duct air pressure be sensed accurately and with fast response since the signals are used to control the duct-corrected airflow. A number of probe configurations are being studied to fulfill this requirement. The following is a summary of the probe configuration with the alternatives planned for engine use.

(a) Total Pressure Pickups

A Kiel probe is shown in paragraph D, figure 10. The Kiel probe is a scaled-up version of "off-the-shelf" probes that are used extensively for airflow measurement. Standard off-the-shelf Kiel probes were tested with respect to response and found to be unacceptably slow and to have excessive attenuation when coupled with a simulation of the engine plumbing and computer pressure cavity. For this reason the scaledup versions were made. Two of these large Kiel probes "teed" together appear to be the best total pressure sensing configuration.

As an alternative to the scaled-up Kiel probe, a rake probe has been investigated for use in sensing the total pressure in the duct. The original configuration of the probe was a simple airfoil shape with holes in the leading edge. This configuration was extremely sensitive to air angle of attack. The probe was then modified by adding a tube extension with a Kiel-type shield around the extension, which resulted in a probe that is insensitive to air angle of attack over an acceptable range. These results are shown in figure 4.

(b) Static Pressure Pickups

Several wall static tap configurations were tested for relative response and noise mission characteristics. It was found, in the frequency range which will effect the duct airflow computer, that two holes 0.100 inch in diameter will produce the desired response and have acceptable noise characteristics.

(c) Downstream Pressure Probe

The probes described above are planned for use near the exit of the fan. If the air velocity at this station approaches Mach 0.9, local shock waves in the vicinity of the probes may produce unreliable pressure

signals and may require that the total static pressure pickups be moved downstream in the duct to a location where the duct velocity is lower. A probe was developed that has the desired air angle acceptance on both the total and static pickups. A photograph of the probe is shown in figure 5, and the air angle acceptance data are shown in figure 6.

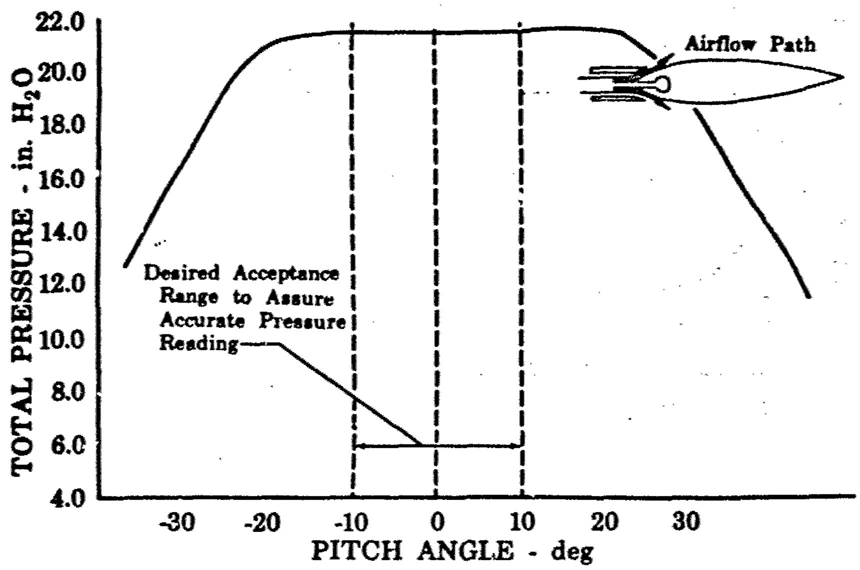

Figure 4. Duct Diffuser Total Pressure Probe with Kiel Extension FD 17161 BIII

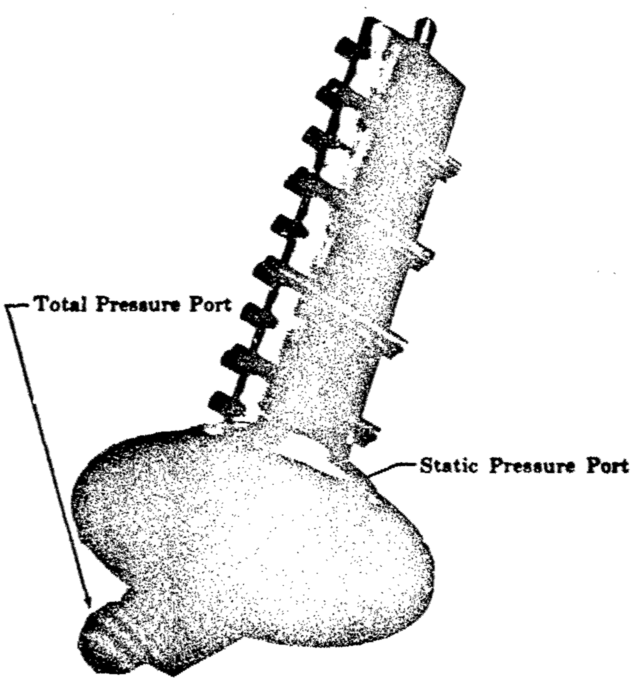

Figure 5. Duct Diffuser Total and Static Pressure Probe FD 16712 BIII

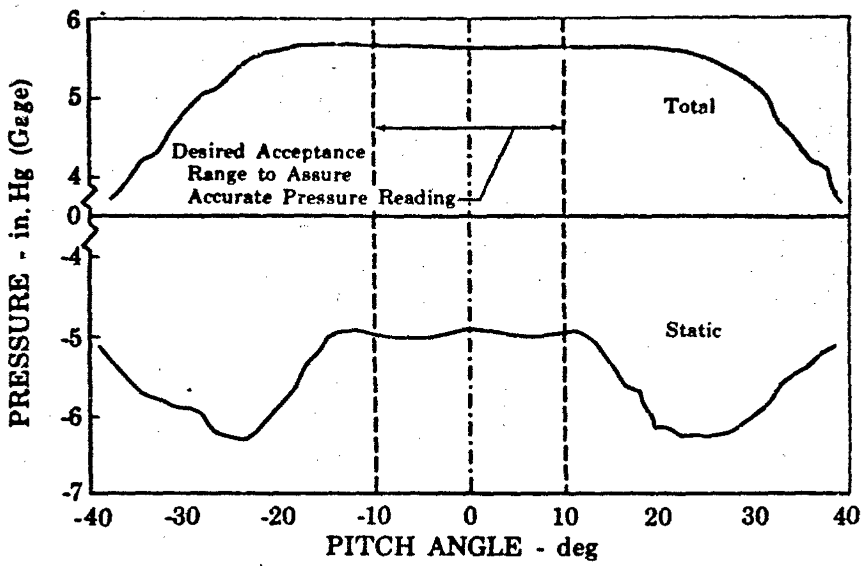

Figure 6. Variations in Total and Static Pressure as a Function of Pitch Angle on Duct Diffuser Total and Static Pressure Probes

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M. COMPONENT TEST FACILITIES

The facilities available for testing JTF17 fuel and control system components are described in detail in the Facilities Plan in Volume V, Report B of this proposal.

There are 44 separate test benches available at the FRDC for the development testing of turbojet engine control components, oil and hydraulic systems, gear boxes, and ignition systems. Of these 44 stands, 15 test benches include the capability of simulating high Mach number fuel, oil, and/or ambient temperature conditions. Figures 1 and 2 show a typical control bench. Twenty-five of the test benches will be assigned directly to the JTF17 engine program. Seven of these benches will require minor modification to be completely acceptable for the JTF17 requirements. The modifications required include increasing the air supply capacity to accommodate the duct heater turbopump, installing actuator loading rigs so that the duct nozzle actuator-control system can be properly tested with respect to dynamic response, increasing the drive horsepower, and in general increasing the flow capacity of the plumbing, flowmeters, counters, etc. Additional equipment will be added to the facility to be used to environmentally test the ignition system (exciters, harnesses, igniters, and automatic restart switches) at simulated SST ambient temperature and pressure conditions. This bench will include a fuel pressure source for cooling the ignition units.

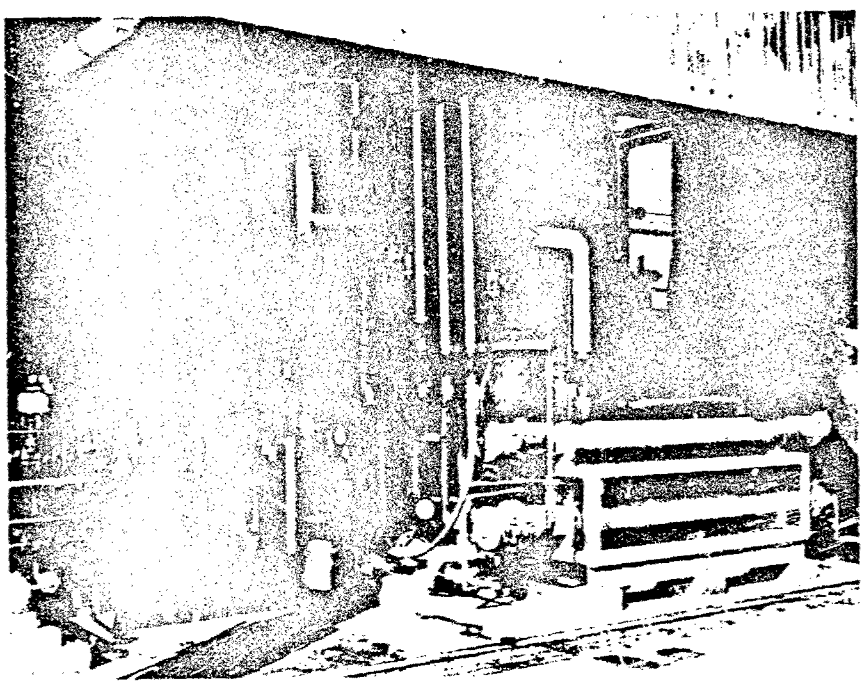

Figure 1. Outside View of D-12 Fuel System Component Test Bench Used for Testing Controls at Elevated Fuel Temperatures

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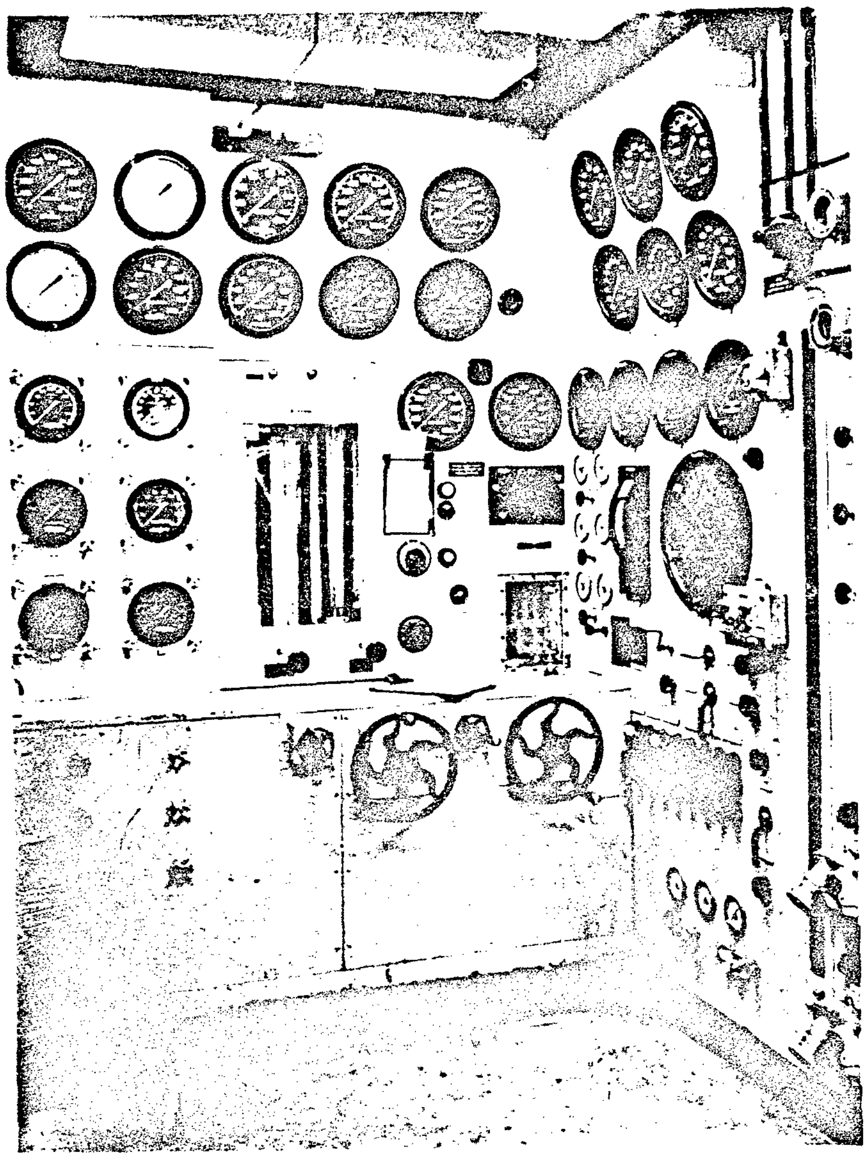


Figure 2. Operator's Control Panel of D-12 Fuel System Component Test Stand (Typical of Stands With Elevated Fuel Temperature Capabilities) FE 11633 BIII

Two electronic control benches will be required to support the program and will be suitable for calibrating, reprogramming, and maintaining the digital electronic engine pressure ratio (EPR) control and the digital electronic airflow computer by simulating inputs, measuring outputs, simulating output loads, and accomplishing troubleshooting. The benches will include a circulating cooler system to simulate fuel cooling and 400-volt, 400-cps, 3-phase and 24-volt power supplies.

J58 experience has emphasized the importance of testing the fuel system components in an environment that simulates as closely as possible the actual flight conditions. The FRDC turbojet component facilities are thus oriented toward high temperature fuel and ambient testing. Similarly, the two companies that are being considered as vendors for the unitized fuel and area control both have extensive facilities required to develop and produce fuel controls for use in high temperature ambient conditions.

Hamilton Standard has 17 low temperature control development test stands and 14 high temperature stands. The low temperature stands have a capability of fuel temperature from -65°F to 250°F, and two stands can be covered by an environmental chamber to give ambient capabilities of -65°F to 350°F. The high temperature stands have a capability of fuel temperature from -65°F to 600°F and ambient -65°F to 1000°F. These stands have a normal fuel flow capacity of 0 to 100,000 pph and 0 to 30 gpm hydraulic systems.

Bendix Products Aerospace Division has facilities that are presently available for use in the development of the unitized fuel and area control and that consist of 17 benches as follows:

1. Five with 60,000 pph flow capacity
2. One with 120,000 pph flow capacity
3. Four with 30,000 pph flow capacity
4. Two with 20,000 pph flow capacity
5. Four subassembly test benches with 7500 pph flow capacity
6. One low bench for testing duct air flow computers.

Nine benches are for room temperature calibration, and eight of the benches are capable of simulating the maximum SST fuel and ambient temperatures.

Bendix plans to build two new benches specifically for the unitized fuel and area control if awarded the contract. The new benches will have a flow capacity of 150,000 pph, control drive provisions, and full fuel and ambient temperature capabilities.

N. CONTROL SYSTEM FAILURE MODES AND EFFECTS ANALYSIS

1. Phase II-C

a. Failure Mode and Effects Analysis Conclusions

A preliminary failure mode and effects study on the proposed control system is being accomplished during Phase II-C. Preliminary comments and conclusions are presented in the following paragraphs.

The JTF17 fuel control system is designed to provide a failsafe system. A study was made to determine the results of any one control system detail part failure, the probability of which is not negligible, and which may be caused by fuel contamination, detail part distortion or fracture, bellows or diaphragm rupture, or sticking of sliding parts. This study was conducted to demonstrate the degree of compliance of the entire fuel system with the following requirements:

1. Any fault during takeoff shall not reduce thrust to below 90% of static augmented rated thrust at 100°F engine inlet air temperature over the altitude range of sea level to 6000 feet altitude.
2. Upon occurrence of any fault during takeoff, cruise, or landing approach, it shall be possible to prevent engine operation beyond established limits. Preventive action shall be possible in such a period of time that damage to the engine will not occur.
3. A fault of any single functional part shall not prevent air starting of the engine or regulation of thrust between 5% above idle and 90% of augmented thrust.

Extensive commercial and military experience with control system components of the type used in the JTF17 engine has shown that basic parts such as housings, castings, levers, rollers, and springs are designed with sufficient margin to prevent their breakage, therefore it was assumed, in making this study, that failures of these parts will not occur. Similarly, it was assumed based on this same experience that the sliding seals used in this design will not incur any significant wear within the specified overhaul period.

The possibility that fracture of the duct heater fuel pump air turbine could occur is negligible because:

1. The low cycle fatigue design criterion is 100,000 cycles.
2. The maximum normal turbine rotor speed is 27,500 rpm. The vortex venturi at the turbine exhaust limits turbine overspeed to 41,000 rpm at no load conditions. The turbine burst speed is 85,000 rpm.
3. The turbine and blades are machined from a PWA 1005A Waspaloy forging. There are no through-holes in the turbine disk.
4. A quality assurance spin proof test to 71,000 rpm is performed on each finish machined turbine disk prior to assembly of the pump.

In view of the above, fracture of the duct heater fuel pump turbine was not considered in the Failure Mode and Effect Analysis (FMEA).

The Phase II-C FMEA revealed that there were some areas which did not conform to the fail-safe requirements established for this system. These areas have been reviewed and design changes have been made to minimize the probability of malfunction and to reduce the seriousness of the consequences of the malfunction.

In making this study, it was recognized that it is possible for a single failure to occur that would not be detected during normal operation of the engine and that, in itself, would not cause a significant fault to result, but could cause a serious problem if the part were continued in service and a second failure occured at some later time. Where the failure analysis indicated that such a problem might occur, design modifications were made to reduce the probability of two successive failures occurring within an overhaul period. In addition, where practical, routine ground service checks, designed to detect such discrepancies or failures as soon as possible after they occur, are being incorporated into the applicable service documents. As an example, the JTF17 engine incorporates two compressor inlet temperature probes to sense inlet temperature. The control system will operate satisfactorily with either one functioning, but will malfunction if both fail. These probes incorporate failure detecting flags that project when a failure occurs. JTF17 engine service documents will recommend ground inspection at regular intervals so that a failure of one system will be detected and corrected before a second failure can occur.

Page limitations prevent including the complete system analysis conducted during Phase II-C herein, but a sample portion is described below to illustrate the type of analysis conducted.

b. Conclusion

The preliminary fault analysis for the proposed JTF17 engine fuel system, as well as previous experience with similar type components, establishes that this system, as modified by the result of this analysis, possesses a level of reliability equivalent to the high level already established for similar components on current commercial P&WA engines such as the JT3 and JT4 series.

c. Sample Phase II-C FMEA

The following paragraph describes the Phase II-C FMEA for a typical portion of the JTF17 system (the duct heater fuel pump). This is presented to show the format used for the preliminary FMEA performed in Phase II-C on the major components of the control system proposed for the JTF17 engine. Similar studies are being made for all other significant components, and the complete Phase II-C preliminary FMEA is available for review on request.

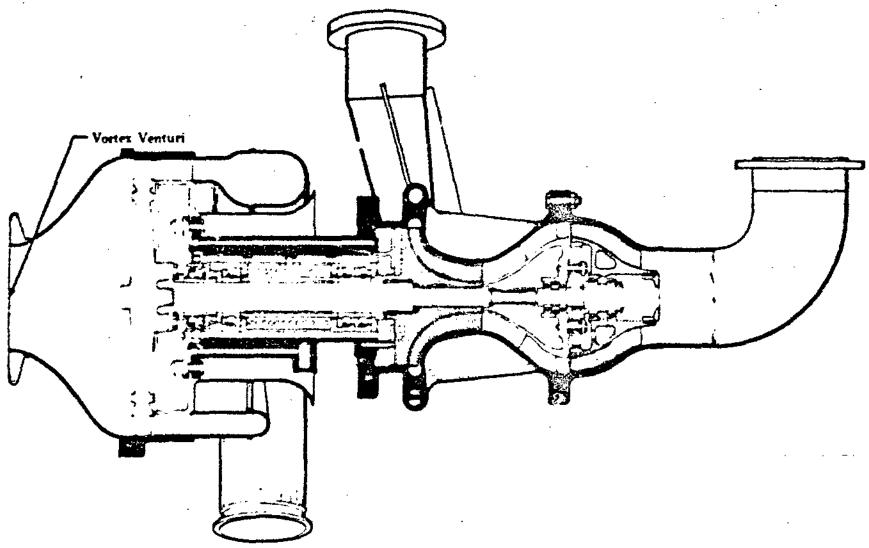


Figure 1. Duct Heater Fuel Pump Cross Section FD 13508A BIII

d. Duct Heater Fuel Pump (See figure 1.)

1. The duct heater fuel pump used on the JTF17 engine is described in paragraph G.
2. Analysis
   The FMEA for the duct heater fuel pump is shown on table 1.

2. Phase III Failure Mode and Effects Analysis

Failure mode and effects studies and the resultant corrective action initiated by the results of the Phase II study will be continued as required through the Phase III program. It is planned that analyses of the fuel, hydraulic, and ignition systems will be updated at approximately 6-month intervals. Computer facilities will be used extensively to support this work as described in Volume IV, Report F, Section II. The extensive mission cycle bench endurance tests which are planned during the Phase III program will be used to supplement this effort, as experience shows that reliability problems associated with high time TBO will be uncovered by these tests long before the airline operators reach this level of operating time on service units. The results of these tests and studies will be periodically reviewed by the reliability group and the cognizant project engineers, and corrective action in the form of design, material, or procedure changes will be made and evaluated as required.

Pratt & Whitney Aircraft will also require the vendors supplying fuel, hydraulic, and ignition system components to provide FMEA in accordance with paragraphs 6.3.4, 6.3.5, and 6.3.6 of the typical component procurement specification summarized in paragraph Q, under the direction of and coordinated with P&WA engineering. The data thus received from the various vendors will be reviewed by P&WA and integrated into the complete systems analysis described above, and corrective action taken as required.

Table 1.

JTF17 FAILURE MODE & EFFECT ANALYSIS

Sheet I

No. 1 of 2

25.6 Duct Heater Fuel Pump

Item	Function	Failure Mode	Failure Effect on Subsystem	Method of Detection	Failure Effect on Engine	Failure Effect on Aircraft	Crew Action Required
Bearings (3) 25.6.1.	Support Main Pumping Shaft	Seizure	SLTO: Pump may seize due to bearing failure resulting in loss of duct heater fuel flow. Pump controller will schedule pump turbine air supply control valve to full open position.	If pump seizes, loss of duct heater fuel flow	If pump seizes, duct heater will flame out and augmentation will be lost due to loss of fuel flow. Engine bleed air to pump turbine will be increased due to full open position of air control valve $F_n = 60\% F_{nma}$	If pump seizes, AP and CR	If pump seizes, reduce to and/or maintain non-augmented PLA range. Adjust F_n level on unaffected engines to obtain desired aircraft conditions.
			Cruise: Same as SLTO	Same as SLTO	Same as SLTO except $F_n = 20\% F_{nma}$ In addition oil temperature will increase and may eventually exceed limits due to loss of duct heater oil cooler fuel flow	Same as SLTO Also may have IFS	If pump seizes, reduce to and maintain non-augmented PLA range. Adjust F_n level on unaffected engines to obtain desired aircraft conditions. Monitor engine oil temperature. May be necessary for IFS and reduction in aircraft Mn to prevent exceeding oil temperature limit.
			Landing: Not affected. If pump seizes, duct heater fuel flow not available	Same as SLTO	Not Affected. If pump seizes, maximum available F_n limited to SLTO conditions.	Same as SLTO	None If pump seizes, same as SLTO if maximum F_n desired.
Fuel Seal 25.6.2	Seals Turbine end of Main Pumping Shaft	Bellows Failure	SLTO: Performance not affected. There will be a loss of fuel through the overboard drain. Maximum fuel loss approximately 100 pph.	Excessive overboard drain fuel leakage.	Not affected.	CR	None
			Cruise: Same as SLTO	Same as SLTO	Not affected.	Same as SLTO	None
			Landing: Same as SLTO	Same as SLTO	Not affected.	Same as SLTO	None
Metering Orifice 25.6.3	Meters Fuel supply to bearings for lubrication and cooling	Contamination	SLTO: Possibility of contamination remote because orifice is protected. If contamination occurs to the extent that complete fuel flow loss occurs, bearings will eventually fail and pump may seize resulting in loss of duct heater fuel flow. Pump controller will schedule pump turbine air supply control valve to full open position.	If pump seizes, loss of duct heater fuel flow	If pump seizes, duct heater will flame out and augmentation will be lost due to loss of fuel flow. Engine bleed air to pump turbine will be increased due to full open position of air control valve $F_n = 60\% F_{nma}$	If pump seizes, AP and CR	If pump seizes, reduce to and/or maintain non-augmented PLA range. Adjust F_n level on unaffected engines to obtain desired aircraft conditions.

Table 1 (continued)

JTF17 FAILURE MODE & EFFECT ANALYSIS

Sheet I — No. 2 of 2

Duct Heater Fuel Pump (Continued)

Item	Function	Failure Mode	Failure Effect on Subsystem	Method of Detection	Failure Effect on Engine	Failure Effect on Aircraft	Crew Action Required
Metering Orifice 25.6.3 (Cont.)			Cruise: Same as SLTO	Same as SLTO	Same as SLTO In addition oil temperature will increase and may eventually exceed limits due to loss of duct heater oil cooler fuel flow. $F_n = 20\% F_{nma}$	Same as SLTO Also may have IFS	If pump seizes, reduce to and maintain non-augmented PLA range. Adjust F_n level on unaffected engines to obtain desired aircraft conditions. Monitor engine oil temperature. May be necessary for IFS and reduction in aircraft Mach No. to prevent exceeding oil temperature limit.
			Landing: Not Affected. If pump seizes, duct heater fuel flow not available.	Same as SLTO	Not affected. If pump seizes, maximum available F_n limited to SLTO conditions.	Same as SLTO	None If pump seizes, same as SLTO if maximum F_n desired.
Splined Connector 25.6.4	Connects Inducer Shaft to Main Shaft	Shear	SLTO: Inducer becomes inoperative. The main stage will continue to operate.	None	Not affected as long as aircraft boost pumps operate.	Not Affected.	None
			Cruise: Same as SLTO	None	Same as SLTO	Not Affected.	None
			Landing: Same as SLTO	None	Not Affected. Maximum F_n is available as long as aircraft boost pumps operate.	Not Affected.	None
Bearing (4 Antifriction) (3 Sleeve) 25.6.5	Support Inducer	Seizure	SLTO: Inducer drive will shear and inducer becomes inoperative. The main stage will continue to operate.	None	Not affected as long as aircraft boost pumps operate.	Not Affected.	None
			Cruise: Same as SLTO	None	Same as SLTO	Not Affected.	None
			Landing: Same as SLTO	None	Not Affected. Maximum F_n is available as long as aircraft boost pumps operate.	Not Affected.	None
Gear Train 25.6.6	Reduces Inducer speed relative to pump speed.	Failure resulting in seizure of inducer	SLTO: Inducer becomes inoperative. The main stage will continue to operate.	None	Not affected as long as aircraft boost pumps operate.	Not Affected.	None
			Cruise: Same as SLTO	None	Same as SLTO	Not Affected.	None
			Landing: Same as SLTO	None	Not Affected. Maximum F_n is available as long as aircraft boost pumps operate.	Not Affected.	None

O. ENGINE CONTROL SYSTEM-ANALYSIS OF RELIABILITY, MAINTAINABILITY, STANDARDIZATION, SERVICE, SAFETY, VALUE ENGINEERING, AND HUMAN ENGINEERING

1. Goals and Objectives

Reliability, maintainability, serviceability, standardization, safety, value engineering, and human engineering goals for the design and development of the fuel, hydraulic, and ignition system components have been established to meet the predicted system goals at maturity which are:

1. Each component will be replaceable on the engine in 30 minutes or less as demonstrated in figure 1 for the unitized fuel control, figure 2 for the gas generator fuel pump, and figure 3 for the hydraulic fuel pump. Figure 4 shows these components in their normal installed locations on the engine.
2. Provide airframe and personnel safety by, (a) limiting engine compressor overspeed during emergencies, and (b) incorporating interlocks to assure engine sequence from forward to reverse at safe conditions and prevent inadvertant selection of reverse or forward thrust.
3. Provide rapid service and minimum cost at overhaul.
4. A premature engine removal rate chargeable to the fuel and control system not to exceed 0.010/1000 hours.
5. In-flight shutdowns chargeable to the fuel and control system not to exceed 0.005/1000 hours.
6. Fuel filters designed such that proper installation of filter elements during initial assembly and each reassembly after inspection is assured.
7. Throughout the design and development activity at P&WA, the P&WA standardization program, as explained in detail in Volume IV, Report F, Section V, will be employed toward meeting the objectives of improved reliability, improved maintainability, reduced costs, assured quality, and assured safety. P&WA will exercise control of subcontractor standardization, as explained in detail in Volume IV, Report F, Section V, through the joint efforts of the P&WA Engineering, Quality Assurance, Manufacturing, and Purchasing Departments.

Field maintenance and installation goals are obtained by providing:

1. A unitized control package in which all control engine functions (gas generator control, duct heater control, nozzle area control) are contained within one package.
2. A unique quick-disconnect packaging concept in which all engine fuel control connections are contained in a single manifold. This manifold is attached to the engine as a separate detail, and the control is assembled to the manifold with a minimum number of mounting bolts. This concept permits rapid control removal and installation as no plumbing connections are required to install and remove the control from the engine. Similar rapid removal concepts are employed for the shutoff lever, power lever, and exhaust nozzle area feedback pulley connections to the control.

3. Accessible and easily removable gas generator and duct heater control valves.
4. Accessible engine trim adjustments and fuel filters.

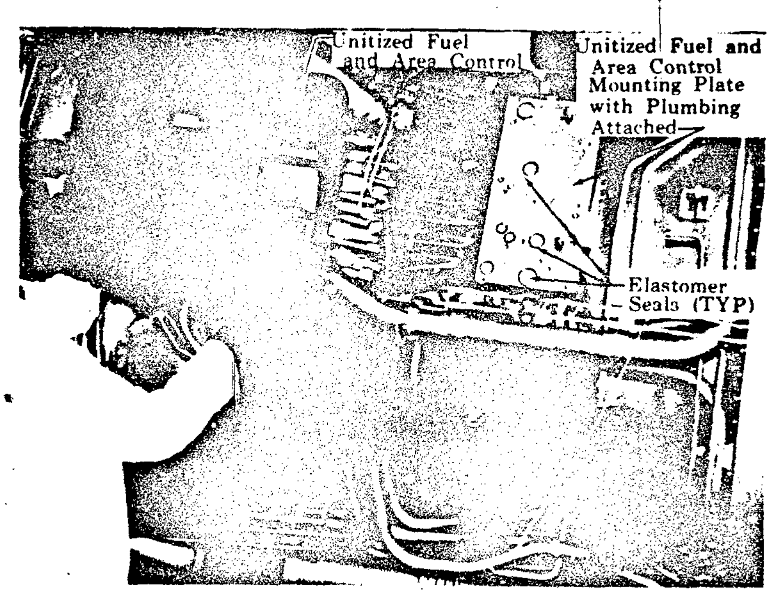


Figure 1. Unitized Fuel and Area Control Mounting FD 16734 BIII

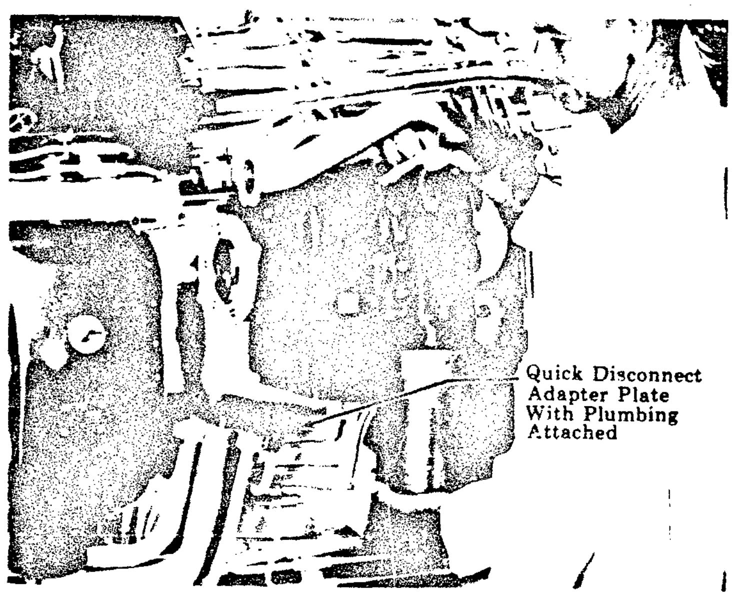


Figure 2. Gas Generator Fuel Pump Mounting FD 16738 BIII

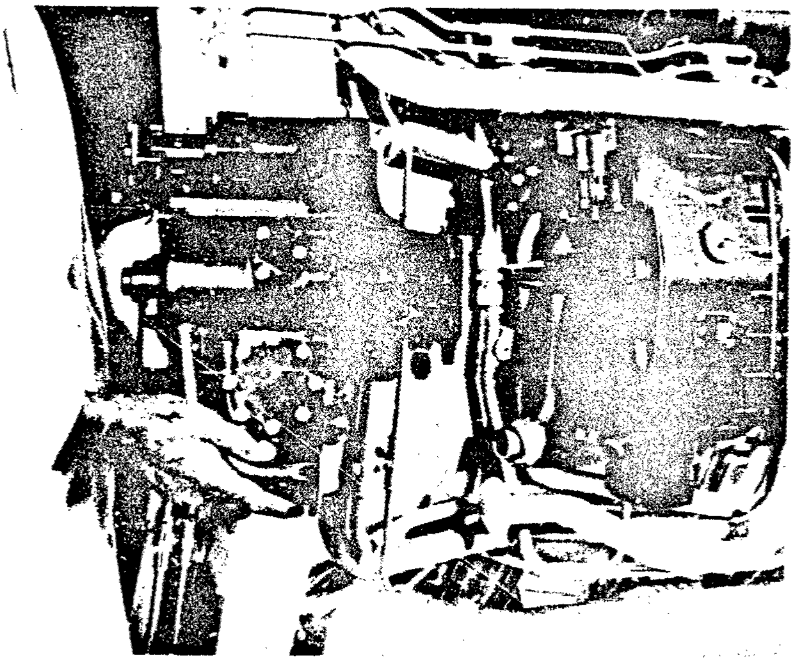


Figure 3. Hydraulic Fuel Pump Mounting FE 60754
BIII

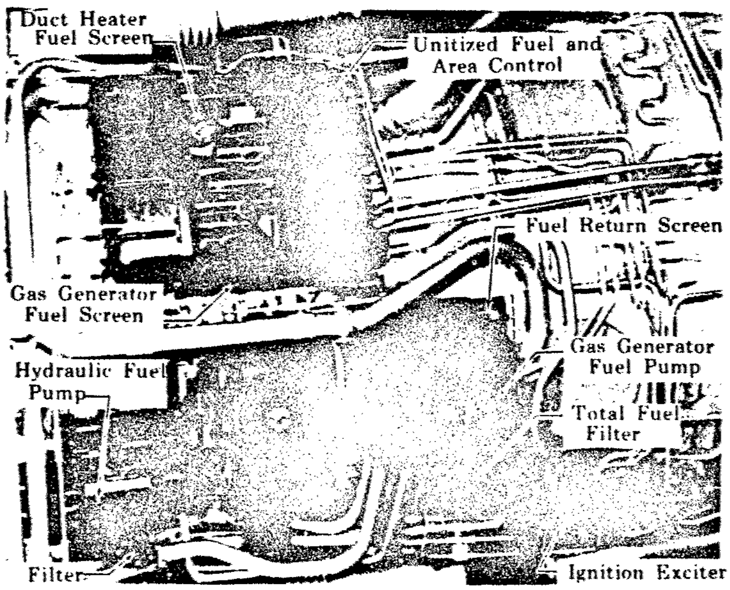


Figure 4. Control Installation Mockup FD 16735
BIII

Overhaul goals were obtained by designing a control with a modular concept that certain components or sections of the control may be easily removed with a minimum of calibration. This modular construction will also reduce calibrating time because the various sections of the control can be calibrated before assembly into the completed units. Control assembly and test will be facilitated by the modular concept in that the computer housing and removable systems can be dry calibrated using a checking fixture.

Overhaul costs will be minimized through:

1. Elimination of costly valves-in-housing matching operations through use of the removable sleeves.
2. Use of high temperature elastometric seals rather than metal seals which would require added housing machining to produce proper sealing surfaces.

While the unitized control is discussed above, P&WA will provide the same general concepts in the other control components. P&WA will maintain direct contact with its vendors and the corresponding organizations to assure that reliability goals are met in the most expeditious and practical manner. Scheduled meetings at no less than one-month intervals will be held to establish the status of compliance toward meeting these goals. They will also be reviewed at critical phases, i.e., design phase-initial design layouts completed, prototype drawings completed, and final production drawings completed.

Development activity, including bench, engine, and flight testing will provide the data to prove the design relative to these goals, and modifications will be incorporated as indicated by these tests to assure these goals as an actuality.

Established quality assurance procedures at both the vendor and P&WA will provide control of the parts to meet the design requirements throughout the program.

The Human Engineering and Value Engineering Programs outlined in Volume IV, Report D and Report F of this proposal will be followed throughout all phases of the design and development of the fuel system components described herein, both at P&WA and at the applicable vendors.

2. Phase III Program

Pratt & Whitney Aircraft has established a requirement with its vendors which specifies that the vendors must establish a reliability program for the Phase III SST activities that makes use of applicable paragraphs of Specification MIL-STD-785. This program establishes that P&WA will supervise the vendor (s) design and development activity to assure that the vendor will design and produce a unit which (1) provides for the best possible reliability, maintainability, serviceability, provisions for safety, value engineering, and human engineering, and (2) meets the reliability and maintainability goals of the component that have been established for that unit.

The reliability requirements placed upon fuel, hydraulic, and ignition system component vendors by P&WA are specified in Paras. 6.1, 6.2, and 6.3 of the typical procurement specification summarized in paragraph Q.

Data generated by the vendors in accordance with this requirement are submitted to the P&WA Reliability group and the cognizant project engineer for integration into the complete system and for review and corrective action where required.

P. ADVANCED TECHNOLOGY AND GROWTH CONCEPTS

The following advanced technology and growth pump and control concepts are being considered for application to the JTF17 engine because they appear to offer certain advantages.

1. A unitized engine fuel pump
2. A hydraulic computer control.

The current design and development status of these components is such that they are not being considered for the initial engine configuration. However, these systems appear to be more simple than the current state-of-the-art design being used and ultimately could achieve even higher reliability levels and lower overhaul costs than the present designs. For this reason, these concepts will be thoroughly evaluated as a part of this program and if the results are favorable, the development of these designs on an alternative basis will be considered with the intent of incorporating them into the basic engine configuration later in the program.

1. Unitized Engine Fuel Pump

a. Description and Vendors

Thompson-Ramo-Wooldridge, Chandler Evans Corporation, and Pesco Products Division are active in conceptual design and proposal programs to provide a satisfactory unitized fuel pump design for the JTF17 engine.

The unitized fuel pump concept combines all the pumping elements required to supply the JTF17 engine fuel and hydraulic systems into one pump, utilizing one common centrifugal boost element to supply the high pressure stages. Gas generator, duct heater, and hydraulic system fuel flow and pressure requirements are supplied as required by this unitized pump.

Various combinations of centrifugal stages, including "vapor core" stages, gear stages, piston stages, and vane pump stages are being considered in conceptual design and performance studies. Engine drive, compressor bleed air turbine drive, and compressor bleed air Rootes blower drive are being considered.

Potential improvements of the unitized concept include:

1. Simplified pump design and reduced weight resulting from a single pump compared to three separate pumps
2. Increased time between overhaul and reduced overhaul cost
3. Simplified engine plumbing
4. Reduced initial cost
5. Increased reliability through reduced number of parts.

b. Proposed Program

P&WA will review the various conceptual designs and proposals submitted by the vendors for confirmation of the potential improvements listed above

and will review the designs for effective solution of problem areas. If the design and proposal review indicates that a satisfactory unitized pump can be developed, considering performance, reliability, maintainability, cost and producibility, P&WA will consider undertaking a program to incorporate the unitized concept in the JTF17 engine.

2. Hydraulic Computer

a. Background

The hydraulic computer control performs most computations with a series of contoured rotating pilot valves that permit scheduling by hydraulic principles. The hydraulic computer control concept promises a smaller and lighter control package coupled with a more flexible control system. The mathematical comparison of a hydraulic computer with a hydromechanical control for the same application indicates that the hydraulic computer may have a potential reliability level significantly better than the current level of the hydromechanical control. The hydraulic computer has the potential of providing a sophisticated control with more flexibility and increased simplicity to perform the mathematical computations required to combine sensed parameters into the desired unique control parameters. This is accomplished with a minimum of mechanical complexity while achieving desired engine performance. The need for computing levers or cams has been reduced by performing most computations with a series of contoured rotating pilot valves. The mechanical execution of this type of control is similar to the methods that have proved to be successful in the past. The potential advantages of this approach may be found in such areas as packaging, manufacturing costs, weight, flexibility of design, potential for improved accuracy, and overall simplification.

b. Hydraulic Computer Development Experience

Both Bendix and Hamilton Standard have studied this control concept. The bulk of the development has been done by Hamilton Standard under Navy Contract NOw64-0691-di. Controls constructed using the hydraulic computer concept have accumulated over 170 hours of engine time and 4597 hours of bench time.

(1) Breadboard Experience

A series of component tests were conducted by Hamilton Standard to confirm the fundamental design concept. Upon completion of these, a breadboard capable of controlling the T55 gas generator was designed and manufactured. An engine test program was then conducted on a T55 engine, which consisted of a series of engine starts, accelerations, steady-state operation at various power levels, and decelerations. Engine starts were normal and speed governing was excellent at all levels of power throughout the range of the engine, with no instability noted. Accelerations and decelerations were accomplished with no indicated problems. In general, the series of engine tests was completed in a trouble-free manner.

(2) Prototype Experience

Upon completion of the breadboard testing described above, a prototype control capable of controlling an engine was constructed. After substantial test stand operation had established that the prototype hydraulic computer exhibited the proper fuel control characteristics under laboratory conditions, a test program was initiated to demonstrate the control's capability under engine operating conditions.

The prototype was modified to provide sea level schedules for the P&WA J60 engine and was physically installed on that engine for a 150-hour engine typical flight cycle test. The program demonstrated the control's ability to start, accelerate, maintain steady-state speed, and decelerate the engine.

c. Proposed Program

A preliminary control schematic applying this control concept for the JTF17 engine has been completed and is presented in figure 1. Initial weight studies have been initiated, and design and analytical studies are being conducted.

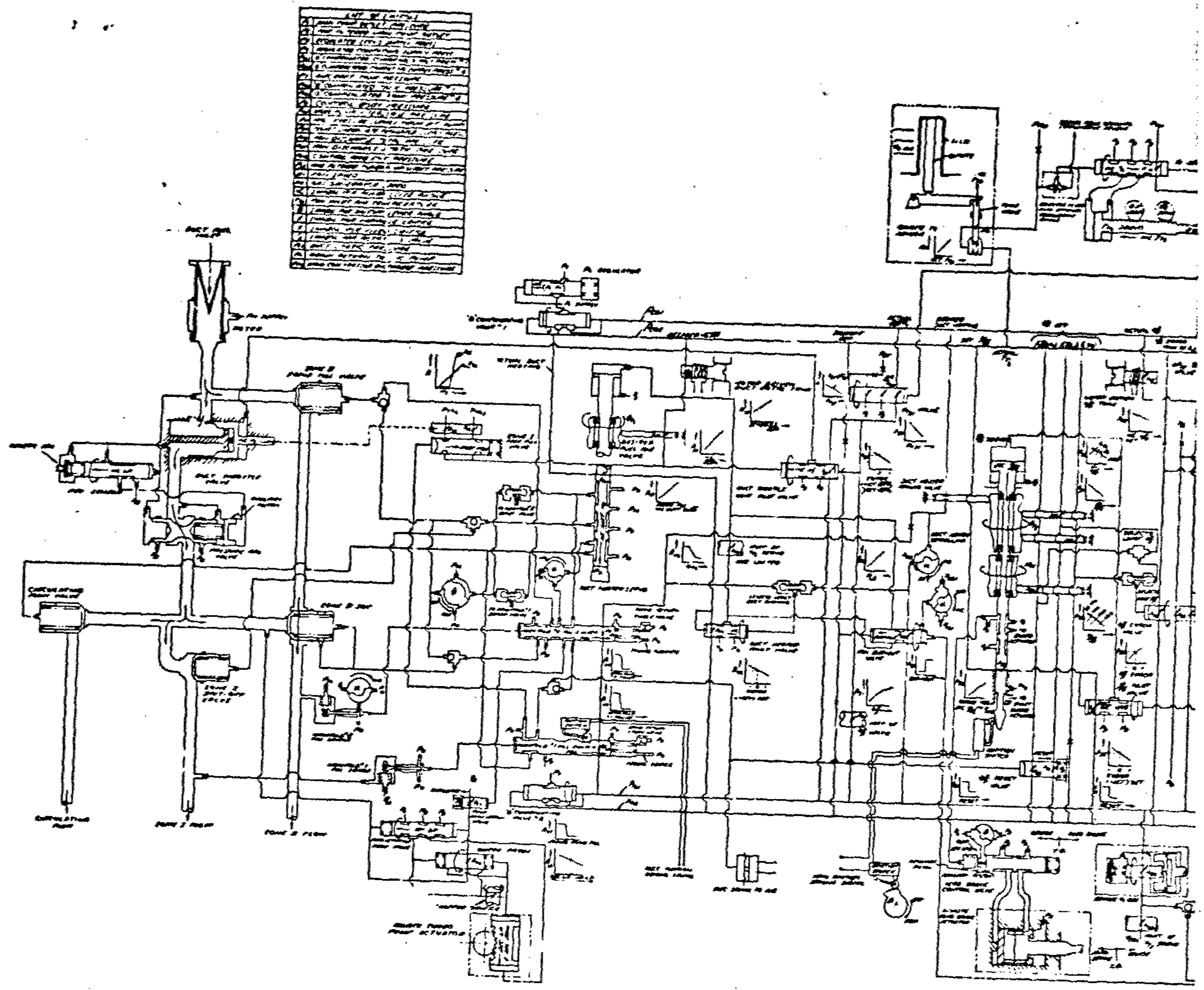
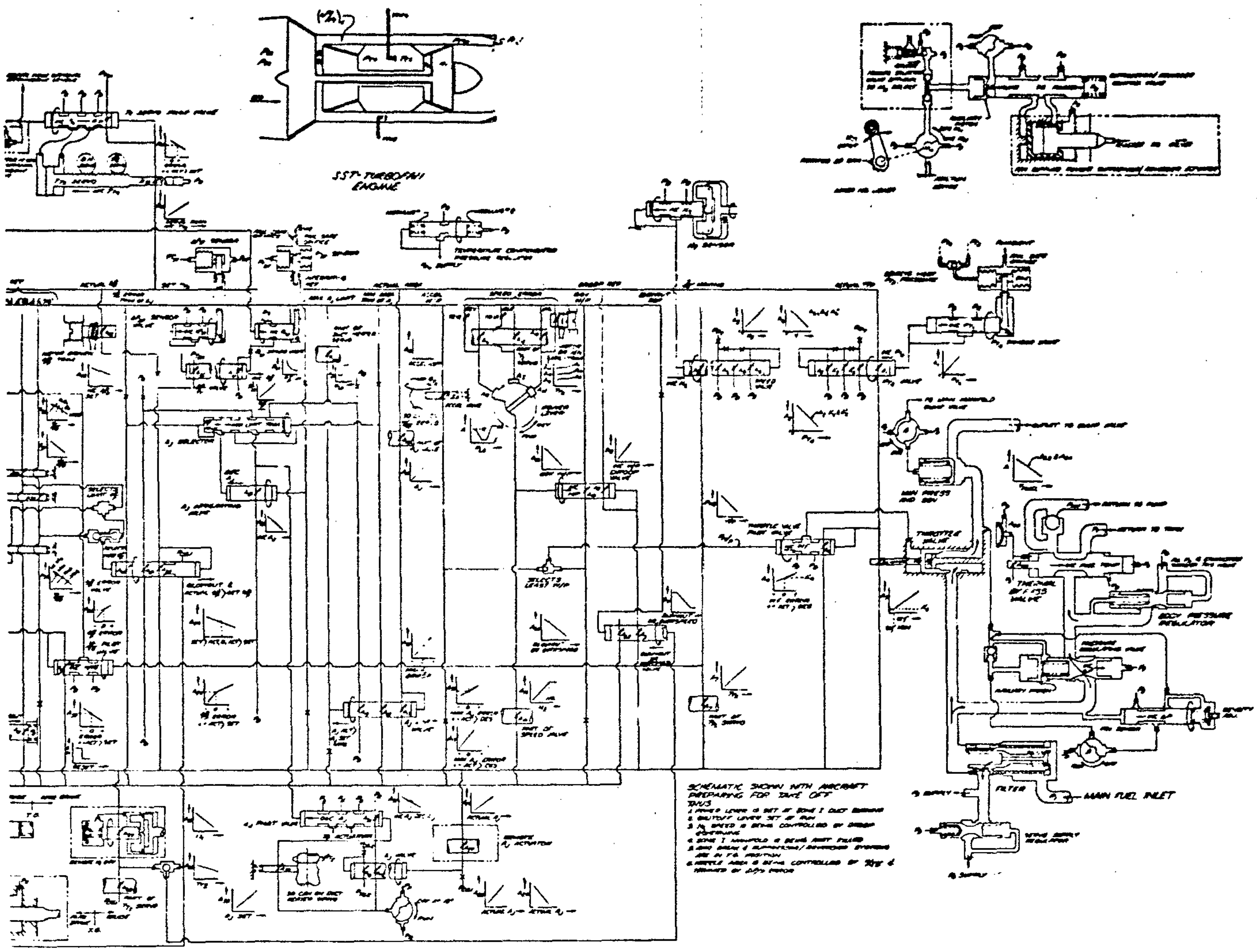


Figure 1. P&WA JTF17 Engine Hydraulic Computer Control Schematic


SST-TURBOFAN ENGINE
SCHEMATIC SHOWN WITH AIRCRAFT PREPARING FOR TAKE OFF
MAIN FUEL INLET

Q. COMPONENT PROCUREMENT SPECIFICATIONS

1. Component Procurement Specifications

The major components of the JTF17 engine control system will be procured from component vendors utilizing component procurement specifications. These specifications establish the requirements for the components and include items such as a description, design requirements, performance and operational requirements, inspection and test procedures, reliability requirements, and vendor responsibilities. A summary of Specification PWA PPS J112 for the Unitized Fuel and Area Control is included in this section of the proposal. The control schedules presented in paragraph D are typical of the data which will be part of the specification. Table 1 presents a tabulation of procurement specifications for the other control system components to be procured from vendors as complete assemblies. These are available for review upon request.

Table 1. JTF17 Component Procurement Specifications

Specifications	Component
PWA-PFS-J107	Gas Generator Fuel Pump
PWA-PPS-J108	Ignition System
PWA-PPS-J109	Hydraulic Pump
PWA-PPS-J110	Duct Heater Fuel Pump
PWA-PPS-J113	Engine Pressure Ratio Control
PWA-PPS-J116	Automatic Restart Ignition Switch
PWA-PPS-J117	Igniter
PWA-PPS-J118	Position Switch
PWA-PPS-J119	Exhaust Nozzle Position Transducer

2. Differences Between Prototype and Final Production Specifications

Purchase Specification for the JTF17 component includes requirements for both the prototype and final production units. Separate specifications are not prepared for prototype and production units.

"SUMMARY"
PRATT & WHITNEY AIRCRAFT
PURCHASE SPECIFICATION
J112
FUEL AND AREA CONTROL
FOR
JTF17 ENGINE

1. SCOPE

This specification establishes the requirements for the fuel and area control to be used on the JTF17 engine. Any deviation from this specification must be coordinated with Pratt & Whitney Aircraft, Florida Research and Development Center.

2. DESCRIPTION

2.1 The control system shall consist of a unitized hydromechanical system operating to:

1. Control engine speed between idle and maximum nonaugmented power.
2. Schedule gas generator fuel flow within desired limits.
3. Schedule and sequence duct heater fuel flow supplied from an air-driven turbopump, over the operating envelope between augmentation cutoff and maximum augmentation, to two individual fuel manifolds.
4. Position the duct exhaust nozzle area to maintain the desired engine inlet airflow.
5. Signal the compressor bleed valves to improve compressor stall margin.
6. Position compressor inlet guide vanes in each of two of three discrete positions.
7. Position a reverser-suppressor in two of three discrete positions.

2.2 The unitized control system, as specified herein, shall consist of the following subassemblies:

1. Unitized Fuel and Area Control which shall contain means for:
   a. Gas Generator Fuel Meter
   b. Duct Heater Fuel Meter
   c. Duct Heater Exhaust Nozzle Control
   d. Compressor Inlet Guide Vane Control
   e. Reverser-Suppressor Control
2. Duct Heater Pump Controller
3. Engine Inlet Temperature Sensors
4. Compressor Bleed Control Valve

3. APPLICABLE PUBLICATIONS

3.1 The applicable specifications and standards listed in ANA Bulletin No. 343q and 400u shall form a part of this specification to the extent specified herein.

3.2 The following specifications and publications shall form a part of this specification to the extent specified herein:

1.	PWA 300-D	Control of Materials
2.	PWA 381	Shipping Closures
3.	PWA PS 356C	Vendor Responsibilities
4.	MIL-E-5009B	Qualification Tests for Turbojet Aircraft Engines
5.	MIL-F-5624E	Jet Fuel (JP-4, 3, 5)
6.	MIL-E-25656	Jet Fuel (JP-6)
7.	MS 33586A	Dissimilar Metals
8.	PWA-QA-6068A	Quality Assurance Specification
9.	PWA 533	Fuel, Aircraft Turbine Engine
10.	MIL-S-8879	Screw Threads
11.	MIL-STD-785	Reliability Program for Systems, Subsystems, and Equipment
12.	MIL-M-26512C	Maintainability Program Requirements for Aerospace Systems and Equipment
13.	MIL-S-38130	General Requirements for Safety Engineering
14.	MIL-I-27686	Anti-Icing Fluid
15.	MIL-F-5161E	Jet Fuel (JP-5)
16.	QQ-M-151a-4	Salt Solution
17.	PWA 522	Jet A, A-1 Fuel
18.	PMC-9041	Calibrating Fluid

4. REQUIREMENTS

4.1 General Requirements - This section covers those applicable requirements not covered in the other sections of this specification. Items include such things as materials, standard parts, identification, etc.

4.2 Test Requirements - The applicable test requirements covering the type of tests to be conducted and shipment of prototype and production units prior to qualification are included in this section.

4.3 Design Requirements - Specific requirements, applicable to the design of the control, are covered in this section. Requirements prohibit the use of certain hardware items such as "Z" wires, snap rings, laminated shims, etc. Specific information pertaining to fuel characteristics and operating conditions, fuel filtration and contamination is included. Required control contamination capability is as follows:

Fuel Contamination - The control shall be designed to operate satisfactorily on fuel containing the following contaminant. The fuel shall be contaminated to the extent of eight grams of foreign matter per 1000 gallons. This foreign matter shall be considered to consist of not less than 68% SiO_2 and shall have a particle-size analysis as follows:

Particle Size Microns	Percent of Total
0 - 5	39 ± 2 by weight
5 - 10	18 ± 3 by weight
10 - 20	16 ± 3 by weight
20 - 40	18 ± 3 by weight
Over 40	9 ± 3 by weight
Through a 200-mesh screen	100 by weight

A detailed description of the mechanical and ambient environment, control external connections, and functional control requirements are covered.

A unique design requirement of the JTF17 fuel and area control is the existence of an interface whereby the engine airflow control and associated scheduling may be implemented with either a hydromechanical or electronic system. These systems will be interchangeable on the prototype controls.

4.4 Performance and Operational Requirements - This section provides the specific operating schedules, accuracy limits, applicable adjustments with required range, and schedule changing requirements. Schedules include such things as acceleration fuel flow, deceleration fuel flow, engine airflow, duct heater fuel flow, etc.

5. INSPECTIONS AND TEST PROCEDURE

Requirements for control acceptance inspection and specific test to be conducted are covered in this section. Inspection requirements include inspection techniques, instrumentation accuracies, and P&WA inspection representation. Test procedures describe the specific requirements of each test to be conducted and the limits governing the tests. Typical tests described are the 75-hour Hot Mission Cycle Bench Test, the 550-hour Bench Test including hot mission cycle and contaminated fuel testing, and the Production Acceptance Bench Test.

6. SUBCONTRACTOR RELIABILITY REQUIREMENTS

Specific subcontractor reliability, maintainability, and durability requirements are outlined in this section.

6.1 General - The vendor shall be expected to have or to establish and implement a reliability program that includes but not limited to the applicable paragraphs of MIL-STD-785.

The vendor shall recognize the concept of inherent reliability of design, i.e., that reliability is limited by design, and that effort must be concentrated early in the design phase.

6.1.1 Surveys - P&WA reserves the right to conduct any surveys deemed necessary to adequately evaluate the vendors capabilities to meet the specification requirements.

6.1.2 Numerical Reliability Goals - The unitized fuel and area control shall have a design which is consistent with the following goals:

Mature engine capability after approximately 5 years should be:

1. A mean-time-between-overhaul (MTBO) of 6000 hours.
2. A component-chargeable premature engine removal rate of 0.0058 per 1000 engine hours.

6.1.3 Reliability Tests - No special component reliability test shall be accomplished specifically and solely to acquire reliability data to verify the requirements of paragraph 6.1.2.

6.1.4 Demonstration of MTBO - Demonstration of the Mean-Time-Between-Overhaul as specified in paragraph 6.1.2 shall be accomplished by the conclusion of Phase IV flight testing.

6.1.5 Demonstration of Premature Engine Removal Rate - Demonstration of the premature-engine-removal-rate as specified in paragraph 6.1.2 shall be accomplished by the conclusion of Phase IV flight testing.

6.1.6 Failure Definition - For purposes of determining conformance to the MTBO and per requirements of paragraph 6.1.2, a failure is defined as an engine chargeable event or an unscheduled removal of the complete engine from the aircraft prior to the expiration of the assigned TBO interval which results in the following:

1. Inability to achieve a satisfactory engine start within 15 minutes from initiation of start sequence.
2. In-flight detected failure of equipment not furnished by the engine contractor, which occurs as a result of failure of the component to provide a proper function within the limits of this specification.

3. A sustained inability to allow the engine to obtain proper thrust ratings that are appropriate for the following conditions:

    a. Takeoff Thrust Setting

    b. Maximum nonaugmented rating and maximum augmented rating during the climb portion of the mission

    c. Normal thrust at any indicated flight condition within 10%.

4. A condition which causes or generates a decision to shut down an engine or retard the throttle to reduce engine thrust greater than 10% of the proper desired value. Engine flameouts (even if a successful restart is achieved) are included if the flameout is chargeable to the engine.

5. Any in-flight inability to operate the engine at performance levels to complete the mission caused by or attributed to the unitized fuel and area control being out of performance levels and limits specified in this specification.

6. Component failures or malfunctions which cause malfunctions in the oil system and/or cause high vibration levels of the engine.

6.2 Maintainability Requirements - The vendor shall be expected to establish and implement a maintainability plan using MIL-M-26512 as a guide.

6.2.1 Maintainability Objectives - The vendor's maintainability program shall direct effort at integrating maintainability concepts in all system analysis performed to this purchase specification.

6.2.1.1 Maintenance Manhours - The unitized fuel and area control (including control, T_{t2} sensors, duct pump control, and compressor bleed control valve) shall have a goal of an average number of manhours to perform the scheduled inspections as follows:

Function	Manhours Required Per Engine
Preflight Inspection	1/6
Postflight Inspection	1/6
Removal and Replacement	3

6.2.1.2 Frequency of Maintenance Actions - The unitized fuel and area control shall have a final goal for frequency of maintenance action of 15 per 1000 engine hours for components used for engines after 5 years of service. The frequencies for maintenance actions shall be consistent with reliability goals.

6.2.2 Maintainability Analysis - A maintainability analysis on the unitized fuel and area control shall be submitted to P&WA engineering prior to initiation of the Formal Qualification Test.

6.2.3 Verification of Maintainability Goals - The requirements for maintainability shall be verified by timed maintenance operations.

6.2.4 Maintainability Reports - Maintainability program status reports shall be submitted to P&WA engineering at intervals not to exceed 3 months.

6.3 Design Review

6.3.1 General - The reliability program shall include milestones for reliability review and evaluation by P&WA engineering and the vendor of significant designs before they are finalized. Three design reviews will be scheduled by P&WA before the milestone design configuration is frozen for production.

6.3.2 Failure Mode and Effect Analysis - This is an analysis of all potential failures and/or malfunctions of all of the components and parts of the end item considering their application and use and their effects on mission capability. The FMEA is an engineering design parameter in the same sense that other product characteristics are design parameters. Accordingly, the procedures used for accomplishing the FMEA parallel those used in arriving at other part characteristics of parts or materials and from stresses anticipated in their application.

6.3.3 Failure Mode and Effect Analysis Completion - The Failure Mode and Effect Analysis as described in paragraph 6.3.2 shall be completed after the design layout or schematic is accomplished.

6.3.4 Failure Classification - This is for the purpose of assessing the design from a criticality point of view by classifying the assumed failure modes in four general categories minor, major, critical, or catastrophic. In mechanical equipment a component or part failure need not be a critical system failure and should be so indicated. The purpose here is to properly classify each failure effect at the system level and then apply reliability data based on field experience to relate the failure mode to probability or chance of occurring within the mission time and/or the useful design life of the component or part.

6.3.5 Critical Items List - The critical items list shall be made from the failure mode and effect analysis and shall be a summary of those components or parts whose single failure results in the probability of vehicle loss (starting critical and flight or vehicle critical).

6.3.6 Records - The vendor shall be expected to maintain records of these documents which shall be continuously updated consistent with the design phase. They shall also be updated with design changes and/or corrective action taken on problems occurring during development and field testing.

6.3.7 Review and Approval - The vendor shall notify P&WA engineering when the design layout phase is complete and a review may be made.

6.3.8 Support Data - The vendor shall be required to submit to P&WA engineering, support data consistent with the design phase at least two weeks in advance of the scheduled review date.

7. VENDOR RESPONSIBILITIES

Vendor responsibilities relating to design changes, supplying parts for change substantiation, support for the development program, and engineering drawings are covered in this section.

R. INSPECTION, QUALITY ASSURANCE, AND ACCEPTANCE PROCEDURES AND CRITERIA

1. Major Control System Components

P&WA oversees each vendor's quality control and inspection of detail parts, subassemblies, and assemblies, and the acceptance test procedures that pertain to those assemblies to assure that the standards established during the design and the development phases of each program are fully adhered to at the vendor's facility. The vendor creates an acceptance test specification, which must be approved by P&WA engineering, and is created to demonstrate the compliance of each component to the functional requirements of the P&WA procurement specification. Satisfactory compliance to the limits of testing required therein are observed by the P&WA vendor quality assurance inspector to provide constant control of the components being shipped. Inspection of the component to the installation drawing to assure physical conformance is also observed by the P&WA vendor quality assurance inspector. The components, upon arrival at P&WA, are again reviewed in detail, this time by P&WA receiving inspection for physical compliance to the installation drawing. Following acceptance of the physical inspection the unit is functionally bench tested at P&WA for compliance to a component calibration schedule which is the P&WA engineering equivalent of the vendor's test specification. The P&WA bench production acceptance testing of the vendor furnished components is continued until consistent correlation of data shows that the vendor test demonstrates acceptable unit quality. The P&WA acceptance test is then placed on a sampling basis. The final acceptance of all components is dependent upon successful engine operation.

A more detailed description of the P&WA Vendor Quality Assurance Control System may be found in Volume IV, Report F, Product Assurance, Section III; and the Configuration Management Plan in Volume V, Report C, Section VI.

The following table is a list of the component calibration schedules which are on file at P&WA and are available upon request. Following the table is a sample of a P&WA Component Calibration Schedule (CCS) which provides the acceptance test requirements for the gas generator fuel pump.

Table 1. P&WA Component Calibration Schedules

CCS No.	
115	Unitized Fuel Area Control
116	Gas Generator Fuel Pump
117	Ignition Exciters
118	Hydraulic Fuel Pump
119	Automatic Restart Switch
120	Position Indicator Switch
121	Ignition Ignitors
122	Exhaust Nozzle Position Transducer
123	Duct Heater Fuel Pump
124	Engine Pressure Ratio Control

This Schedule is Applicable For
Vendors When Referenced in P&WA
Purchase Specification or Directed
by P&WA Engineering

CCS 116

PRATT & WHITNEY AIRCRAFT
FLORIDA RESEARCH AND DEVELOPMENT CENTER

COMPONENT CALIBRATION SCHEDULE
FOR
GAS GENERATOR FUEL PUMP
FOR
JTF17 ENGINES

1. QUANTITY TO BE TESTED

All of the gas generator fuel pumps, PWA P/N, shall be bench tested at P&WA in accordance with this Component Calibration Schedule.

2. GENERAL REQUIREMENTS

2.1.1 Flowbench - A flowbench having at least one accessory drive with a variable speed range of 500 to 4800 rpm. The drive motor power rating shall be at least 125 horsepower. Fuel flow handling capacity shall be at least 500 to 50,000 pph. Viewed from the anti-drive end, pump rotation is counter-clockwise.

2.1.2 Boost Pump and Pressure Regulator - A boost pump and pressure regulator capable of maintaining fuel pressure at the inlet of the fuel pump at 20$\pm$5 psig over a total fuel flow range of 500 to 50,000 pph. This system should also have the capability of being adjustable from 5 to 30 psig.

2.1.3 Heat Exchanger - A heat exchanger to maintain the fuel temperature at the pump and flowmeter inlets at 100° $\pm$5°F.

2.1.4 Filter - A filter containing a 10- to 20-micron element and installed in the stand line supplying fuel to the pump inlet.

2.1.5 Filter - A 60 to 100-mesh screen-type filter to catch any chips produced by the pump. This filter must be installed between pump discharge and stand return.

2.1.6 Test Fittings - Test Fittings to connect to the pump ports.

2.1.7 Hand Valves - Three hand valves to set pump discharge pressure, oil cooler flow and return flow as shown in figure 1.

2.1.8 Pump Mounting Bracket - A pump mounting bracket to support the pump at the mounting trunnion.

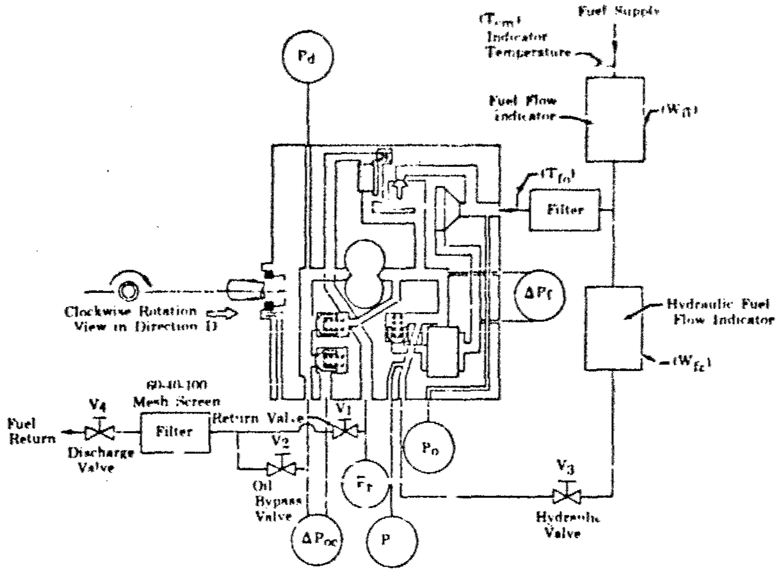


Figure 1. Pump Test Position and Schematic Diagram — FD 17170 BIII

2.1.9 Instrumentation - Instrumentation for taking the measurements listed below with the accuracy specified:

a. N_p (Pump drive speed) with a range of at least 500 to 4800 rpm and an accuracy of ± 5 rpm of indicated reading within this range.

b. W_f (Fuel flow) with a range of at least 500 to 10,000 pph and an accuracy of $\pm 1\%$ of indicated reading within this range.

c. P_o (Pump inlet pressure) with a range of at least 10 to 160 psig and an accurach of ± 2 psig within this range.

d. P_i (Interstage pressure after filters) with a range of at least 20 to 200 psig and an accuracy of ± 2 psig within this range.

e. P_d (Pump discharge pressure) with a range of at least 100 to 1500 psig and an accuracy of ± 10 psig within this range.

f. T_{fm} (Test fluid temperature at flowmeter inlet) with a range of at least 70°F to 110°F and an accurach of 2°F within this range.

g. T_{fo} (Test fluid temperature at pump inlet) with a range of at least 70°F to 110°F and an accuracy of 2°F within this range.

h. P_f (Pressure drop across main filter) with a range of at least 0 to 25 psid and an accuracy of ± 0.5 psi within this range.

i. P_r (Return pressure before strainer) with a range of at least 10 to 60 psig and an accuracy of ± 2 psig within this range.

j. P_{oc} (Oil cooler bypass valve) with a range of at least 20 to 50 psid and an accuracy of ± 1 psi within this range.

k. W_{f2} (Hydraulic pump flow) with a range of at least 3500 to 50,000 pph with an accuracy of $\pm 1\%$.

2.1.10 Blank Main Fuel Filter

2.1.11 Blank return flow strainer

2.2 Test Fluid - Test fluid shall be PMC 9041 Calibrating Fluid.

2.3 Depreservation - If the pump has been preserved with flushing oil, it must be depreserved by flowing a minimum of five gallons of PWA 533 or PMC 9041 Fuel through the pump.

2.4 Installation - The pump shall be mounted on the flowbench in a position similar to its mounting on the engine as shown in figure 1 or engineering approved equivalent.

2.5 Data Required

2.5.1 The following data shall be recorded on each data sheet:

a. PWA part number
b. Pump serial number
c. Vendor part number and revision numbers
d. Test fluid type

2.5.2 The following data shall be recorded when noted in the calibration procedure:

a. N_p (Pump drive speed)
b. W_{f1} (Fuel flow)
c. P_o (Pump inlet pressure)
d. P_i (interstage pressure)
e. P_d (Pump discharge pressure)
f. T_{fm} (Fuel temperature at flowmeter inlet)
g. T_{fo} (Fuel temperature at pump inlet)
h. P_r (Return press before strainer)
i. P_f (Press drop across main filter)
j. P_{oc} (Oil cooler bypass valve)
k. W_{f2} (Hydraulic pump flow)

2.6 Torque requirements - Torque requirements on component fittings shall not exceed the requirements specified in the Clearance Index Chart and Table of Limits. Torque requirements on component housing bolts, caps, etc., shall not exceed the limits when specified in this CCS.

2.7 Anti-Seizure Compound - Suitable anti-seizure compound shall be component fittings when the mating fittings do not incorporate silver plating. Anti-seizure compound shall also be used on component bolts, caps, and other threaded connections when reinstalling these parts after making adjustments or other engineering approved removals. This anti-seizure compound shall be used sparingly but adequately.

2.8 Voi-Shan Seals - Voi-Shan seals on component fittings shall be used in accordance with the engine assembly drawing specifications. Reuse of these seals is acceptable.

3. TEST REQUIREMENTS

3.1 General Test Conditions

3.1.1 Test Fluid Temperature - T_{fo} and T_{fm} shall be maintained at $100^{\circ}F \pm 5^{\circ}F$.

3.1.2 Pump Inlet Pressure - P_o shall be maintained at 20 ± 5 psig unless otherwise specified.

3.1.3 Return Valve (V_1) - All tests shall be performed with the return valve (Bypass from main fuel control) closed except 3.2.4, 3.2.5, and 3.2.6.

3.1.4 Setting Test Conditions - All test conditions shall be set by approaching the test point of the variable in the direction indicated. Any overshoot shall require setting of the variable in the proper direction.

3.1.5 Oil Bypass Valve (V_2) - All tests shall be performed with this valve open except 3.2.4 and 3.2.7.

3.1.6 Hydraulic Valve (V_3) - Shall be set to regulate W_{f2} to 3500 ± 50 pph at all points except 3.2.5.

3.1.7 Discharge Valve (V_4) - Shall be set in conjunction with V_1 to establish pump discharge pressure.

3.1.8 Pump Speed - Pump speed shall always be set in accordance with P_d as in figure 2 unless otherwise specified. After setting pump speed, P_d shall be increased to the value specified for the applicable test conditions. When starting the pump, the discharge valve shall be open enough to avoid exceeding figure 2 limitations. Pump speed shall not exceed 4800 rpm during this CCS.

3.2 Test Procedure

3.2.1 Leakage - At each of the following test points, record any shaft seal leakage over a 5-minute period and any external leakage. Record N_p, P_{oc}, P_f, P_r, W_{f2}, W_f, P_o, P_i, P_d, T_{fo}, and T_{fm} at the end of each test point below.

Test Point	Pump Inlet (P_o) psig	Pump Drive Speed (N_p) rpm	Pump Discharge Pressure (P_d) psig
1	50 ±5	500 ±5	200 ±5
2	20 ±5	4000 ±5	800 ±10

Limits: Shaft seal leakage shall not exceed 50 drops for the 5-minute period at each test point. There shall be no measurable external leakage at either test point.

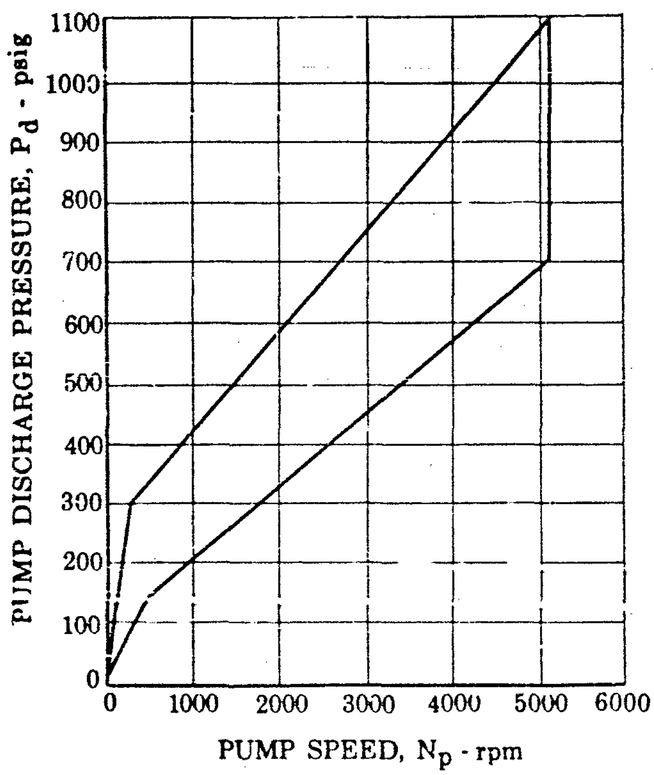


Figure 2. Pump Operating Envelope

FD 17171
BIII

3.2.2 Relief Valve Test

3.2.2.1 Relief Valve Flushing - To be done only when pump is known or suspected to have contained preservative oil immediately prior to accomplishment of this CCS. With N_p at 4350 ±20 rpm and P_d set initially in accordance with figure 2, slowly adjust the discharge valve (V_4) to cause the relief valve to open. Opening of the relief valve is evidenced by a reduction in flow through the pump. Do not increase P_d in excess of 1300 psig or allow fuel flow to decrease to less than 10,000 pph during this flushing procedure. Maintain the relief valve in the open condition for

15 ± 5 seconds. Reduce P_d to 400 psig for 30 seconds and then repeat the above procedure. No data is required during this test other than a log entry "Relief Valve Flushed."

3.2.2.2 Relief Valve Setting - This paragraph to be run only if the relief valve is adjusted or replaced. With N_p at 4350 10 rpm and P_d set initially in accordance with figure 2, slowly adjust the discharge valve (V_4) to set each of the following test points in the order shown. Record N_p, W_f, P_o, P_i, and P_d, at each test point except Point 3. At point 3 record only N_p, P_d, and W_f.

Test Point	Pump Drive Speed (N_p) rpm	Pump Discharge Pressure (P_d) psig	Fuel Flow (W_f) pph
1	4350 ± 15	1100 ± 10	Record
2	4350 ± 15	1200 ± 10	Record
3*	4350 ± 30	Record	500-600

*DO NOT MAINTAIN THE LOW FUEL FLOW CONDITION AT TEST POINT 3 FOR MORE THAN 10 SECONDS.

Limits: From test point 1 to test point 2, W_f shall not decrease more than 2000 pph.

At test point 3, P_d shall not exceed 1300 psig.

NOTE: If the calibration does not meet the specified limits, the relief valve preload must be reset.

3.2.3 Gear Stage Performance - Set the following test points in the order shown. Record N_p, W_{f1}, P_o, P_d, P_i, T_{fo}, P_f, W_{f2} and T_{fm} at each test point.

Test Point	Pump Drive Speed (N_p) rpm	Pump Discharge Pressure (P_d) psig	Minimum Fuel Flow Limits (W_f) pph
1	50 ± 5	210 ± 5	5500
2	2400 ± 5	245 ± 5	7500

NOTE: Read complete paragraph of Limits below prior to proceeding with test points 3, 4, and 5.

Test Point	Pump Drive Speed (N_p) rpm	Pump Discharge Pressure (P_d) psig	Minimum Fuel Flow Limits (W_f) pph
3	4000 ± 5	1000 ± 10	Record
4	4550 ± 5	1000 ± 10	40,000
5	4550 ± 5	300 ± 10	Record

Limits: W_f must be within the specified limits at each test point. P_d shall be maintained at 1000 ± 10 psig from 3500 rpm through test point 4 only.

W_f shall increase steadily with increasing rpm and the operation of the pump shall be observed for evidence of relief valve opening high frequency P_d fluctuation of more than 25 psi and constant or dropping W_f with increasing N_p are indication of relief valve opening. Evidence of relief valve opening shall not be cause for rejection but shall be brought to the attention of engineering.

3.2.4 Impeller Bypass Valve Operation - Open valve V_1 and close valve V_2. Without pump turning set W_{f1} to 90,000 pph. Record P_o, P_i, P_r, P_d, W_{f1}, T_{fo}, and T_{fm}.

Limits: P_i shall not exceed P_r in excess of 4 psi.

3.2.5 Impeller Performance - Using valves V_1, V_3, and V_4, set the following points in the order shown. Record W_{f1}, W_{f2} N_p, P_o, P_i, P_d, T_{fo}, P_f, and T_{fm} at each test point.

Test Point	Pump Drive Speed (N_p) rpm	Fuel Flow (W_{f1}) pph	Hydraulic Flow (W_{f2}) pph	Pump Discharge Pressure (P_d) psig
1	4550	540	3500	210 $\pm$10
2	4550	40,000	50,000	1000 $\pm$10

Limits: The impeller pressure rise (P_i-P_o) shall be neither less than 140 psid nor greater than 160 psid.

3.2.6 Main Filter Bypass Valve Operation

3.2.6.1 Install blank main fuel filter element.

3.2.6.2 Open valves V_1 and V_4, slowly increase boost pressure until flow begins through pump. Record W_{f1}, P_o, P_i, P_f, T_{fm} and T_{fo}.

Limit: P_f shall be 18 to 20 psid when flow begins.

3.2.6.3 Increase boost pressure until 90,000 pph flow is established. Record W_{f1}, P_o, P_i, P_f, T_{fm} and T_{fo}.

Limit: P_f shall not exceed 22 psid.

NOTE: Pump is not to be operative during these tests.

3.2.6.4 Remove blank main fuel filter element and install B-M element.

3.2.7 Oil Cooler Bypass Valve Operation - Close Valves V_1 and V_2 and set the following test points. Record N_p, W_{f1}, P_o, P_d, P_i, P_f, ΔP_{oc}, T_{fo} and T_{fm}.

Test Point	Pump Drive Speed (N_p) rpm	Pump Discharge Pressure (P_d) psig	Minimum Fuel Flow (W_{f1}) pph
1	550	210 ±5	5500
2	4550	1000 ±10	40,000

Limits: ΔP_{oc} shall be 30 ±1 psid.

3.2.8 Strainer Bypass Valve Operation.

3.2.8.1 Install blank return flow filter.

3.2.8.2 Set the conditions of test point 1 below with valve V_1 closed. Other points are then to be set by adjusting V_1 and V_4. Record N_p, W_{f1}, P_o, P_i, P_r, P_f, P_d, T_{fm}, and T_{fo} at each point.

Test Point	Pump Drive Speed (N_p) rpm	Pump Discharge Pressure (P_d) psig	Minimum Fuel Flow (W_{f1}) pph
1	4550 ±5	1000 ±10	40,000
2	4550 ±5	1000 ±10	1000 pph less than point 1
3	4550 ±5	335 ±10	13,000
4	4550 ±5	210 ±10	540

Limits: Strainer bypass valve pressure drop (P_r-P_i) shall be 20 ± 2 psid at points 2, 3, and 4.

3.2.8.3 Remove blank filter and install B/M filter.

3.3 Final Inspection

3.3.1 Pump Strainer Inspection

3.3.1.1 Remove the interstage strainer and inspect for chips or foreign material. If foreign material is found, engineering shall review and provide disposition.

3.3.1.2 Clean and reinstall the strainer and cover.

3.3.2 Leak Check - At each of the following test points, record any shaft seal leakage over a 5-minute period and any external leakage. Also record N_p, W_{f1}, P_o, P_i, P_d, T_{fm}, P_f, W_{f2}, and T_{fo} at the end of each test point.

Test Point	Pump Inlet (P_o) psig	Pump Drive Speed (N_p) rpm	Pump Discharge Pressure (P_d) psig
1	50 ±5	500 ±5	200 ±5
2	20 ±5	4550 ±5	1000 ±10

Limits: Shaft seal leakage shall not exceed 50 drops for the 5-minute period at each test point. There shall be no measurable external leakage.

3.3.3 Stand Strainer Inspection - At the conclusion of the test, the 60- to 100-mesh screen type strainer shall be checked for chips or foreign material. If any material is present, engineering shall review and provide disposition.

4. PRESERVATION AND STORAGE

After completion of testing, the pump shall be prepared for storage by preserving with MIL-O-6081B, grade 1010 oil. The oil must be supplied to the pump inlet at a 15 to 30 psig pressure. Rotate the pump shaft as shown in figure 1 at 100 to 500 rpm. Flow a minimum of 5 gallons of oil through the pump. Drain and install port covers. Protective covers, containers, and/or stands shall be used to prevent damage or contamination of the pumps.

5. APPLICABLE FIGURES

Figure 1 Pump Test Position and Schematic Diagram

Figure 2 Pump Operating Envelope

APPENDIX A

UNITIZED FUEL AND AREA CONTROL (HSD)

The following is a description of the schematic concept of the hydromechanical fuel control proposed by Hamilton Standard for the JTF17 turbofan engine.

The fuel control for the JTF17 engine is a hydromechanical unit utilizing a computing system common to many Hamilton Standard control systems. The control consists of three distinct but interconnected systems: a gas generator limiting and governing system, a two-zone duct heater system, and a duct exhaust area control system. The control outputs to these systems are computed from the measured inputs listed below and shown in figure 1.

1.	High rotor speed	N_2
2.	Engine inlet temperature	T_{t2}
3.	Primary combustor pressure	P_b
4.	Fan discharge total pressure	P_{T3}
5.	Fan discharge static pressure	P_{s3}
6.	Duct exhaust nozzle area	A_j
7.	Power lever angle	PLA
8.	Shutoff lever angle	SOLA

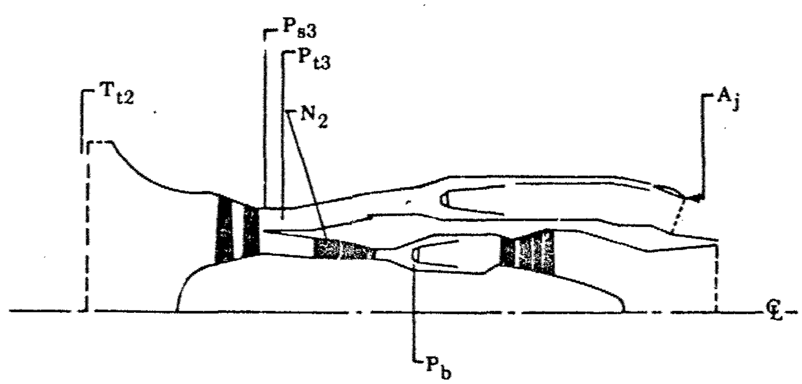


Figure 1. JTF17 Engine Control Parameters FD 17169 BIII

Two of these parameters, P_{T3} and P_{s3}, are used to compute the pressure ratio, $(P_{T3}-P_{s3})/P_{T3}$, which is utilized in the airflow control system.

In addition to the three integrated systems mentioned above, the control also signals:

1. Compressor bleeds as a function of N_2 and T_{t2}
2. Compressor inlet guide vanes as a function of N_2 and T_{t2}
3. Reverser-suppressor position as a function of PLA.

In addition to the external inputs measured by the control, the control system incorporates a number of sequencing and interlock devices to assure proper integration and operation of the total control system. To adequately discuss the control functions, the system will be divided into four major groups: (1) the gas generator control, (2) the duct exhaust

nozzle control, (3) the duct heater fuel control, and (4) the independent functions.

The description is based upon the control schematic shown in the integrated system layout in figure 2, paragraph B.

A. GAS GENERATOR CONTROL

1. T_{t2} Sensor

The engine inlet temperature, T_{t2}, sensing system utilizes a remotely mounted gas-filled bulb and bellows element. The bulb is mounted in the inlet air stream and converts a change in temperature of the inlet air to a change in pressure of the gas sealed in the bulb and bellows. The bellows in turn operates a flapper valve. The flapper valve is connected to a hydraulic signal transmission line supplied with main control regulated servo pressure. The remotely mounted flapper valve modulates the pressure in the signal line. The modulated pressure is carried through the signal transmission line to a sensing bellows in the main control body. The transmission line incorporates a double check valve to permit the use of a second remote sensor. The check valve system averages the two sensors or selects the higher computed T_{t2} if the difference between the two signals is indicative of a temperature difference greater than a particular valve.

2. T_{t2} Pilot Valve and Servo

The sensing bellows in the control body converts the pressure signal from the remote sensor to a force. This force is applied to a multiplying lever system that displaces a double-acting pilot valve. The pilot valve controls the position of the T_{t2} servo. As the servo is displaced it displaces a feedback roller that acts to restore the force balance on the multiplying lever.

To reduce the sensitivity of the servo to changes in fuel and ambient temperatures, a unique temperature compensation linkage was devised. The temperature compensation system utilizes two springs in opposition to generate the net position linkage load and two other springs to generate the net rate linkage load. The two pairs of springs are constructed of materials that have different temperature characteristics and consequently a different rate change with temperature. The springs will be preloaded such that a change in environmental temperature will not result in a change in load of the rate and position springs.

The temperature servo, shown schematically, rotates four separate three-dimensional cams and translates one three-dimensional cam. These cams serve each of the three primary control subsystems. In addition the servo positions a two-dimensional cam used to provide a temperature bias to the slew rate of the duct burner scheduling servo.

3. Windmill Bypass Valve

The windmill bypass valve is positively positioned by a cam operated by an independent shutoff lever. The windmill valve serves as an arming and signaling valve to all shutoff valve and dump valve functions.

In the off position, the windmill valve routes high pressure directly to the back side of the gas generator shutoff valve, and to both zone manifold shutoff valves through the zone valve controller. The gas generator dump valve will receive a high pressure and the gas generator throttle valve regulating valve will receive a low-pressure signal calling for the bypassing or dumping of flow. The throttle valve regulating valve will be reset to call for a slightly low differential pressure and will open to recirculate fuel flow at a low pressure rise.

As the windmill valve is moved from the off to the on position, the pressure signals will be as shown in the schematic. Low pressure will be routed to the manifold dump valve and the throttle valve pressure regulating valve will be reset to modulate at the normal value of differential pressure. Gas generator interstage pressure will be routed to the back side of the gas generator minimum pressure valve, and a drain pressure signal will arm the duct valve controller and allow opening of the duct shutoff valves as other conditions permit.

4. Power Lever Mounted Cams

The power lever is shown schematically as having three cams. Only the two cams which apply to the gas generator system are discussed in this section. The cam functions which apply to the duct heater system are discussed later.

5. Gas Generator Speed System

The power lever shaft rotates a three-dimensional cam which is translated by T_{t2} to produce a portion of the gas generator speed set signal. This portion of the signal positions one end of an adding bar. The power lever shaft also rotates a two-dimensional cam which positions a set of trim rollers. The trim rollers are moved from an idle speed trim platform, over an intermediate speed trim platform, to a maximum speed trim platform. The independently adjustable trim platforms control the vertical height of the trim rollers. The vertical position of the rollers is used to position the opposite end of the adding bar described above. The gas generator desired speed signal is taken from a third point on the adding bar and is therefore a function of the basic schedule described by the three-dimensional cam and the setting of the trim platform in the speed regime desired.

The desired speed signal is compared by a linkage system to an actual speed signal generated by the gas generator speed servo. The error between the desired speed and actual speed positions the primary combustor ratio rollers and thereby adjusts the gas generator fuel flow until there is no speed error.

6. Forward-Reverse Signal Cam

The power lever shaft also operates, through a rotating cam, a two-position pilot valve, which is a forward-reverse signal switch. This pilot valve supplies a signal to the reverser-suppressor control valve. The reverser-suppressor control valve is discussed later.

7. Power Lever Boost System

The power lever is provided with a boost system to maintain input torque at a minimum value. The system consists of a vane-type control valve and a servo operated drive. As the power lever is moved, the control valve allows high pressure to be directed to one end of a servo and rack which is then driven in the direction of pressure unbalance. As the servo moves, it in turn moves the power lever through the rack. As an integral part of the lever assembly, the sleeve in which the control valve rotates is also repositioned until it establishes a null condition of the control valve so that pressures at both ends of the servo and rack are equal and the system is in balance at a new power lever position. In the event of boost system failure, the power lever will be driven by a pin which provides positive engagement between the valve and power lever after a predetermined amount of relative motion has occurred.

8. Gas Generator Speed Sensor and Servo System

The gas generator speed sensor is a flyweight-operated double-acting pilot valve being directly positioned by the centrifugal force. As the valve is displaced from the null position it actuates, through pressure-unbalance, a speed servo piston. As the servo moves, it positions a feedback roller through a feedback cam-multiplying lever and spring system to restore a force balance on the pilot valve. This multiplying lever and spring system is fuel and ambient temperature compensated in exactly the same manner as the T_{t2} system.

9. Speed Servo Mounted Cams

The cam surfaces which are laterally displaced by the gas generator speed servo include four three-dimensional cams. All four cams are rotated by T_{t2}. One cam contains surfaces describing gas generator acceleration limiting schedule. The second cam contains surfaces describing compressor inlet guide vane and actual speed schedules. The third cam contains surfaces describing the duct heater speed permission signal, the reverser-suppressor permission signal, and the compressor bleed schedule. The fourth cam contains the duct heater pressure ratio desired schedule. The duct heater permission signal routes regulated servo pressure to the duct heater operation when requested by power lever angle. The reverser-suppressor permission signal allows servo supply pressure to be routed to the reverser-suppressor control valve when gas generator speed is 90% or less and reverse mode operation is requested by power lever angle.

10. Fuel Flow Computation

The gas generator speed and flow limit control for the JTF17 engine is based upon the W_f/P, or ratio unit computer system. Ratio unit output of the control is proportional to the physical position of the ratio rollers. The ratio rollers are spring loaded toward high ratio values and positioned by a two-dimensional ratio cam toward low ratio conditions. The ratio cam is positioned by an adding bar that compares the desired speed signal generated by the speed set cam to the

actual speed signal generated by a cam on the speed servo. The ratio cam is positioned as a function of the error that exists between the actual speed and the desired speed. An acceleration limiting function will provide a maximum ratio unit stop for the ratio cam for every gas generator speed and inlet temperature condition.

11. P_b Sensing and Multiplying

Primary combustor pressure (P_b) is sensed by a pair of matched bellows. The outside of one of the bellows and the inside of the other is exposed to the environmental ambient pressure. The inside of one of the bellows is completely evacuated to provide a zero absolute pressure reference. The outside of the second bellows is surrounded by P_b pressure. Therefore, the paired bellows generate a force that is proportional to the absolute pressure of the compressor discharge flow. This force is transmitted to the ratio rollers by a multiplying lever. The ratio rollers in turn rest on a second multiplying lever that positions the throttle valve pilot valve and is maintained in force balance by a feedback roller positioned by the throttle valve servo. The force generated by the P_b multiplying lever, acting through the W_f/P_b rollers, is proportional to fuel flow ($Pb \bullet W_f/P_b = Wf$). As the P_b pressure is changed, or as the position of the ratio rollers is varied, the force transmitted to the throttle valve pilot valve multiplying lever is changed and the force balance of the lever is disturbed.

12. Throttle Valve Pilot Valve and Servo

As the force balance of the throttle valve pilot valve multiplying lever is disturbed, the lever will move an amount proportional to the fuel flow error. This error forces the throttle valve from the null position and changes the modulated pressure at the throttle valve servo. The change in modulated pressure causes the throttle valve to move in a direction to restore the force balance on the pilot valve multiplying lever by changing the lever ratio through which the burner pressure force input acts. As the force balance is restored, the pilot valve returns to null and the system is back in equilibrium with the throttle valve at a new flow scheduling position.

13. Duct Heater Blowout Reset Switch

If an excessive pressure ratio is encountered, the blowout reset valve will reset the gas generator ratio roller position, resulting in the fuel flow being reduced to a low value regardless of the other system parameters. The blowout reset switch will automatically withdraw when the engine is again operating within normal limits.

14. Fuel Inlet Filter

The control contains a coarse filter located in the fuel inlet line. The flow pattern over the main filter element is from outside to inside. Incorporated in the filter body is a fine mesh servo-supply screen through which a portion of the main filter flow passes from the inside of the filter to the control servo-supply system. The servo filter screen is of the wash type; as the main flow passes through the filter it tends to wash away the dirt accumulated on the

servo filter screen. To improve the washing action of the servo-screen the velocity of the main flow is increased as it passes the wash filter. A check valve type bypass is provided around the servo flow filter.

15. Throttle Valve Pressure Regulating Valve

The function of the throttle valve pressure regulating valve is to maintain a constant pressure difference across the throttle valve metering window regardless of the size of the window opening. The system chosen for the gas generator throttle valve is a direct acting valve with flow compensating buckets plus a rotating pressure sensor pilot valve with an integral reset piston. It may be seen from the schematic that both the pressure regulating valve (PRV) and the PRV sensor receive a signal equal to throttle valve inlet pressure on one end and throttle valve discharge pressure on the opposite end. The end of the valve exposed to throttle valve discharge pressure also receives a mechanical force input from a preloaded spring. Both the PRV and the PRV sensor seek a position where the force difference created by the inlet and discharge pressures acting on the throttle valve are just equal to the mechanical force of the spring. In the event that the PRV sensor is exposed to a low difference in throttle valve pressures, the sensor is displaced to cause an increase in the modulated pressure to the integral reset piston. The reset piston will then be displaced to cause an increase in the mechanical spring load transmitted to the PRV and the PRV will in turn move to increase the throttle valve inlet pressure. The system displays the accuracy and force level advantages of a sensor-equipped PRV system. In the event of a sensor failure, the regulating valve is capable of operating without the sensor with only a slight shift in the regulated pressure valve. This fail-safe condition is assured by limiting the stroke of the integral reset piston to maintain a percentage of normal mechanical spring force regardless of the sensor-modulated pressure.

16. Throttle Valve

The gas generator throttle valve is a servo-operated throttle valve. The servo displacement is controlled by a throttle pilot valve. The throttle valve metering window area is increased as a linear function of throttle valve servo displacement. Because of the constant pressure difference maintained across the throttle valve window by the regulating valve, fuel flow through the valve is directly proportional to the window for all flow conditions.

17. Minimum Pressure Valve

The fuel flow from the gas generator throttle valve passes through a minimum pressure valve before leaving the fuel control. The minimum pressure valve consists of a piston that must be lifted from its seal to expose the outlet port. The pressure required to open the valve is controlled by a spring on the back side of the piston and gas generator interstage pressure. The function of the valve is to prevent fuel from passing through the control until a sufficiently high internal pressure level has been reached to assure proper and accurate positioning of the control servos. The minimum pressure valve also serves as

a positive shutoff valve. With introduction of high pressure into the area behind the piston, the spring force and fuel pressure differential pressure will close the valve.

18. Servo Pressure Regulator

A portion of the servo supply pressure drawn from the main filter is regulated to a constant level above the control drain pressure. The valve is a sensor-operated regulator which bleeds sufficient pressure into the regulated servo supply system to maintain a preselected pressure. The valve sensor compares regulated pressure to control case pressure plus a preset spring force. If the regulated pressure falls too low, the sensor reduces the modulated pressure acting on the regulating valve and the regulating valve moves to bleed more pressure into the regulated supply system.

19. Case Pressure Regulator

The case pressure regulating valve is a sensor-operated regulating valve very similar to the servo supply regulator. Case pressure is compared to an absolute pressure source plus a preset spring load across the regulating valve sensor. If too high a case pressure is sensed, the modulated pressure behind the regulating valve piston will be reduced by the sensor, and the regulating valve will move to bleed case pressure into the pump inlet area.

20. Thermal Bypass Valve

The thermal bypass valve provides for bypassing of inlet flow to the gas generator fuel control when the temperature reaches a predetermined value. The valve will control to two temperature levels as a function of flow. The system consists of a rotating pilot valve, a temperature sensitive valve, a variable orifice, and a bypass valve. The pilot valve is driven to the bypass position by regulated pressure which is modulated by the temperature valve, the variable orifice and pilot valve. The temperature sensitive valve is located in the regulated pressure line. As temperature increases, the valve moves to increase the pressure for driving the pilot valve against a spring load to the bypass position. This pressure is further trimmed by the variable orifice which is moved by the throttle valve feedback linkage as a function of flow. At the required flow and temperature condition, pressure to the pilot valve will be high enough to drive it to the point where the pilot valve itself will increase pressure causing it to be driven completely to the bypass position. At this point, high pressure is directed to the thermal bypass valve moving it to the bypass position.

B. DUCT EXHAUST NOZZLE CONTROL

The duct nozzle area is set to a nominal schedule as a function of power lever angle biased by T_{t2} and this setting trimmed by a duct heater airflow parameter to maintain the requested airflow.

1. Pressure Ratio Sensor and Servo

The primary parameter sensed in trimming the duct exhaust nozzle area is the pressure ratio in the duct, $(P_{T3}-P_{s3})/P_{t3}$. The fan pressure ratio sensor is a force balance device. A force proportional to the dynamic pressure $(P_{T3}-P_{S3})$ of the air downstream of the fan is compared to a force proportional to the total pressure (P_{t3}) of the air stream. A force balance linkage divides the dynamic and total pressure forces trigonometrically to establish a force ratio proportional to the pressure ratio. As the pressure ratio varies, force unbalance in the linkage system occurs which causes linkage motion away from steady-state null conditions. This linkage motion is applied to a rotating four-way pilot valve through a lever system. The hydraulic output pressure from the pilot valve actuates a servo piston and feedback linkage system which is capable of restoring the force balance of the linkage system. When pressure ratio varies, the pilot valve will cause the servo and feedback linkage to readjust the force balance until a force equilibrium condition is attained. When the sensor is nulled the servo and feedback linkage output motion is a direct measure of pressure ratio. The position of the servo is a unique pressure ratio.

2. Desired Pressure Ratio Signal

A desired pressure ratio signal is generated by a cam and linkage system as a function of speed (N_2) and temperature (T_{t2}). The cam followers are moved from one surface to the other in anticipation of duct burning initiation and power lever angle step changes. After the transient is accomplished the cam followers are returned to the run portion of the schedule cams. The switching from the run cam to the reset cam is accomplished by the pressure ratio reset piston and causes an increase in the exhaust area and a resultant increase in the duct pressure ratio. The pressure ratio reset piston incorporates a remote adjustment device so that while operating on the run portion of the cam the duct pressure ratio may be adjusted to achieve optimum airflow operating conditions.

3. Area Control Valve

The actual pressure ratio in the duct is compared to the desired pressure ratio by a summation link. A portion of any error observed is applied directly to the area scheduling linkage to effect an area trim. In addition, a spool valve positioned by pressure ratio error meters flow to an integrating piston which adds its output to the area system. The proportional signal serves to achieve fast response and the integral signal serves to eliminate all errors introduced downstream of the integrating piston. The area scheduling linkage displaces, via a flapper-operated servo, the area control valve. The area control valve modulates pressure to the exhaust nozzle actuators, and mechanical feedback from the actuators is applied to the area control linkage to reposition the area control valve when the desired area is achieved.

4. Maximum Area Stop

The purpose of this stop is to maintain a fixed area, smaller than that called for by the pressure ratio sensor system, during low engine speed operation. During low engine rotor speed operation the nominal airflow through the engine is reduced and the quality of the pressure ratio signals is poor. Under these operating conditions, the maximum area stop is engaged to maintain a preset maximum nozzle area. The stop will remain until the duct burner scheduling servo reaches a predetermined value when a pressure signal will cause the maximum area stop to withdraw.

5. Area Control Fail-Safe Valve

In the case of a failure of the mechanical area feedback mechanism, a spring on the area feedback cam will position the cam to indicate a maximum exhaust area. This maximum area signal will cause the area failsafe valve to activate the fan blowout valve and shut off the zone burner fuel flows.

6. Duct Heater Blowout Signal

If loss of the flame front is encountered in the duct heater, it will be sensed by the pressure ratio sensor as a large increase in pressure ratio. This increase in pressure ratio will cause a large enough error to pressurize a failsafe hydraulic circuit. As the fail-safe circuit is pressurized, it causes the duct heater blowout valve to shut off both Zone I and Zone II duct burning, and it also caused the blowout reset piston to reduce the fuel flow in the gas generator until the large pressure ratio error is corrected. As the pressure ratio error diminishes, the gas generator is restored to normal operation; however, duct burning cannot be resumed until the power level is retarded from the duct heating regime and advanced to the Zone I ignition position.

7. Electronic Airflow Control

The prototype version of the unitized control will incorporate an interface which permits the hydromechanical duct airflow control to be removed and replaced with an electronic control. The electronic control will incorporate the additional function of automatic engine pressure ratio control. The communication links required between the hydromechanical and electronic sections will be within this interface.

C. DUCT HEATER FUEL CONTROL

1. Duct Schedule Pilot Valve

The same power lever used to control gas generator operation is also used to control the level of duct heating. The power lever contains a cam which, above a predetermined position, will arm the duct schedule pilot valve and then control its position. During duct heater operation, servo supply pressure is ported to the duct heater blowout

valve to be available as a pressure lock in the event the valve receives a signal to prevent duct heater operation. Such a signal could result from a loss in area feedback or from duct heater blowout and would be received in the form of servo supply pressure ported to the back side of the blowout valve by either the failsafe valve or the duct heater pressure ratio sensor through the proportional valve. The power lever cam positions the duct schedule pilot valve through a series of levers when the blowout valve is withdrawn. However, the movement of the linkage connecting the cam to the valve is restricted by the zone ignition indicator valves. These valves limit the maximum duct flow schedule until ignition has been established in each of the two zones of duct burning.

2. Duct Schedule Servo

The duct heater scheduling servo is positioned by the duct scheduling pilot valve and positions a feedback lever to null the pilot valve when the servo has reached its desired position. The rate of scheduling servo motion is controlled by a variable orifice that is positioned as a function of T_{t2}. The scheduling servo laterally positions a three-dimensional cam that contains the duct burner flow schedule and A_j schedule.

3. Duct Scheduling W_f/P_b System

As in the gas generator W_f/P_b system, compressor discharge pressure is sensed by a pair of matched bellows. The configuration of these bellows is identical to that of the gas generator system. The force from the matched bellows, which is proportional to absolute pressure of compressor discharge flow, is transmitted to the ratio rollers by a multiplying lever. The ratio rollers in turn rest on a second multiplying lever that positions the duct throttle valve pilot valve. Because of the nonlinearity of the duct throttle valve, feedback to the pilot valve is accomplished by means of a nonlinear linkage incorporating a second set of rollers to restore balance to the multiplying system. The ratio rollers are positioned by the duct scheduling servo as a function of power lever angle to establish desired W_f/P_b. This set of rollers is equipped with provisions for a remotely adjustable maximum ratio stop.

4. Low Speed Protection Valve

The duct scheduling pilot valve is restricted from initiating duct heater burning until the gas generator speed is above a certain value. The low speed protection valve, which is controlled by the gas generator speed servo, cuts off the servo supply pressure source for the duct scheduling pilot valve. Only when the gas generator is above a certain speed can duct heating be scheduled.

The low speed protection valve also prevents the reversers from operating above a predetermined speed by porting drain pressure to the reverser control valve and holding it in the forward thrust position regardless of reverse arming valve position. When engine speed decreases to below this set point, the valve will move to a position

where signal pressure from the arming valve will be ported to the reverser control valve, allowing the clamshells to be moved to the reverse thrust position when requested.

5. Zone I and Zone II Fill, Shut Off, and Dump Valve Sequencing

The duct heater controller valve coordinates the various phases of duct ignition and fuel manifold quick filling. When the gas generator speed servo activates the duct permission switch and the duct pressure ratio is within normal limits, the controller may be positioned by power lever to produce the following signals:

1. The Zone I manifold dump valve will be signaled to close.
2. The Zone I ignition sequencer will energize the duct heater igniters.
3. The pressure ratio reset piston will be signaled to increase the duct nozzle area in anticipation of duct heater initiation.
4. The Zone I fuel shutoff valve will be signaled to open, causing gas generator pump interstage flow to be ported to the Zone I fuel manifold for quick filling.

At the end of quick fill, the circulating flow valve will be signaled to close by the Zone I fuel manifold fill sensor. The duct heater fuel scheduling servo will be held at lightoff fuel flow ratios. The Zone I fuel manifold fill sensor signals the duct burner ignition switch to begin moving such that 0.25 second after the end of quick fill duct heater ignition is stopped. With the end of duct heater ignition, the duct heater scheduling servo is released from lightoff fuel flow ratios, and the power lever may position the servo to select higher ratios.

As the power lever angle is advanced through the transfer from Zone I, to Zone I plus Zone II, the following signals are produced:

1. The Zone II manifold dump valve is signaled to close.
2. The Zone II fuel shutoff valve is signaled to open, causing gas generator pump interstage flow to be ported to the Zone II fuel manifold for quick filling.
3. The duct heater fuel scheduling servo is limited to transfer fuel flow ratios.

At the end of quick-fill, the Zone II manifold fill sensor will release the hold on the duct burner scheduling servo transfer fuel flow ratios and allow the power lever to position the servo to select higher fuel air ratios up to maximum duct heat.

6. Duct Fuel Flow Computation

The duct fuel flow computation is accomplished through a roller position and multiplying lever system similar to that used in the gas generator system.

7. Fuel Flow Ratio Roller

The computation system consists of a set of rollers, positioned by the cam on the duct scheduling servo, whose position is a function

of the desired ratio of W_f/P_b. This set of rollers operates between two levers, one lever transmits a force proportional to gas generator burner pressure, from a bellows assembly as described in the gas generator section. The other lever positions the duct heater throttle valve pilot valve and receives a feedback force, through a set of rollers attached to the throttle valve, proportional to throttle valve position.

8. Filter

The inlet to the duct burner fuel flow system is protected by a coarse screen. After the flow has passed through the coarse screen, a portion of the flow is drawn off through a fine filter for use as a computing fluid. The fine filter is an integral part of the main filter body. A check valve type bypass is provided around the fine filter.

9. Throttle Valve

The duct heater throttle valve and servo are very similar to the gas generator control throttle valve except for physical size. The duct heater valve is larger to accommodate the higher flow requirements. The throttle valve servo is positioned by the throttle valve pilot valve, and feedback is through a set of rollers attached to the throttle valve.

10. Regulating Valve

The zone throttle valve regulating valve is a flow-compensated servo-operated valve, with the pressure sensor pilot valve working through a reset piston in the same manner as the gas generator regulating valve, and with the same reliability safeguards. The major difference between the two valves is that the duct regulator controls the throttle valve pressure drop by changing the throttle valve discharge pressure with a variable restrictor in the discharge line, while the gas generator regulator controls the throttle valve pressure drop by controlling the throttle valve inlet pressure with a bypass for excess pump flow. The difference in the two systems is allowed through the use of a centrifugal pump to supply the duct system, while the gas generator system is required to use a positive displacement type of pump.

11. Zone I Shutoff Valve

The Zone I shutoff valve serves to prevent flow to the Zone I manifold until certain desired conditions have been met. In the closed position the valve is maintained in a pressure balance condition by the routing of fine filtered flow from the duct inlet filter to the back side of the valve position. In this condition the valve is positively held against its seat by a mechanical spring force. When the valve receives a signal to open, the valve piston is exposed to drain pressure, and when the outlet pressure is sufficient, the shutoff valve will open against its spring load and allow flow to pass to the Zone I manifold.

12. Zone II Shutoff Valve

The Zone II shutoff valve works in the same manner as the Zone I shutoff valve.

13. Zone II Quick-Fill Valve

The quick-fill system draws fuel from the gas generator pump interstage to fill the Zone II manifold. When the fuel manifold is full, as sensed by the Zone II manifold fill sensor, the quick-fill valve will shuttle porting duct heater metered flow to Zone II manifold. The fuel flow split between Zone I and Zone II is accomplished by the pressure flow characteristics of the flow nozzle.

14. Circulating Flow Valve

The circulating flow valve is controlled by the fill, shutoff, and dump valve controller through the Zone I manifold fill sensor. Sequencing of the valve is such that it is open for all duct heater off conditions. When the fill sensor indicates Zone I fuel manifold is full, the fill sensor will shuttle, allowing the circulation flow valve to close.

15. Duct Heater Secondary Cooling Flow Valve

During duct heater operation, duct heater flow is required for cooling such that total control inlet flow is 3000 pph minimum. This is accomplished by the secondary cooling flow valve which is attached to the throttle valve feedback lever. When metered flow is 800 pph the valve passes 2200 pph from control inlet to gas generator interstage. As metered flow increases, cooling flow decreases until such time that metered flow is 3000 pph and cooling flow is reduced to zero.

16. Manifold Fill Sensors

The manifold sensors consist of a piston which senses manifold pressure on one side and duct static pressure on the other side, and a spring loaded pilot valve. When operating in the nonaugmented regime the spring loads the sensor piston in the zone off position. As fuel is supplied to the manifold, fuel pressure will overcome the spring force and move the pilot valve to the zone on position porting pressure to the proper systems.

17. Duct Burner Ignition Switch

The duct burner ignition switch, which is positioned by signals from Zone I manifold sensor valve, is used to move an ignition indicator valve in addition to energizing the duct heater electrical ignition system. The servo sequences the unlatching of the Zone I ignition indicator until burning has had sufficient time to be established. This function prevents the too rapid addition of fuel to the duct burners at the initiation of burning. An orifice in the signal line is used to control travel rate of the ignition valve. When the ignition timer servo has traveled its full stroke, it opens

a hydraulic port which signals the indicator valve to withdraw. Withdrawing of the Zone I indicator valve permits the duct scheduling servo to advance to the Zone II burner ignition point upon demand, and withdrawing of the Zone II ignition allows the scheduling servo to advance to maximum power position.

18. Zone Indicator Valves

As previously described, the zone indicator valves sequence the movement of the scheduling servo until after Zone I ignition hold is released, or in the case of Zone II, until the fuel manifold has been quick filled. The Zone I indicator valve also sends a signal to the reset, indicating duct heat initiation.

19. Turbopump Controller

The centrifugal pump used to supply high pressure fuel to the duct burner metering system is driven from a turbine powered by high pressure air bled from the gas generator compressor. The airbleed is controlled by a butterfly valve in the bleed duct. The butterfly valve is positioned by an integrating servo which received hydraulic signals from the pump controller mounted in the fuel control package.

The pump controller consists of a split G-load compensated, pilot valve and a proportional servo piston. The valve is constructed to compare zone throttle valve inlet pressure to regulating valve outlet pressure. If this pressure differential is too high the pump flow will be reduced to reduce the engine power absorbed by the pump. The split pilot valve consists of two double-acting pilot valves; one valve routes an immediate input to the remote butterfly valve integrating piston, while the second valve routes flow to the proportional piston. As the proportional piston moves, it repositions the split pilot valve to null position by means of a feedback spring, and opens a pilot valve which is directly positioned by the servo. The servo-positioned valve remains open and combines its flow with that from the split valve until the split valve is displaced to reposition the proportional piston and servo positioned valve to their null position. The concept involved in the controller is called a derivative plus proportional integrating system and allows the incorporation of two system time constants for fast response and stability.

D. AUXILIARY CONTROL FUNCTIONS

1. Compressor Inlet Guide Vane Control

The compressor inlet guide vane control is a two position pilot valve positioned by N_2 and T_{t2}. During an engine start, the vanes will be in the cruise position and will move to the take-off position at the proper N_2 as engine speed is increased. The pilot valve will be moved from the takeoff position to cruise as T_{t2} is increased. Operation of the inlet guide vanes for the aero-brake function is presented elsewhere in the proposal.

2. Reverser-Suppressor Control

The reverser-suppressor control is a two position pilot valve which requests the reverse or forward position of the reverser-suppressor as a function of power lever angle. The reverse positions cannot be requested when N_2 is equal to or greater than a predetermined speed.

3. Compressor Bleed Control Valve

The compressor bleed control valve is a remotely mounted pneumatic unit that signals the compressor bleed to open or close as a function of T_{t2} and N_2. A two-position pilot valve in the control provides a hydraulic signal to actuate the remote control unit.

4. Secondary Air Control

Provisions are included in the control for signaling a secondary air control. This signal may be used to position a valve or series of valves which would regulate the aircraft nacelle secondary air.

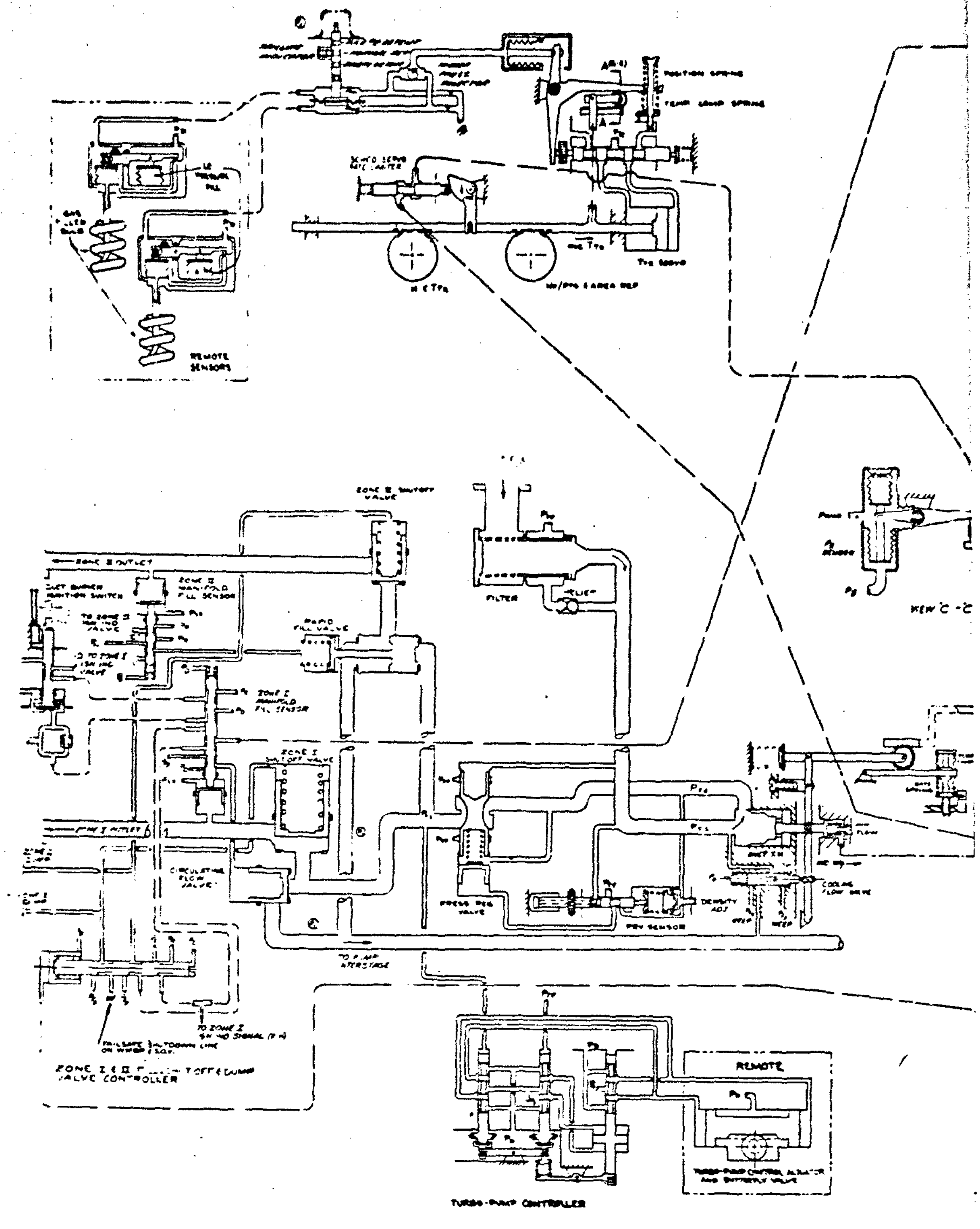
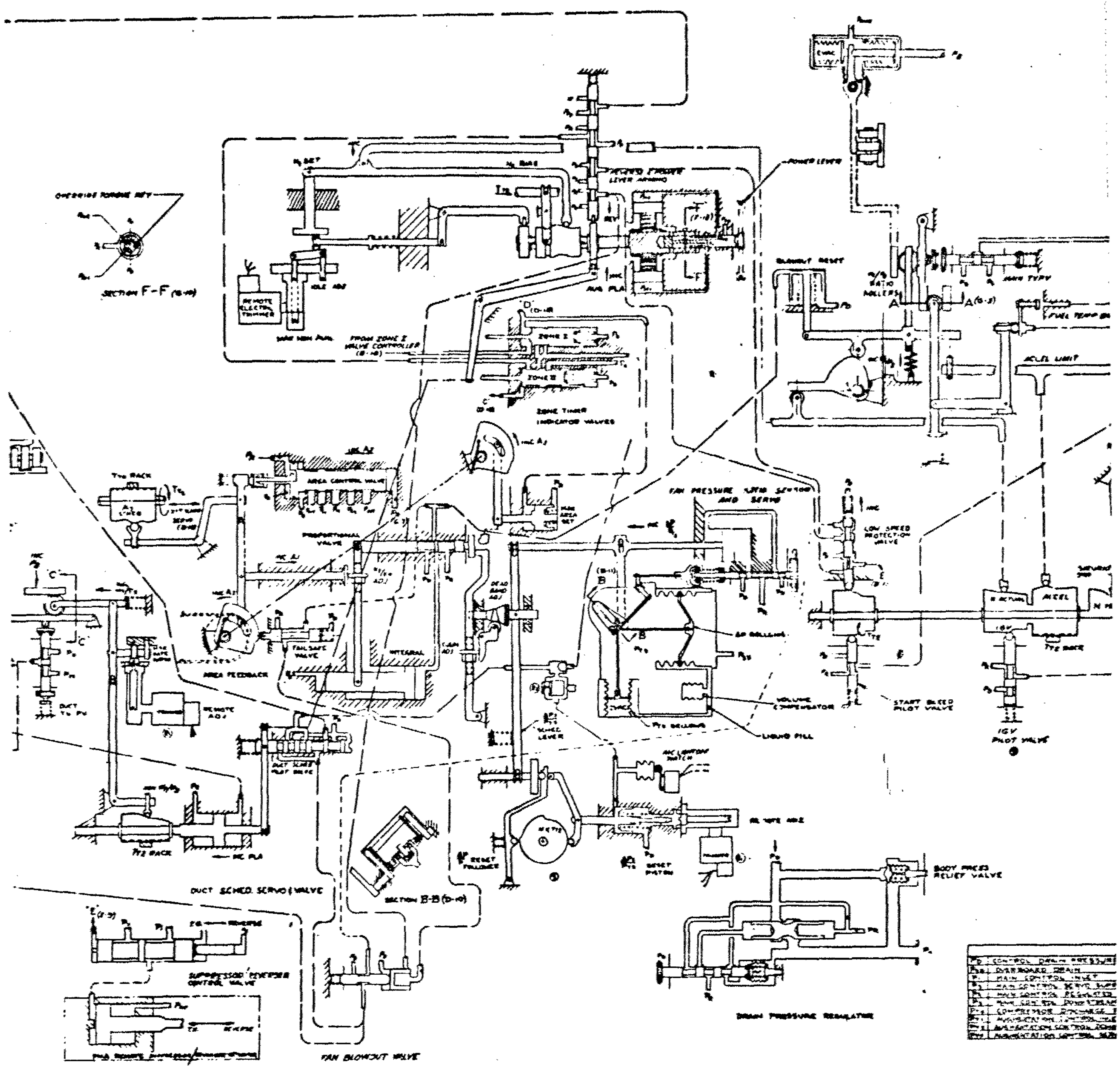
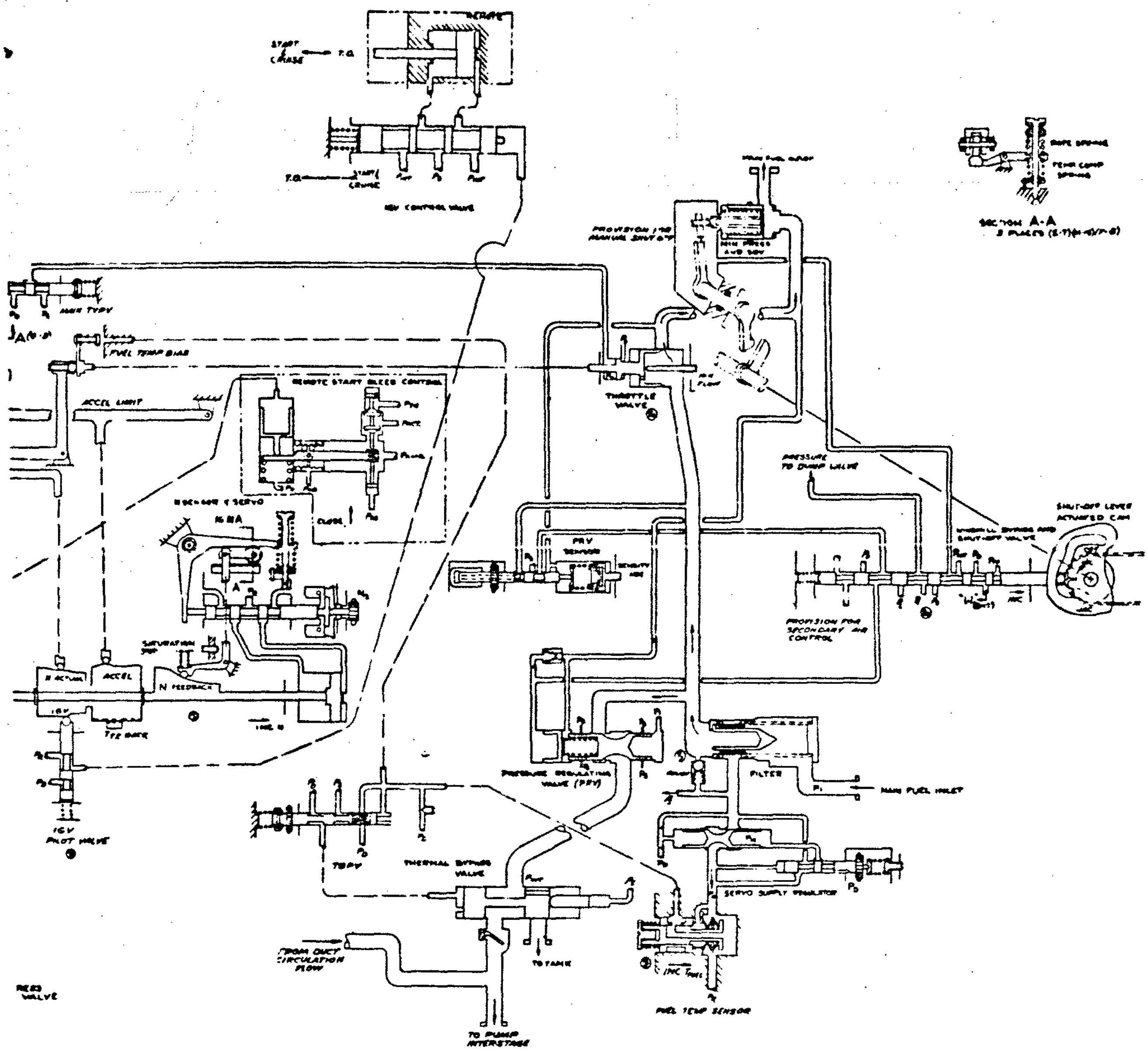


Figure 2. Unitized Fuel and Area Control Schematic, Hamilton Standard


SECTION F-F (B-10)
AREA FEEDBACK
AREA CONTROL VALVE
PROPORTIONAL VALVE
FAILSAFE VALVE
INTEGRAL
ZONE TIMER INDICATOR VALVES
FAN PRESSURE RATIO SENSOR AND SERVO
LOW SPEED PROTECTION VALVE
START BLEED PILOT VALVE
IGV PILOT VALVE
VOLUME COMPENSATOR
LIQUID FILL
DUCT SCHED. SERVO & VALVE
SECTION B-B (D-10)
SUPPRESSED REVERSER CONTROL VALVE
FAN BLOWOUT VALVE
BODY PRESS RELIEF VALVE
DRAIN PRESSURE REGULATOR
BLOWOUT RESET
POWER LEVER
ACCEL LIMIT
FUEL TEMP

2



CODE			
[illegible]	CONTROL DRAIN PRESSURE	[illegible]	MODULATED SERVO PRESSURE
[illegible]	OVERBOARD DRAIN	W_F	MAIN FUEL FLOW
P_1	MAIN CONTROL INLET	[illegible]	E.N.C. HYDRAULIC PUMP OUTLET
[illegible]	MAIN CONTROL SERVO SUPPLY (FINE FILTER)	[illegible]	E.N.C. MODULATED SERVO PRESSURE
[illegible]	MAIN CONTROL REGULATED SERVO SUPPLY	[illegible]	E.N.C. DRAIN PRESSURE
[illegible]	MAIN CONTROL DOWNSTREAM THROTTLE VALVE	[illegible]	FAN DISCHARGE STATIC PRESSURE
[illegible]	COMPRESSOR DISCHARGE PRESSURE	[illegible]	FAN DISCHARGE TOTAL PRESSURE
[illegible]	AUGMENTATION CONTROL INLET PRESSURE	[illegible]	COMPRESSOR INLET TEMPERATURE
[illegible]	AUGMENTATION CONTROL DOWNSTREAM THROTTLE VALVE	[illegible]	PUMP INLET PRESSURE
[illegible]	AUGMENTATION CONTROL SERVO SUPPLY (FINE FILTER)	[illegible]	PUMP INTERSTAGE

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BIII

3

APPENDIX B

UNITIZED FUEL AND AREA CONTROL (BENDIX)

A. GAS GENERATOR FUEL CONTROL

The engine parameters used in performing the wc-- functions of the fuel control are rotor speed (N_2), engine inlet temperature (T_{t2}), and primary combustor pressure (P_b). A schematic drawing of drawing of the proposed control is shown in figure 1.

1. Engine Rotor Speed Actuation System

The rotor speed sensor and actuator is presented in figure 1 at the bottom to left of center.

Speed is sensed as the force output of a set of centrifugal weights and is applied at a fixed lever arm to a moment balance servo system. This force is balanced at null by a spring operating at constant height acting through a roller positioned by a feedback cam. Force or moment error is detected by a rotating servo slide valve which positions a piston to rotate the speed cam shaft. A feedback cam is contoured to position the roller to balance the "speed squared" weight force with shaft rotation proportional to this force. The speed cam shaft, in addition to the feedback cam, drives the acceleration cam, the governor cam, the pressure ratio scheduling cam, the compressor bleed schedule cam, the inlet guide vane schedule cam and the duct arming cam.

2. Engine Inlet Temperature Sensors and Actuator

The T_{t2} sensor and actuation system is shown in figure 1 in the lower left corner.

Two gas filled tube sensors are used to sense compressor inlet temperature. A helium filled helical tube, which is located in the air stream, exerts pressure on a diaphragm in the sensor proportional to temperature in accordance with the gas laws. This gas pressure is balanced by fuel pressure differential across a bellows. A bleed servo valve system, operated by the balance beam position, regulates the fuel pressure differential as required to maintain the moment balance on the beam. Since two sensors are used, a signal selector valve located in the control selects the pressure signal which represents the highest temperature indication and delivers this pressure to the temperature receiver bellows. This pressure acts through a linear pressure-to-travel servo actuator to translate a group of speed cams and through levers to translate the $\Delta P/P$ trim, speed request, plateau and duct schedule cams. A feedback lever closes the servo actuator loop.

Since two sensors are used with a selector to deliver the proper signal in the event of failure of one probe, the failure would go unnoticed. An indicator is included to show by inspection that one of the sensors is out. A pair of opposing bellows will displace the indicator lever if the signals differ by more than the allowable tolerance. The indicator is spring loaded against a tab and when the allowable error is exceeded, slips past the tab thus locking in the failed error position.

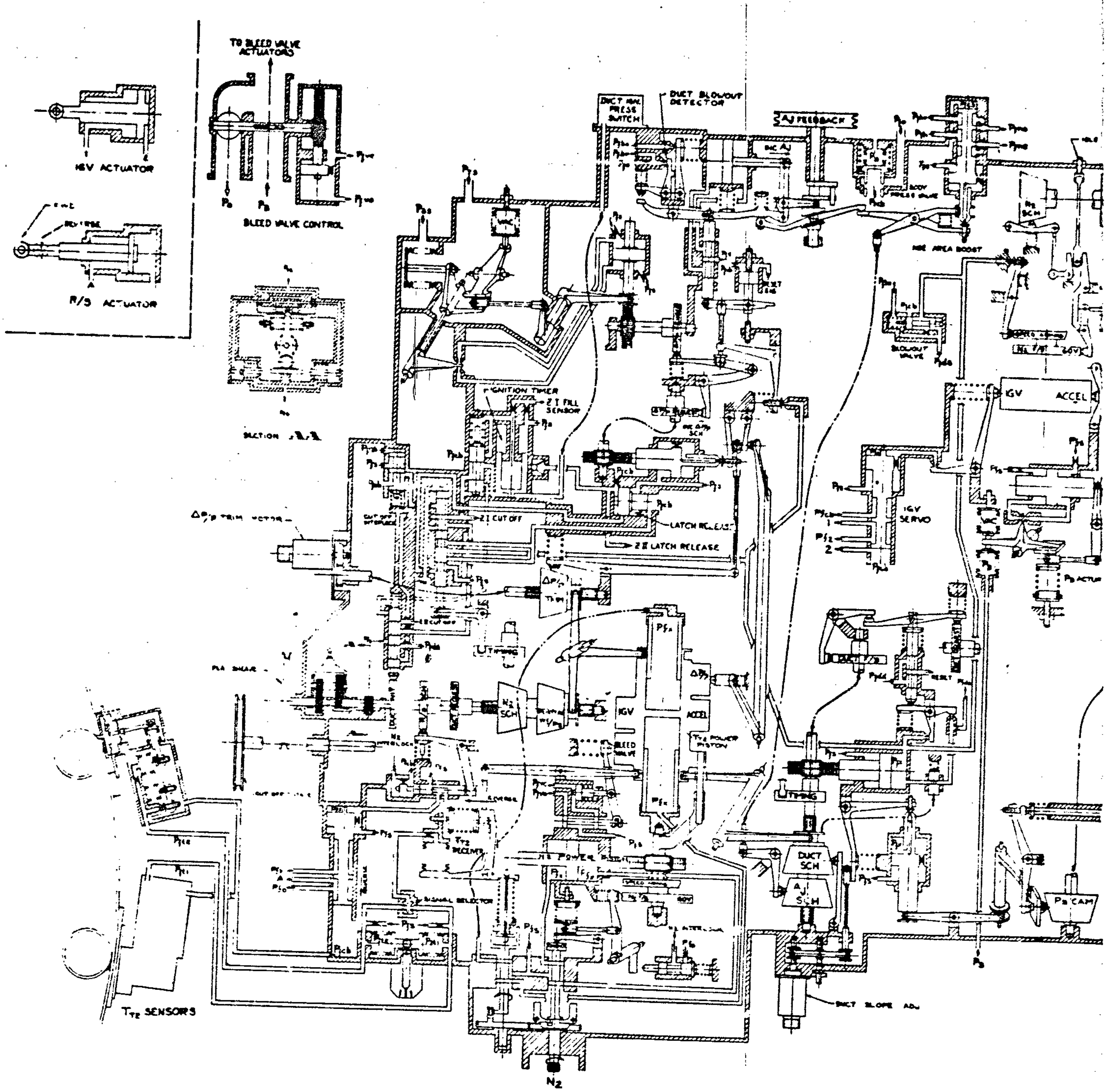
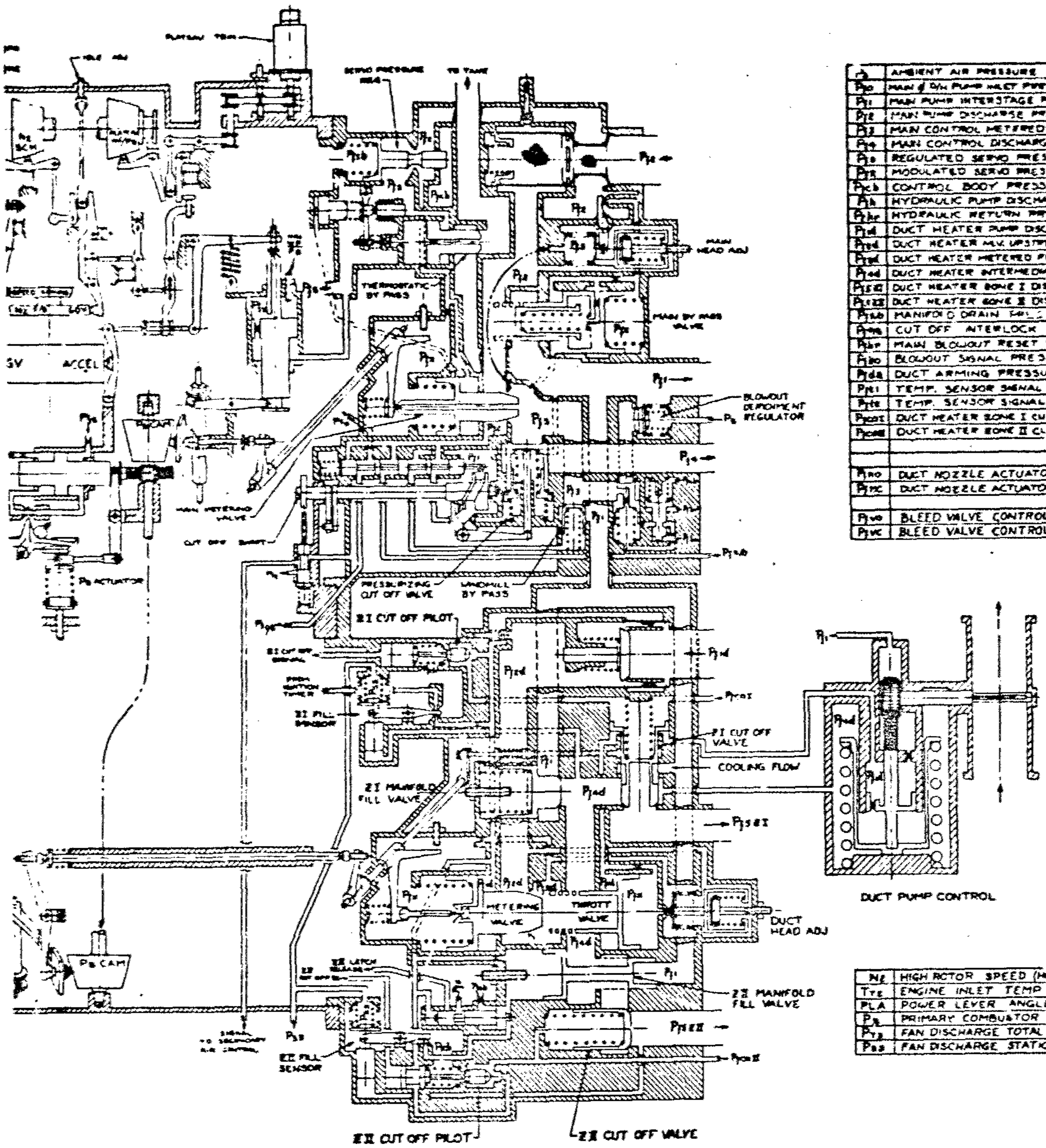


Figure 1. Unitized Fuel and Area Control Schematic, Bendix Products Aerospace Division



P_a	AMBIENT AIR PRESSURE
P_{fo}	MAIN & D/H PUMP INLET PRESSURE
P_{f1}	MAIN PUMP INTERSTAGE PRESSURE
P_{f2}	MAIN PUMP DISCHARGE PRESSURE
P_{f3}	MAIN CONTROL METERED PRESSURE
P_{f4}	MAIN CONTROL DISCHARGE PRESSURE
P_o	REGULATED SERVO PRESSURE
P_m	MODULATED SERVO PRESSURE
P_{cb}	CONTROL BODY PRESSURE
P_h	HYDRAULIC PUMP DISCHARGE PRESSURE
P_{hr}	HYDRAULIC RETURN PRESSURE
P_{f1d}	DUCT HEATER PUMP DISCHARGE PRESSURE
P_{f2d}	DUCT HEATER MV UPSTREAM PRESSURE
P_{f3d}	DUCT HEATER METERED FUEL PRESSURE
P_{fid}	DUCT HEATER INTERMEDIATE FUEL PRESSURE
P_{f5I}	DUCT HEATER ZONE I DISCHARGE PRESSURE
P_{f5II}	DUCT HEATER ZONE II DISCHARGE PRESSURE
P_{f5b}	MANIFOLD DRAIN PRESSURE
P_{fis}	CUT OFF INTERLOCK SIGNAL PRESSURE
P_{br}	MAIN BLOWOUT RESET PRESSURE
P_{bo}	BLOWOUT SIGNAL PRESSURE
P_{da}	DUCT ARMING PRESSURE
P_{te1}	TEMP. SENSOR SIGNAL PRESSURE
P_{te}	TEMP. SENSOR SIGNAL PRESSURE
P_{coI}	DUCT HEATER ZONE I CUT OFF PRESSURE
P_{coII}	DUCT HEATER ZONE II CUT OFF PRESSURE
P_{no}	DUCT NOZZLE ACTUATOR OPEN PRESSURE
P_{nc}	DUCT NOZZLE ACTUATOR CLOSED PRESSURE
P_{vo}	BLEED VALVE CONTROL OPEN PRESSURE
P_{vc}	BLEED VALVE CONTROL CLOSED PRESSURE

N_2	HIGH ROTOR SPEED (HIGH COMPRESSOR)
T_{T2}	ENGINE INLET TEMP
PLA	POWER LEVER ANGLE
P_B	PRIMARY COMBUSTOR BURNER PRESSURE
P_{T3}	FAN DISCHARGE TOTAL AIR PRESSURE
P_{S3}	FAN DISCHARGE STATIC AIR PRESSURE

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3. Primary Combustor Pressure Actuator

The primary combustor pressure, P_b, actuation system is shown in figure 1 in the center.

Primary combustor pressure is sensed by a bellows which exerts a moment on the servo lever. This moment is opposed by a constant height spring acting at a distance from the fulcrum determined by the position of the roller. The flapper displacement due to moment unbalance results in piston displacement to renull the system through roller displacement. Thus, piston displacement is proportional to P_b. The constant of proportionality is variable with load of the reference spring. A cam is rotated by the P_b actuator. The cam is used as the multiplier to establish the main gas generator flow. P_b limiting, if required for the main gas generator flow, may be obtained by slope reversal of the cam contour.

4. Rotor Speed (N_2) Request and Plateau Setting

The power lever sheave and speed and plateau schedule cams are at the left side of figure 1 and are repeated in the upper center portion.

The power lever sheave drives a step-up gear which in turn drives the N_2 schedule cam in rotation. The step-up gear carrier pivots freely except for the restraint of a spring centered servo slide valve to which it is linked. Differential torque displaces this valve to actuate power boost pistons on the output shaft. The schedule cams are 3D cams translated by T_{t2}. The outer radius of the N_2 schedule cam represents the idle setting, and increasing speed requests are represented by cam fall. The plateau cam rises with increasing ratio setting.

5. Acceleration Flow Scheduling System

The acceleration scheduling system is shown on figure 1 near the center.

The acceleration cam is rotated by the speed (N_2) system and translated by the T_{t2} system. The contour provides a schedule of fuel-air ratios (W_f/P_b) as a function of speed and temperature. The cam arrangement provides cam fall with increasing ratios to permit governor "bumping". This cam provides a suitable schedule for engine start and acceleration, and a zero speed ratio level for emergency engine operation in the event of speed shaft failure. The follower link travel is boosted to the main flow multiplier P_b cam. At the multiplier the fuel-air ratios appear as follower displacement axially on the cam which rotates as a function of P_b. A second follower is displaced by the initial follower such that its displacement is the product of W_f/P_b and P_b. This product is therefore proportional to the required fuel flow and is the output of the computer to the fuel metering section. A level adjustment in the computer permits presetting the output level position for purposes of interchangeable mating of components.

6. Speed Governing System

The governing system is shown in figure 1 in the upper center.

This is a bump-type governor system in that the travel of the governor cam follower has no effect on the fuel schedule until its schedule and the acceleration schedule become identical. At this point the governor lever "bumps" the acceleration follower link, lifting the follower from the acceleration cam, and the governor schedule assumes complete scheduling control. Speed schedule is a function of power lever angle biased by T_{t2}.

The governor cam is rotated by speed (N_2). The output at the follower is a continuous fall rate with increasing speed. The governor follower lifts one end of a walking beam and the other end is positioned by the N_2 schedule cam. As illustrated, the governor follower is moved to the left with increasing speed, moving the opposite end to the right. The schedule cam positions the center of the walking beam. Prior to governor take over, the spring on the schedule follower merely loads the acceleration cam follower through the linkage with the schedule follower suspended above its cam. As the governor cam falls with increasing speed, the schedule cam follower also falls toward its cam until it bumps. Continued fall of the governor cam lifts the follower linkage from the acceleration cam, and the ratio schedule is established by the governor cam.

Whe he governor is in control of the acceleration follower link, boost a.. multiplication are identical to that previously described.

The idle speed adjustment is also shown. The outer tracks of the adjustment linkage remain parallel at maximum speed and are wedge shaped when the idle speed is selected and is being adjusted. Thus, this adjustment can adjust the speed level desired by means of wedging action and simultaneously change the linkage ratio so as not to affect a change at the other speed level.

7. Plateau System

The plateau scheduling system is shown in the upper center portion of figure 1.

This is a bump-type system in that the position of the plateau cam follower has no effect on the fuel schedule until the governor schedule and the plateau schedule become identical. At this point a lever in the governor linkage "bumps" the plateau follower, which results in transfer of the schedule to the plateau cam. This schedule is a function of power lever angle and T_{t2} with a remote control motor driven adjustment. When the plateau system is in control of the acceleration follower link, boost and multiplication are identical to that previously described.

The plateau adjustment consists of a mechanism for manual or remote control motor positioning of a roller between nonparallel levers, which effectively changes plateau cam height of the follower lever. It affects all plateau settings by equal ratio values.

B. DUCT EXHAUST NOZZLE CONTROL

The purpose of the duct exhaust nozzle control is to regulate the engine airflow by throttling of the duct airflow. The nozzle area is

basically scheduled by Power Lever Angle biased by T_{t2}. This schedule is subject to trim by the airflow parameter $(P_{T3} - P_{S3})/P_{T3}$ system simply referred to as $\Delta P/P$. The $\Delta P/P$ parameter is in turn scheduled by N_2 and T_{t2}.

1. Duct Pressure Ratio ($\Delta P/P$) Scheduling and Error Computing System

The pressure ratio scheduling system is shown schematically in the left portion of figure 1.

The basic $\Delta P/P$ schedule is established by a cam contour on the speed-temperature cluster. Cam travel is transmitted by a shaft to the boost linkage. The boost input linkage incorporates a pair of nonparallel levers with a wedging roller between. This roller is positioned by a level adjusting remote control setting motor acting through a temperature biased cam. In this manner the trim authority is limited in the up and down trim regions as a function of T_{t2}. The boost piston rotates the $\Delta P/P$ request cam, which is contoured as the natural logarithm of the schedule. The logarithmic schedule positions one end of the walking beam error link.

The sensor is a parallel track moment balance system which balances the moment of a $P_{T3} - P_{S3}$ differential pressure force, sensed through a bellows, acting at a fixed lever arm against a P_{T3} pressure force which is sensed by an evacuated bellows and acts through the roller-to-pivot lever arm. The roller position is the pressure ratio, delta P/P, when the moments balance. An error in the sensed delta P/P from the schedule value rotates the torque tube sealed output shaft, displacing the double servo flapper valves. The piston displaces accordingly, which in turn translates the roller to return the system to null balance. In this manner the piston position is proportional to delta P/P, as sensed. The piston rotates the delta P/P sensed cam which, like the request cam, is contoured as the natural logarithm of the sensed ratio. This cam positions the remaining end of the walking beam error link. Logarithmic contours are used to provide incremental changes which are percentages of the ratio at the initial point.

With one end of the error link positioned by the schedule and the other by the sensed pressure ratio, displacement of the center from its null position represents percent error between schedule and actual pressure ratios. This error acts through the reset lever to position a slide valve which operates the integrating piston, and the slide valve position itself is the proportional input to the area feedback. The proportional plus integral settings are used to correct minor errors in scheduling of the duct nozzle area. The reset lever pivot position is displaced when an error exists in the duct fuel scheduling system to effectively call for a percentage increase in the scheduled delta P/P. The reset signal displaces the reset piston rapidly, but when it is cutoff removal of reset is at a rate dependent upon flow rate through the discharge bleed.

Duct blowout is detected by displacement of the proportional linkage by an amount representing a 15% high delta P/P error. At this point, an auxiliary link bumps and opens the blowout detector.

2. Duct Nozzle Area Control - Nonduct Heating

The portion of the schematic applicable is shown in the upper left center of figure 1.

During power lever selection throughout the nonduct heating regime, the duct request input to the area servo scissors link is constant. The control parameters for the duct area are therefore only the delta P/P system and the area limiter system. Normally, the delta P/P system will control the nozzle area. As previously discussed, an error from schedule displaces the proportional valve which moves the area feedback follower link, causing the scissors to move the flapper of the boost system to the nozzle area control slide valve. The integral piston is also moved at a rate proportional to the error to displace the area feedback cam axially, thus adding to the integrator piston input to the scissors. The output of these mechanisms is servo boosted to actuate the slide valve which controls flow to the duct nozzle actuators. Area feedback from the nozzle tends to null the previous scissors inputs, resulting in a nozzle area rate of change proportional to the integrator rate which in turn is proportional to the initial error in delta P/P. The system returns to rest when the delta P/P error has been satisfied, causing the proportional piston to return to null and the integrator to stop.

The sequence described above provides for proper duct nozzle operation at and near maximum rotor speed, where a variation in nozzle area with speed is desired. However, during start and operation in the lower rotor speed range, a constant nozzle area is satisfactory. The desired delta P/P schedule results in essentially a constant duct area and since the signal strength of the delta P/P parameter is low and erratic during engine start and at low speeds, it is desirable to schedule the area to a fixed value. The delta P/P schedule in this range is established to provide a slightly larger nozzle area than actually desired. The area scheduling cam, shown at the bottom of the schematic to left of center establishes a scheduled area such that at the limit of the integrator area trim authority, the area will be 4.5 square feet. In this manner the maximum nonduct heating area is limited.

3. Duct Nozzle Area Control Duct Heating

When duct heating is in operation, the duct nozzle area (A_j) scheduling cam is rotated as a function of power lever angle and translated by T_{t2}. The duct request minus feedback error to boost provides proper hold and sequencing as required to accomodate the duct metering system. Correlation between A_j and fuel is maintained by scheduling A_j simultaneously with duct ratios. The A_j schedule is transmitted to one arm of the scissors link at the top of the schematic. An increase in A_j request raises the scissors input lever which closes the area boost flapper. This results in a slide valve displacement to increase nozzle area. As the nozzle area changes, the feedback cam raises the feedback arm of the scissors to renull the servo slide valve. The slide valve is nonsymmetrical and provides for a faster opening rate than closing rate.

The area is subject to continuous setting during duct heating to maintain delta P/P as scheduled by the proportional plus integral output of

the delta P/P system. Area lead during increase of duct heating and lag with decrease is provided by the reset of delta P/P with the error valve linked to the duct request of feedback error lever.

4. Nozzle Area Feedback Cable Failure

When a cable failure occurs, the A_j feedback cam rotates beyond its normal range to a level indicating an area so small that no request can be satisfied. The resulting error drives the nozzle wide open. For most operating conditions the resulting delta P/P error will trigger the blowout detector and shut down the duct flow.

C. DUCT HEATING FLOW CONTROL FUNCTION

Fuel flow to the duct burners is scheduled during steady state as a direct function of power lever angle biased by T_{t2}. Transiently, however, flow is sequenced through a latch device. Various arming devices are required to prevent duct heating until engine operating conditions are favorable.

1. Duct Heater Arming and Blowout Circuit

Duct flow responds to power lever request changes through a servo actuation system. The arming circuit prevents operation of the scheduling system by venting servo pressure from the actuation system to P_{fcb} until all other requirements for duct heating are met.

The duct initiation valve is vented to P_{fcb} through the Zone I cutoff during nonduct heating operation. Movement of the power lever to the duct heating range repositions the valve due to cam lift and ports P_{fs} to the system through the line labeled P_{fda}, which is channelled through the blowout valve to the speed arming valve. The speed arming valve is closed only when the engine is above 80% speed. Since both duct burning and speed selection is made with the power lever, a high speed must be requested when the power lever is advanced to the duct burning regime.

Once duct heating has been initiated and Zone I cutoff pressure becomes P_{fs}, retardation of the power lever does not vent the arming signal to P_{fcb} until the actuator has reduced the fuel to the minimum level and cutoff Zone I flow. This interlock is provided to prevent severe transients in the duct.

In the event of duct blowout, the blowout detector at the pressure ratio sensor (previously described) vents P_{fbo} to P_{fcb} causing the duct blowout shuttle valve to stroke to the left. This cuts off the normal duct arming supply to the duct actuator and vents the actuator to P_{fcb}, causing the actuator to retract at maximum velocity and cutoff from the duct. As long as the duct arming signal is supplied to the blowout shuttle, the duct system is not operable. Retraction of the power lever to the nonduct heating range is required to vent the shuttle and permit the return spring to rearm the blowout system. Reinitiation of duct burning is disallowed until the duct fuel has been shut off and the Zone I cutoff pressure returns to P_{fcb}. This interlocking action assures that a relight cannot be attempted until the duct system has returned to its starting position.

2. Duct Fuel Flow Computation

The portion of the control used in computation of duct heating fuel flow is shown schematically below the center of figure 1.

As discussed in the section on area control, the duct request cam forms the request for the duct heater meter. Actuation of the scheduling camshaft is by a double diameter piston controlled by a flapper servo. The flapper position is an indication of the error between the request and the camshaft position as indicated by the duct feedback cam. The camshaft drives a timing cam, a duct scheduling cam and the area scheduling cam.

Prior to the circuit being armed, the actuator is at the extreme right, the timing valve is at its top position, and the scheduling cam is at minimum schedule. Once armed, the request holds the flapper closed, causing the actuator motion. The feedback cam is on a flat, causing the actuator to attempt to continue stroking. The timing valve is moved along with actuator motion and provides a signal pressure to the Zone I cutoff to open and vents P_{fs} to the latch release. The hold land in the timing valve arrests the duct actuator at this point by venting P_{fx} pressure. The system is now in a hold state with Zone I turned on at minimum schedule level. The Zone I signal also arms the timer and starts the ignition.

The manifold fill system is located in the duct metering body to be discussed later. When the manifold is filled, the fill detector vents the duct ignition piston to continue the duct ignition for a period of time by supplying pressure to a pressure actuated electrical switch. Near the end of the piston stroke, a latch release port is pressurized, and the latch release plunger strokes to close the P_{fx} hold vent. Completion of the piston stroke results in decay of chamber pressure and the adjacent shuttle valve strokes. This valve vents the piston which turns off the ignition and returns the piston to its initial position. A port is also uncovered which provides Zone I signal pressure at P_{fs} level to the Zone II arming port in the arming valve. Zone II may now be started to quick fill if the original request is made at the power lever. As the piston advances in Zone I following latch release, the valve recloses the port to the latch release as well as the hold port, and the release plunger returns to the retracted position with the hold port vented. The duct scheduling cam is at its minimum schedule level during Zone I initiation and latch. If the power lever request is within the Zone I range, the scheduling piston continues to stroke until the feedback cam satisfies the request.

If the Zone II is selected and the shuttle valve is stroked, the Zone II cutoff signal is transmitted to the met ing section to open the cutoff, initiating Zone II manifold fill. If maximum Zone I is reached before the Zone II manifold is filled, further downstroke of the valve vents P_{fx} from the scheduling piston at this point, resulting in schedule hold. When the Zone II manifold fill sensor indicates a fill and the shuttle valve (see the Flow Section) operates, a plunger provides P_{fs} pressure to the latch release plunger, closing the P_{fx} vent and releasing the scheduling piston into Zone II.

Advancement into Zone II closes the vent line and the arming valve supplies P_{fs} at this point. Thus, retarding the power lever below the Zone II position prevents shutoff of Zone II until the scheduling piston has reached the maximum Zone I position.

A dual servo is shown at the duct scheduling piston. At low values of T_{t2}, both servos are operative and the system is relatively fast in scheduling. As T_{t2} increases with flight Mach number, a range is reached where the inlet duct is started and rapid transients are not advisable. A T_{t2} actuated valve cuts off one of the servos, resulting in slower scheduling of fuel and area above this temperature.

The follower linkage of the fuel ratio schedule cam to the boost servo is a differential lever arrangement with a roller adjusting the lever ratio. The levers are parallel at the minimum schedule level, thus making the manual adjustment a duct schedule slope adjustment. The boost is a followup servo and strokes a follower on the multiplying cam. The multiplying cam is rotated by the P_b servo, also used by the main multiplier. The height of the movable follower is picked up by a second follower, the level of which is proportional to fuel flow and is the input to the metering section.

D. FLOW METERING SECTIONS

This portion of the control contains the main and duct fuel meter valves, head regulators, cutoff valves, and mechanisms associated with operation of these valves.

1. Main Flow Section

Referring to the overall system schematic, fuel enters the flow section from the main engine driven pump through a wash type filter (for servo flow) and a screen total flow filter. Wash filtered fuel is channeled to the servo pressure regulator which throttles the fuel from entering P_{f2} pressure to maintain a constant P_{fs} - P_{fcb} pressure drop. P_{fs} supplies all computer servos. Nozzle flow is scheduled by a plug type metering valve across which the P_{f2} - P_{f3} drop is held constant. Fuel then passes through the pressurizing cutoff valve to the engine nozzles. Excess pump delivery is returned to pump interstage from P_{f2} by action of the bypass valve.

Scheduled fuel flow appears as a push rod position at the flow body-computer interface. A follower lever with a high-to-low pressure fuel seal amplifies this request and positions the metering valve with a following servo piston.

Metering valve servo supply is wash filtered P_{f2} referenced to metered pressure, P_{f3}. The metering valve is contoured to provide an effective orifice area proportional to valve displacement from the closed position; in this manner, the valve position is also directly proportional to scheduled fuel flow. The adjustable follower on the positioning lever permits independent calibration of the flow body to preset levels, assuring interchangeability. The minimum flow stop is also shown acting at the servo arm.

The bypass valve is of the integral plus proportional type. A bellows sensing the P_{f2} - P_{f3} head is opposed by a temperature compensated spring load. An error in sensed head displaces a flapper servo from null, causing the integrating piston to reposition a sleeve type bypass valve. A plug valve within the integrator valve also senses P_{f2} - P_{f3} across its piston area, which is opposed by a spring. This is the proportional valve which acts to correct the regulated metering head during rapid transients. The shutoff shaft rotates the shutoff cam which operates the shutoff pilot valve and provides a mechanical positive drive to close the shutoff valve in the event that hydraulic actuation should fail. The shutoff pilot valve, as the shutoff lever is moved to turn on, switches $P_{fs/o}$ from P_{f2} to P_{fo} to close the manifold drain, switches the shutoff valve reference pressure from P_{f2} to P_{f1}, switches the windmill bypass reference pressure from P_{f1} to P_{f2}, and switches signal pressure $P_{fo/s}$ from P_{fo} to P_{f2}. These changes occur in the sequence named. The manifold signal is externally supplied to the engine manifold drain valve. The shutoff valve is permitted to open when referenced to P_{f1} but performs as a pressurizing valve at low flows to assure sufficient pressure for servo operation. The windmill bypass valve passes metered flow to the bypass channel during shutoff and maintains pressurization at that time. A P_{f2} reference closes it.

The $P_{fo/s}$ signal is used to operate a duct shutoff interlocking valve shown at the left on the schematic adjacent to the timing valve. Unless pressurized to P_{f2} pressure, this valve vents all duct shutoff pressures, thus assuring duct shutoff with the main.

2. Main Burner Schedule Reset on Duct Burner Blowout

Main burner reset on duct burner blowout is accomplished through two servo orifices which sense pressure ratio error at the proportional link in the computer through a set of levers. At 5% error the main burner derich valve opens, reducing pressure on the spring loaded switch piston in the fuel body. The spring load is such that the valve will not shuttle until the pressure is also removed from the blowout piston. At 15% error the second orifice, called the blowout detector, opens. This vents the floating piston, and with no pressure force resisting the spring, the valve shuttles which reduces the pressure back of the derichment shutoff valve. This shutoff opens, allowing metered fuel to be regulated through the compressor discharge referenced regulator. At the second servo valve opening, the blowout valve is actuated which shuts the duct down and the pressure source to the second servo valve is removed until rearmed by recycling of the power lever.

As the pressure ratio error is reduced below 15%, the second servo valve closes but there is no pressure buildup because the pressure source is shut off. The main burner remains deriched until the error reaches 5% and the first servo valve closes, building up the pressure behind the switch piston and the valve shuttles closed. This closes the derichment shutoff valve and the main burner returns to schedule.

3. Duct Flow Section

Referring to the system schematic, the duct metering section is in the lower right hand portion. Flow enters from the air turbine driven pump at pressure P_{f1d}. The wash filtered P_{f1d} fuel is used in the metering and throttling servos. Discharge from the wash filter is at P_{f2d} pressure. The flow then passes through the metering valve, the throttling valve, and the shutoff valves to the zone nozzles. A check valve assures wash filter velocity.

Scheduled fuel flow appears as a pushrod position at the flow body-computer interface. Servo positioning of the metering valve is as previously described for the main system. Metering head, $P_{f2d} - P_{f3d}$, is maintained by an integrally positioned throttling valve. This pressure head is sensed by a compensated, spring opposed, bellows with head error establishing the integrating rate of the throttling valve. Using P_{f1d} wash filtered fuel at a pressure above P_{f2d} to power the metering valves and throttling valves assures that piston leakage is from clean-to-dirty fuel, thus preventing contaminant from seizing the pistons.

The metering valve has a dual gain contour to provide sufficient accuracy within the positioning accuracy attainable at low flows.

The zone shutoff pilot valves are operated by servopressure, controlled by the duct control valves. The valves vent the pilot pistons to P_{fcb} during shutoff and P_{fs} during flowing operation. Each pilot piston operates a double seating poppet valve. In shutoff position, the poppet valve supplies entering fuel pressure to the back side of the shutoff valve, and the differential pressure assists spring closing of the valve. In operating position, with P_{fs} supplied to the pilot piston, the back side of the shutoff valve is vented to P_{fcb} by the poppet valve allowing the $P_{f4d} - P_{fcb}$ differential acting on the outer circumferential area of the shutoff valve to overcome the spring and open the valve. The pressure behind the shutoff valve is supplied to external ports as control pressures for the manifold drain valves.

Zone I and Zone II flow sections are identical as described thus far. A groove on the Zone I shutoff valve is vented to P_{f4d} and is aligned with the cooling flow port when the valve is in shutoff, permitting a flow through the pump and control when duct heating is not in use. The multiplying cam is specially contoured to provide 3000 pph for cooling flow at shutdown. While operating with flow levels less than 3000 pph, a separate valve opens to vent unmetered flow through the shutoff valve groove to the cooling flow outlet through a metering bleed. This adds 2200 pph to the pump flow.

A zone manifold fill detector and manifold quick-fill valve is incorporated for each zone. The detectors consist of an area matched bellows and piston, spring-loaded to operate at a preset rise in manifold pressure above duct pressure. This piston is referenced to manifold pressure and the bellows to duct pressure. When the Zone I shutoff valve opens, pressure P_{f4d} is essentially Zone I manifold pressure and is low when the manifold is empty. The double acting servoflapper

in the Zone I detector will close the ignition piston vent and open the manifold fill servo orifice, which results in opening of the Zone I quick-fill valve. When the manifold is filled, the pressure rise on the piston strokes the flapper to close the fill servo orifice and consequently the quick-fill valve, and to also vent the ignition piston port to start the ignition hold previously described.

The Zone II detector piston is always referenced to P_{f5ZII}, and consequently the fill servo orifice is open for all conditions except Zone II operation. Thus, during nonaugmented and Zone I operation, the Zone II quick-fill shuttle is stroked to its left relative to the schematic drawing. In the left position, the main flow channel is closed and the fill port is open. When the Zone II shutoff valve opens, flow to the manifold is supplied through the quick fill opening. When the manifold fills, increase in manifold pressure acting on the detector piston closes the servo port to stroke the fill valve to the right. This action shuts off the manifold fill and opens the Zone II outlet to the metered flow channel. Division of metered flow is subsequently determined by spray nozzle characteristics. A small slide servo is also operated by the manifold fill valve actuation pressure to transmit a pressure to the fluid latch releasing the hold on the scheduling piston.

4. Turbopump Control System

This system operates to regulate the pressure drop across the duct fuel metering section ($P_{f2d} - P_{f4d}$) to a value slightly higher than minimum requirements for control metering functions. In this manner, the operating range of the control throttling valves is decreased, which improves metering head regulation while extracting a minimum bleed air from the compressor to operate the turbine. This system is shown schematically in the lower right hand portion of the overall schematic.

The control is basically proportional, with the differential head opposed by a spring. The piston and spring are sized sufficiently large to provide actuation torque at the output shaft to the air valve. Stem area is sized to provide an unbalanced force with pressure level which, in turn, is a function of metered fuel flow. Thus, as the flow increases, the controlled pressure drop also tends to increase. This effect crudely attempts to offset the proportional droop which normally takes place with air valve opening.

E. AUXILIARY CONTROL FUNCTIONS

Auxiliary functions are those performed by the control but are not directly involved with metering or scheduling of fuel or control of the duct nozzle throat area.

1. Compressor Bleed Valve Control

The compressor bleed valve control consists of a servo actuated double butterfly valve to supply pneumatic actuators for the compressor bleeds with compressor discharge pressure as the air supply. The compressor bleeds are opened and closed at scheduled values of a speed and

temperature function. The bleed valve schedule is a contour on the speed temperature cluster cam. The servo piston operates full stroke when the cam moves the slide servo across the null value. A hysteresis piston assures positive operation by resetting the pivot on the follower arm as the slide valve crosses the null point.

2. Thermostatic Bypass to Tank

The thermostatic bypass valve is designed to route main engine control bypass fuel to the airframe tanks when the fuel temperature entering the control exceeds 300°F, and metered main fuel flow is less than 5000 pph. The entering sample consists of the servo regulated supply to the control servos. A spring opposed bimetallic stack positions a servo which in turn operates the bypass valve. This valve normally closes to open the bypass valve at 300°F to 310°F. If the metering valve is opened to 6000 pph or more, a port is uncovered which actuates a piston compressing the spring against the bimetallic stack. This delays opening of the bypass until the temperature is in the 335°F to 345°F region. An externally accessible adjustment will be provided so that the thermal bypass valve settings can be set 50°F lower.

3. Inlet Guide Vane Control

The inlet guide vane control positions the compressor stators to two positions by supplying pressure to the engine mounted actuator. Position shift occurs as a function of N_2 and T_{t2}. This schedule is on a cam shown in the center of figure 1. At the scheduled point, the cam strokes a flapper servo to provide boost power to the controlling slide servo. Hydraulic supply is main pump discharge, P_{f2}, and drain is to control body P_{fcb}. Stroking the slide valve reverses the pressures in the load lines.

4. Reverser-Suppressor Control

The reverser-suppressor control consists of a servo powered slide valve which vents load line (A) to airframe boost pressure, P_{fo}, in the forward selected position and supplies hydraulic pressure, P_{fh}, to select reverse. This is shown schematically in the lower left hand corner of figure 1. The slide servo boost piston integrates stop to stop at the command of a cam driven slide valve. The cam rotates with power lever angle, cam rise requesting forward thrust. Thus positive drive is provided to the command servo in the forward selection. This servo has a piston integral with one end. A speed actuated valve is bumped open by the speed feedback lever and supplies pressure to this piston at speeds greater than 90% of rated. This pressure acting on the piston overrides the cam follower spring and prevents the command servo from moving to the reverse position until the rotor speed falls below 90%.

5. Secondary Air Control Signal

The cutoff shaft operates a valve to provide a secondary air control signal when the cutoff lever is fully advanced. A cam operated valve is shown which provides P_{f1} normally and P_{f2} at full cutoff lever advance.

SECTION IV
FUEL AND LUBRICATION SYSTEM

A. DESCRIPTION

1. General

The JTF17 engine fuel and lubrication systems will utilize the same type of hardware that has proven successful in service operation on current Pratt & Whitney Aircraft commercial and military engines. This experience includes the J58 engine, which has accumulated considerable sustained flight time above Mach 3. The development experience, personnel, and facilities used to develop the fuel and oil systems for the J58 will be used in designing and developing the fuel and oil systems for the JTF17 engine. The extensive background and experience accumulated through millions of hours of commercial jet engine operation and thousands of hours of supersonic military jet engine operation at extreme environmental conditions will be directly applied to the JTF17 engine design.

The complete fuel and lubrication system for the JTF17 engine is composed of the following five subsystems:

1. Gas generator burner fuel supply system
2. Duct heater fuel supply system
3. Hydraulic system
4. Lubricant supply and scavenging system
5. Breather system.

The integrated fuel and lubrication system components and interconnecting lines are shown in figures 1, 2, 3, and 4. For clarity, the single-piece unitized control is shown as a separate component in each subsystem flow path. Detailed descriptions for the fuel system and hydraulic system components are included in Volume III, Section III. Description of the interconnecting lines is included in Volume III, Section II, paragraph I.

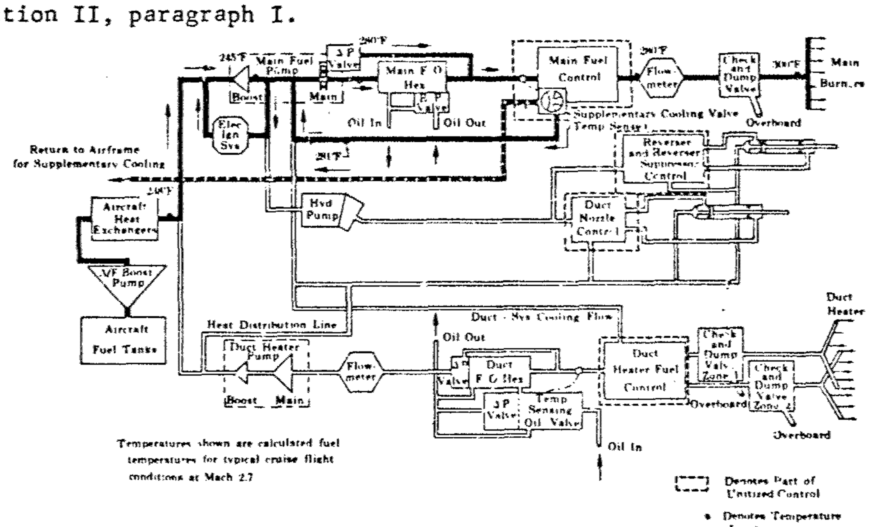

Figure 1. JTF17 Gas Generator Subsystem

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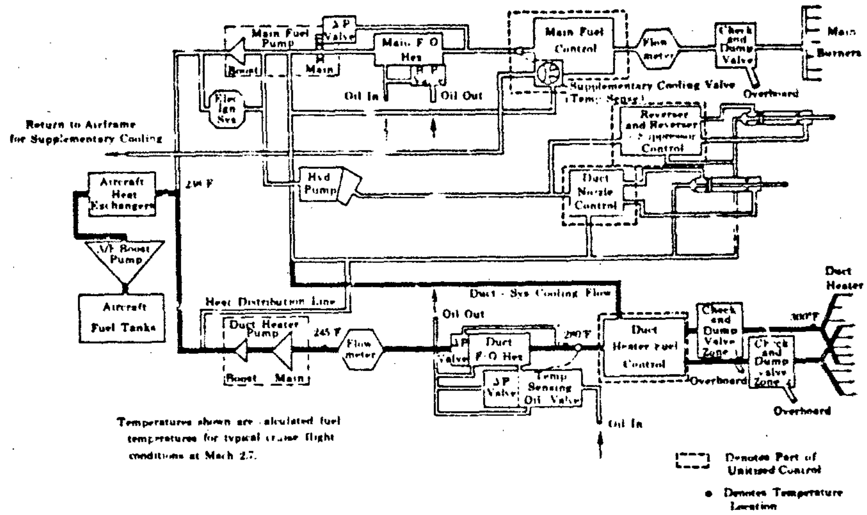

Figure 2. JTF17 Duct Heater Fuel Supply Subsystem

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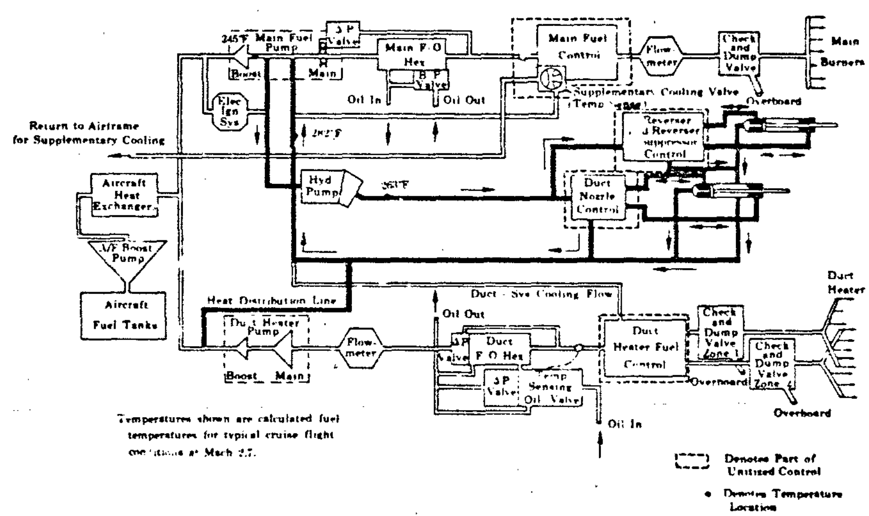

Figure 3. JTF17 Hydraulic System

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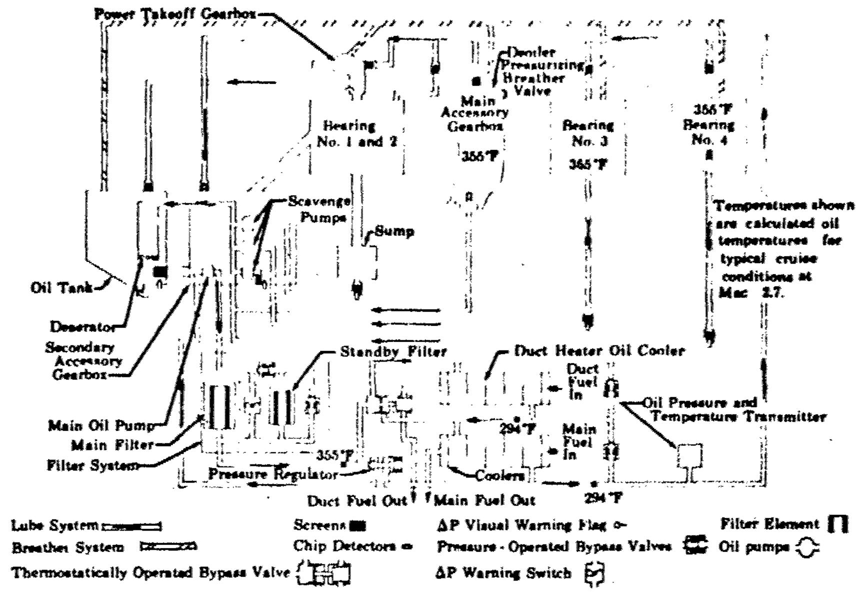

Figure 4. JTF17 Lubrication and Breather System Schematic FD 16544 BIV

2. Gas Generator Fuel Supply System

Fuel supplied to the engine separates into two basic paths; one for the gas generator and one for the duct heater system. Fuel in the gas generator system first passes through the impeller or boost stage of the fuel pump where the pressure is increased to approximately 150 psia. This point in the flow path is called the pump interstage. It is used as a supply and return point for the hydraulic system, a supply for ignition system cooling, and a return point for recirculation flow from the main fuel control. Fuel is then pumped through the gear stage where the pressure is raised to a level necessary to overcome the pressure drop of downstream components from the oil cooler to the burner nozzles. Both impeller and gear stages of the pump are gear-driven. Fuel next passes through the gas generator fuel-oil cooler and is then routed through the main fuel control. The control incorporates a thermal bypass valve that either returns unscheduled flow to pump interstage if below a maximum temperature limit or to the aircraft fuel tanks. Fuel is returned to the aircraft tanks only during descent with this system. The fuel passes from the fuel control, through a flowmeter and then through a check and dump valve, which is used to drain residual fuel from the manifolds when the combustors are shut off to prevent accumulation of fuel varnish deposits. The fuel then is routed to the gas generator fuel manifold and then to the nozzles.

3. Duct Heater Fuel System

The duct heater fuel pump has two impeller stages and is driven by an air turbine. Turbine power is supplied from the gas generator diffuser high pressure air bleed source. The fuel pressure is increased by the pump to a level necessary to feed all downstream components to the duct heater nozzles. Fuel passes from the pump discharge through a

flowmeter and then through the duct heater fuel-oil cooler, which is located upstream from the control to utilize full pump flow. The fuel passes from the cooler through the duct heater fuel control, where it is separated as scheduled by the control into two parallel flow paths for Zone I or Zone II duct heater fuel nozzles. Prior to entering the heater nozzles, the fuel in each zone passes through a check and dump valve. These valves have the same protective function as described for the gas generator system.

To eliminate the heat returned to the aircraft fuel tanks except during descent, the engine system is designed to operate with the maximum permissible fuel temperature into both gas generator and duct burner systems. To accomplish this, and maintain an optimum heat balance between the two systems, some fuel is transferred from the hydraulic system to the duct turbopump inlet through a line known as the heat distribution line. In addition, to provide a cooling flow through the duct heater pump during nonaugmented operation, a duct system cooling flow line is incorporated that bleeds fuel from the duct heater fuel control back to the gas generator fuel system.

4. Hydraulic System

The primary function of the hydraulic system is to supply high pressure fuel to the reverser-suppressor and nozzle actuators. As shown in figure 3, fuel is supplied to this system from the gas generator fuel pump interstage. The fuel pressure is increased by 1500 psi as the fuel passes through a gear-driven, constant pressure rise, variable displacement hydraulic pump. The high pressure fuel is routed to the reverser-suppressor control, duct nozzle control, and their respective actuators. These actuators are described in detail in Volume III, Report B, Section II, paragraph F. A constant cooling flow of 15 gallons per minute is provided to components and flow lines at temperatures well below that which would cause fuel varnish deposits. Flow is increased to 55 gallons per minute maximum during periods of component actuation.

5. Lubrication System and Scavenge System

The oil tank is engine mounted. Oil from the tank is gravity fed to the main oil pump located inside the main accessory gearbox. The oil pump is a gear-driven, positive-displacement gear pump, which increases the pressure by approximately 100 psi. This high pressure oil then flows from pump discharge through the main filter system. This system consists of a primary filter, standby filter, and a bypass valve for each. Under normal operating conditions, full pump flow passes through the primary filter. If the primary filter pressure drop exceeds 35 psi due to a clogged filter element, the primary bypass valve opens and routes oil through the standby filter. If the standby filter becomes clogged, the standby bypass valve opens and shunts total oil flow past the filter system. Provisions are made in the primary filter to activate a cockpit warning light if pressure drop is excessive. A differential filter pressure indicator, with a pop-out button, is also provided in the filter assembly for ease in ground maintenance checks.

Pressure rise across the oil pump is maintained at a constant level by a pressure regulator located downstream of the main filter system. The pressure regulator meters oil to the system at a pressure and flow rate that is constant regardless of changes in engine speed. Total pump flow passes through the main filter system and that portion of total flow by-passed for pressure regulation is returned to the pump inlet through the pump bypass flow line. Regulated oil then passes through two oil coolers connected in series. These are shell and tube type heat exchangers using fuel as the oil coolant. The first cooler in the series circuit uses duct heater fuel as the coolant and the second uses gas generator fuel as the coolant. Both coolers have bypass vales to shunt the oil around the cooler if the pressure drop exceeds 28 psi. The duct heater cooler also has a thermostatically operated oil bypass valve.

An oil temperature and pressure transmitter, located downstream of the coolers, transmits the regulated pressure and temperature to the cockpit.

The oil is distributed downstream of the coolers through tubing to oil metering jets in the bearing compartments and gearboxes for cooling and lubrication of bearings and gears. Additional protection against bearing wear and damage is provided by passing oil through filtering screens upstream from the oil jets.

Oil that has been supplied to the bearings and gears drains into sumps where it is picked up by scavenge pumps and delivered back to the oil tank. Five pumps are used in the scavenge system. Three of these are located in the accessory gearbox and handle return flow from the No. 1 and 2 bearing compartment sump, the No. 3 compartment, and the gearbox. The other two are located in the main gearbox and the No. 4 bearing compartment. The inlet of each pump is covered with a large area coarse strainer to prevent metal particles and other foreign objects from damaging the pumps. These pumps are gear-driven and use gears for positive oil displacement. All of the scavenge pumps return oil to the oil tank through a de-aerator. Scavenge oil is directed tangentially against the inner wall of the de-aerator. The oil and foam form a vortex inside the de-aerator, centrifuging heavier oil to the wall, where it spirals downward and exits through holes in the lower periphery of the de-aerator. The air is forced to the center of the vortex and exits through a pipe in the center. The de-aerated oil settles to the bottom of the tank and the cycle is repeated.

6. Breather System

The breather system vents the air that flows past bearing compartment seals to the atmosphere. The primary functions of the system are to maintain the lubrication system pressure at a level that ensures a positive pressure differential across bearing compartment seals; to prevent oil loss, and to provide adequate inlet pressure for the oil pumps to prevent cavitation. Breather pressure is maintained at an equal level between compartments, oil tank, and gearboxes by interconnecting plumbing. The remainder of the breather system consists of a de-oiler and a breather pressurizing valve. The de-oiler is used to separate entrained oil from the air entering the breather pressurizing valve to minimize engine oil consumption.

B. DESIGN OBJECTIVES AND REQUIREMENTS

1. Introduction

The fuel and oil system for the JTF17 engine is designed to provide adequate cooling and lubrication for the engine gears, bearings, seals and bearing compartments and to maintain fuel and oil temperatures below the level at which coking might occur in the oil systems, and varnish deposits may be left in the fuel system. In addition, it is required that high component reliability be achieved through the use of fluid bypass valves, redundant oil filtration and breather system air filtration. Maintainability goals required that filters and chip detectors be located external to the engine and that maximum oil fire and cabin bleed air contaminant safety protection be provided.

2. Specific Objectives and Requirements

The specific design objectives and requirements are:

1. Breather pressurizing air supplied to the bearing compartments will be maintained at a temperature no higher than that of present subsonic engines.
2. The fuel temperature at the gas generator and duct heater fuel nozzle will be maintained below 300°F for type Jet A fuel, and 350°F for PWA 533 fuel, to avoid varnish deposits in the fuel systems resulting from fuel decomposition.
3. The oil temperature in all compartments and gearboxes will be maintained below 400°F to avoid coking.
4. The oil system shall adequately cool and lubricate the engine gears and bearings and cool the bearing compartments and gearboxes to levels required to obtain the design TBO and a minimum oil change frequency of 300 hours.
5. The oil system shall be supplied as part of the engine and all components will be designed with adequate margin to assure that oil temperatures are maintained below the level at which coking might occur.
6. Maximum protection will be provided to avoid foreign object accumulation in the compartment seal pressurization supply air system, and oil vapor contamination of the cabin air bleed system.
7. Easy access will be provided for service checking of pumps, filters, and chip detectors.
8. The rate of heat generation from all sources within the fuel and lubricant system will be kept to a minimum.
9. The temperature level and flow rate of air supplied to the bearing compartments will be kept to a minimum.
10. Maximum oil consumption is limited to 0.25 gallon per hour.
11. Maximum allowable engine fuel inlet temperature, as shown in figure 5 for type Jet A fuel, and in figure 6 for PWA 533 fuel.
12. A constant oil pressure of 45 psi across oil jets.
13. An oil flow commensurate with a maximum 75°F oil temperature rise across bearing compartments.
14. A minimum breather pressure of 4 psia.
15. A minimum scavenge pump capacity at idle rpm of three times the flow being delivered to protect against possible flooding of bearing compartments and gearboxes.

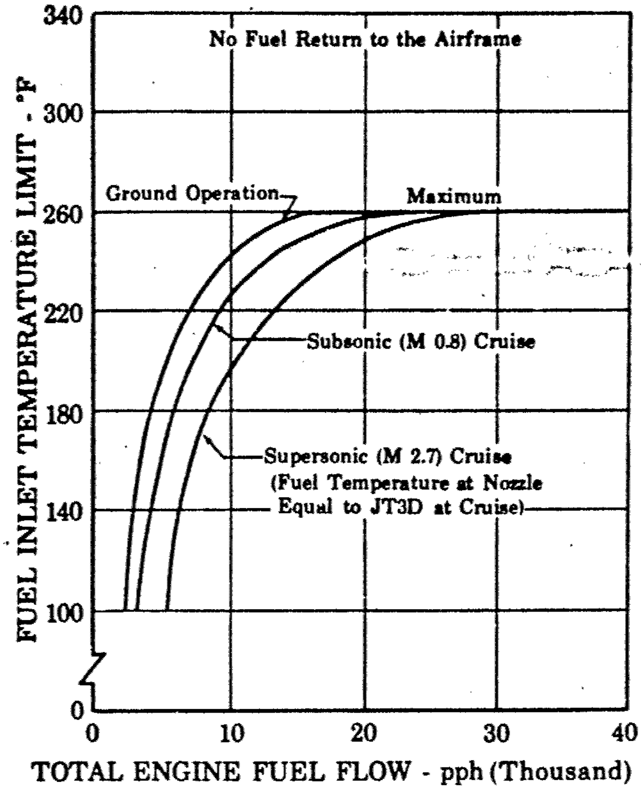

Figure 5. Fuel Inlet Temperature Limits for PWA 522 (Jet A, A-1) FD 17635 BIV

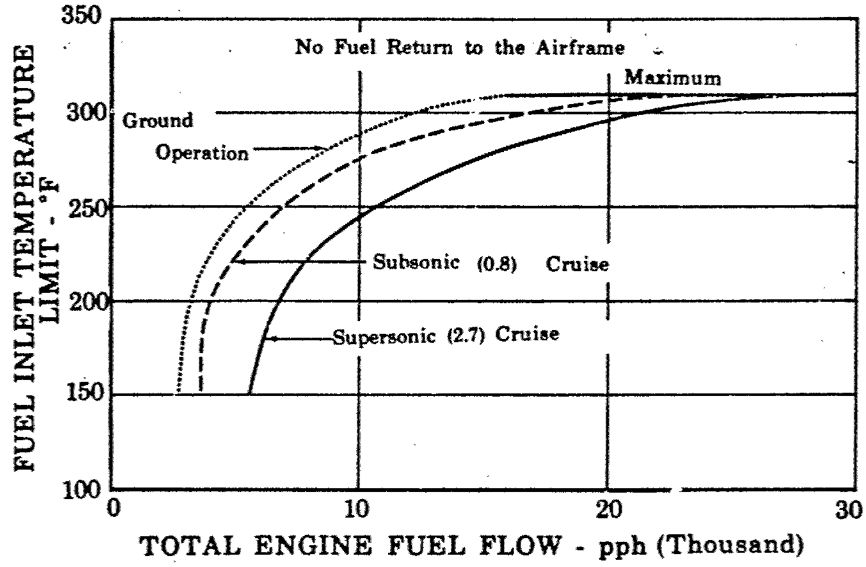

Figure 6. Fuel Inlet Temperature Limits for PWA 533 FD 16995 BIV

C. DESIGN APPROACH

1. System Design

a. General

The primary requirement in designing the lubricating system is to provide adequate engine lubrication with oil temperatures substantially below the coking level. The three prime desing goals for the fuel systems are to: (1) permit satisfactory engine operation for the design TBO periods with current commercial aviation Jet A and Jet A-1 fuels, (2) to maintain fuel temperatures below the varnish deposition level, and (3) prevent heat return to aircraft fuel tanks during takeoff, climb, acceleration, and cruise.

The amount of heat returned to the aircraft fuel tanks is a function of the engine heat rejection, fuel inlet temperature, maximum allowable fuel temperature, and the flow rate of fuel to the combustors. Because dissipation of heat generated by the aircraft-engine system is one of the major engineering problems that must be solved in a supersonic aircraft of this type, every effort has been made to minimize the heat rejected to the engine. Heat rejection design studies performed during Phases II-B and II-C have shown that bearing compartment seal rubbing friction is a major contributor to engine total heat generation. Seal heat generation has been virtually eliminated in the JTF17 engine by incorporating nonrubbing hydrostatic seals. The seals are discussed in detail in Volume III, Report B, Section II, paragraph G. Another major source of heat generation is the heat transferred to the oil from the ambient air surrounding the oil tank and bearing compartments. The JTF17 engine uses a "hot tank" oil system, which means that the oil tank is upstream of the oil coolers. This tank system has proved satisfactory on several commercial engines and results in minimum temperature differences between environmental air and oil, and therefore minimum heat transfer at high Mach numbers. At subsonic conditions the use of a hot tank system further results in a heat transfer benefit, because heat is transferred from the tank to the cooler environmental air.

Minimum heat transfer to the oil in bearing compartments is achieved by: (1) using relatively cool fan discharge air to pressurize bearing compartment seals; (2) designing for a controlled low flow rate of this air into the compartments; and (3) bathing the outside of the compartments with the same low temperature air. The latter provides an environment around the JTF17 engine bearing compartments that is no hotter than the environment in present subsonic commercial aircraft engines.

Full fuel pump flow, except for the amount being bypassed at high fuel flow conditions to avoid excessive pressure drop, is passed through the oil coolers in each fuel system to accomplish maximum transfer of heat from oil to fuel for all conditions. This is accomplished in both the gas generator and duct heater fuel systems by locating the coolers upstream of the respective fuel controls.

Two oil coolers are incorporated in this system to permit utilization of all of the heat absorbing capacity of the fuel in both the gas generator and duct heater systems. The system supplies hot oil from the tank directly into the duct heater oil cooler. Since thrust requirements for both airframe contractors necessitate duct heating for supersonic climb and cruise, the major portion of the high mach number oil cooling task can be accomplished with duct heater fuel. Thus, during cruise, the gas generator fuel system is not required to carry a high cooling load, and therefore, the fuel is delivered to the gas generator fuel nozzles at a lower temperature.

The duct heater oil cooler utilizes a fuel temperature sensing oil bypass valve to shunt oil around the cooler when fuel discharge temperature reaches a preset value. Diverting oil around the cooler normally occurs only at a low duct heater flow rate, or when the duct heater is not operating.

Maintaining gas generator fuel temperature below maximum temperature limits and at the same time preventing heat return to the aircraft without requiring any aircraft crew action is accomplished by incorporating an automatically regulated fuel bypass valve on the main fuel control. This valve senses cooler discharge fuel temperature, and the temperature regulation is a scheduled function of flow through the valve. At cruise conditions, when fuel temperature is below the sensor setting, the valve bypasses fuel back to the pump interstage. The basic reason for this biased setting is to ensure that fuel is burned as close as possible to the maximum allowable fuel temperature, which ensures against returning heat to aircraft except during descent.

During some descent conditions when total fuel flow is low but fuel and oil temperatures are high, the sensor setting limit will be reached and the bypass valve will return some heat to the aircraft fuel tanks. Because the duct heater is not operating during descent, the duct heater oil cooler bypass valve temperature setting is reached, and most of the heat load is transferred to the gas generator cooler. The combination of two coolers, each with attendant temperature control valves, results in burning fuel in both the gas generator and duct heater at maximum allowable temperature at all conditions thus minimizes heat return to aircraft fuel tanks.

At the present time, the JTF17 does not incorporate an air-to-fuel heater for fuel de-icing purposes because initial coordination with the air frame manufacturers indicated that fuel temperatures at the engine inlet would be above potential fuel icing conditions during all critical flight conditions. A fuel de-icing system can be provided on the engine if the need is established during detailed systems coordination with the airframe manufacturers.

2. Components

a. Fuel and Hydraulic System Components

All gas generator, duct heater, and hydraulic system components are described in Volume III, Report B, Section III.

b. Fuel Nozzles

Description and operation of the fuel nozzles are included in Volume III, Report B, Section II, paragraphs B and D.

c. Lines and Fittings

Descriptions of lines and fittings is included in Volume III, Report B, Section II, paragraph I.

d. Lubrication System Components

(1) Oil Tank

The oil tank is a welded sheet steel container that is shaped to fit the engine contour at the fan discharge case. Figure 7 shows the basic tank configuration which is the same basic design as used on JT3, JT4, JT8, and J58 engines. The total tank capacity of 6.22 gallons is based on a maximum allowable oil consumption rate of 0.25 gallon per hour. Of the total capacity, 4.2 gallons are usable, 1.5 gallons are allowed for expansion space, and 0.52 gallon is unusable. Assuming that oil is consumed at the maximum rate, the supply is sufficient for approximately 17 hours of operation without oil addition.

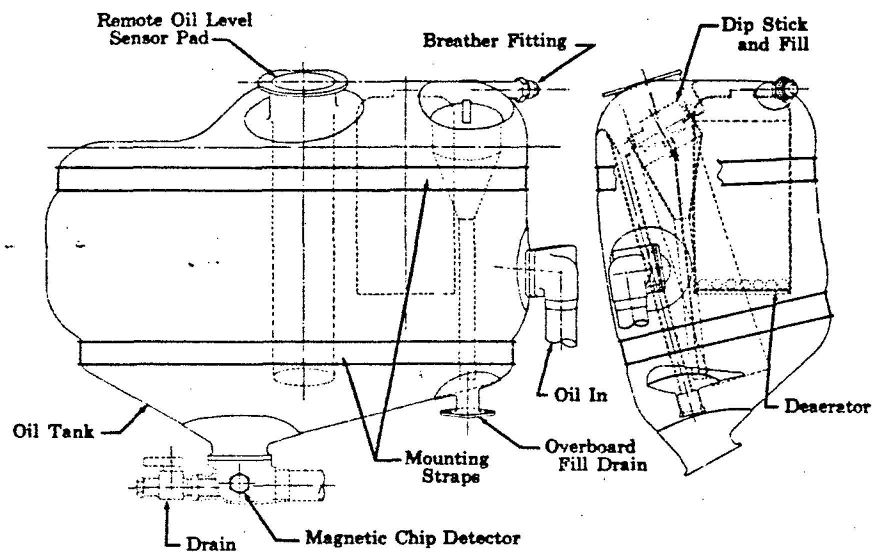

Figure 7. Main Oil Tank FD 16545 BIV

The tank is fabricated by welding and brazing two halves of close tolerance sheet metal shells to form a lightweight, leak-proof structural vessel. All connections to the tank body are flanged forgings, butt-welded to ensure pressure-tight integrity, and to distribute the connections loads into the body. Integral rods tie the two halves of the tank together. These rods protect the vessel from distortion due to transient differential pressure gradients that could result from rapid changes in aircraft environment. All tubing is supported

by these rods to damp vibrations of the tank's internal piping. All corners and contours of the vessel have large radii to reduce stress concentrations and to avoid local thinning during fabrication. The tank contours and parting line between halves are carefully selected to permit the sheet metal to be formed as simply and inexpensively as possible.

The tank is mounted on the compressor fan duct on the forward side and to the accessory gearbox on the aft side by four sets of brackets and two metal straps. Flexible strips serve as bumper pads in the locations of the brackets and provide protection from chafing by the steelstraps. This method of mounting isolates the oil tank from thermal growth and maneuver deflections of the engine case. The straps, which are attached to the brackets and pulled tightly over the flexible strips around the outside of the tank, hold the tank securely against the brackets. This type of support is being used successfully in many Pratt & Whitney Aircraft engines, including the J58 engine.

A valve is located in the bottom of the oil tank to facilitate both draining and flushing. This can be done without removing the tank or the normal oil outlet line to the oil pressure pump. The drain also permits the removal of any sediment and moisture that might accumulate at the bottom of the tank.

(2) Main Oil Pump

The main oil pressure pump is a gear-type, positive-displacement pump and is located inside the secondary accessory gearbox. Figure 8 shows the pump details. This location results in minimum heat transfer from ambient air to the pump.

Figure 9, which is a calculated pump performance map, shows oil system and pump bypass flow rates at idle and rated pump speeds.

The construction of the pump is the same as that used in JT3 and JT4 engines except for the use of silver-lead-indium plated journal inserts to provide additional life. The gear tip clearances are closely controlled for maximum pump efficiency.

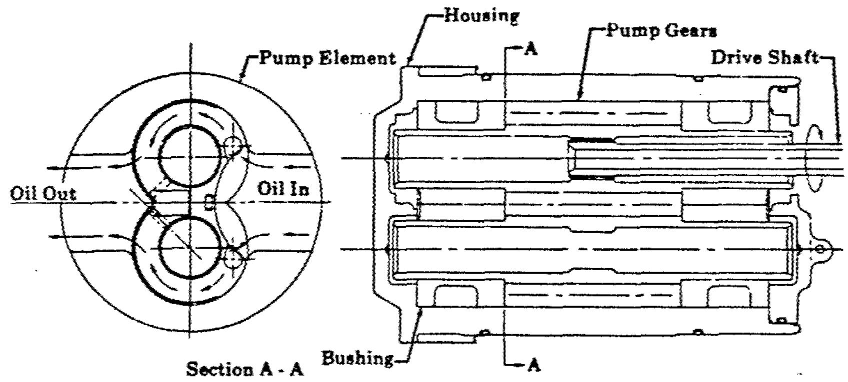

Figure 8. Oil Pump

FD 16571
BIV

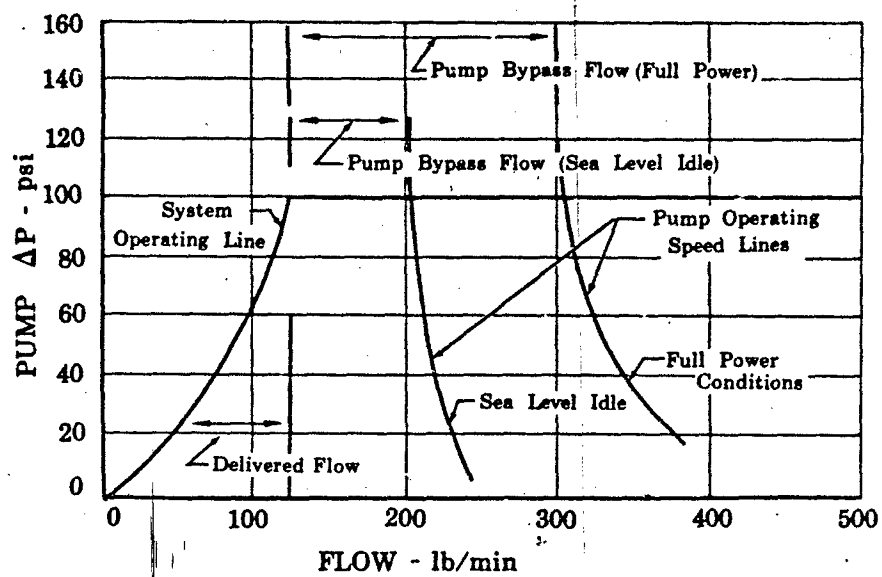

Figure 9. JTF17 Main Oil Pump Performance Map FD 17637
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(3) Oil Filters

The main oil filter incorporates two 40-micron, metal mesh-type filter elements. Two filters are provided for redundancy. In event that the primary filter becomes contaminated, flow is diverted to the standby filter. The filtration capacity of the standby filter is a minimum of 200 hours under normal operation. A warning light will be provided in the aircraft cockpit to indicate when the main filter has been bypassed. Additional contaminant protection for the bearings is provided by passing the oil through "last chance" filters prior to reaching the bearing compartment metering jets. These filters are located externally for ease of maintenance. The mesh size in these filters is slightly smaller than the downstream metering jets.

(4) Oil Coolers

The oil coolers are shell and tube-type heat exchangers. Figure 10 shows the cooler details. The fuel makes one pass through tubes that are crimped to provide maximum turbulence for improved heat transfer characteristics at minimum pressure drop. The oil flows in a counter direction over the tubes within the shell. The basic cooler concept and material selections have been proved in current commercial engines that have accumulated many hours of subsonic experience, and in the J58 engine design, which has accumulated many hours of operation at Mach 3. The particular geometry used in the JTF17 engine design is the result of an oil cooler optimization study resulting in maximum heat transfer capability at minimum weight.

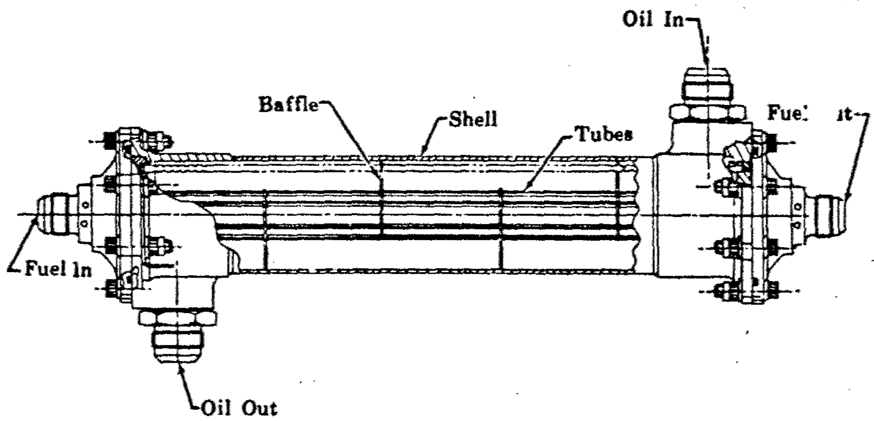

Figure 10. JTF17 Oil Cooler

FD 16886
BIV

(5) Metering Jets

All oil flow is delivered to the bearings, seals, and gears through metering jets located at the point of desired cooling or lubrication. The JTF17 engine uses venturi-type metering jets that provide controlled cone angles for direct impingement on gears and bearing scoops. The combination of the constant displacement oil pump, regulating valve, and proper jet sizing will ensure a constant oil pressure and adequate flow at all operating conditions.

(6) Scavenge Pumps

The scavenge pumps are similar in construction to the main oil pump. Each pump is sized to deliver approximately three times the flow supplied to it while the engine is at idle speed. This basic design configuration and flow ratio has been used on all Pratt & Whitney Aircraft jet engines. Unlike the main oil pump, which has to pump against the pressure drop of the coolers, plumbing and oil metering jets, these pumps only have to deliver flow to the oil tank at a pressure head equal to that of the plumbing downstream of the pump.

(7) De-Oiler

The breather system de-oiler is located in the main accessory gearbox just upstream of the breather pressurizing valve and is used to prevent oil loss through the breather flow system. The de-oiler geometry incorporated in the JTF17 engine is the result of an optimization study to provide a design with maximum de-oiling capability at a minimum pressure drop. The design is the same basic configuration as is used in many other Pratt & Whitney Aircraft jet engines.

(8) Breather Pressurizing Valve

The breather pressurizing valve is provided to keep the pressure in the oil system at a minimum of 4.0 psia to ensure satisfactory oil pump performance at extreme altitudes and, is sized for the flow that would be experienced if one of the bearing compartment seals failed. The basic concept represents simplicity of design with a minimum of moving parts. The bellows fabrication technique, and material have been proved durable in the J58 engine and commercial engines. A schematic diagram of the valve is shown in figure 11.

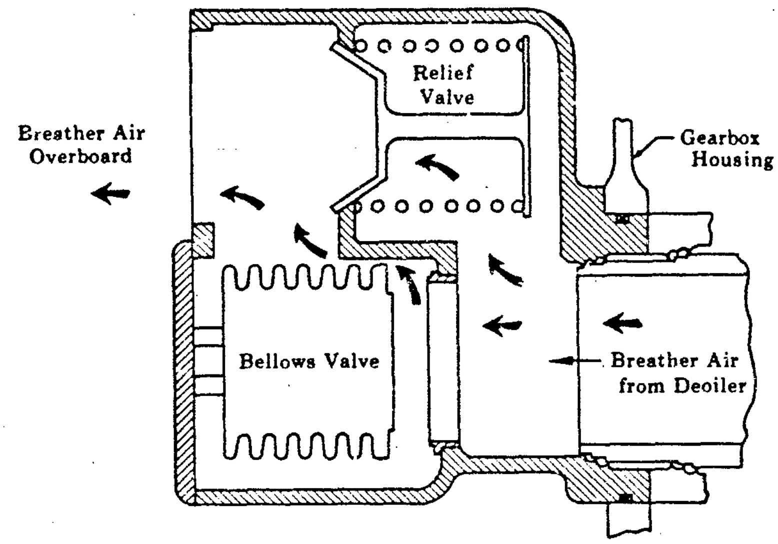

Figure 11. Schematic Diagram of JTF17 Breather Pressurizing Valve FD 16992 BIV

D. FUELS

1. General

The JTF17 engine is designed to operate on commercially available Type A aviation kerosene. This fuel is widely used in current subsonic commercial engines, and is available throughout the world.

Type A kerosene can be used in the JTF17 engine, in spite of its higher operating temperature, because the engine is designed to keep fuel temperatures into the burners at essentially the same level as the current subsonic JT3D engine. As a result, the same degree of success in preventing varnish deposits in the fuel manifolds and nozzles will be obtained in this engine as with current P&WA commercial engines, which have achieved 8000 hr TBO. The fuel temperature into the burner system is maintained at current commercial engine levels, in spite of the significantly higher fuel inlet temperature of the SST, because features are incorporated to keep the fuel temperature rise from the engine inlet to the burner system to a minimum. These features, which are described above, involve reduced heat rejection from the oil to the fuel, and reduced heat transfer from ambient air into the fuel in the lines and manifolds. To keep internal engine fuel temperature below the design limits, fuel temperature into the engine must be kept within the limits specified in figure 5.

If aircraft thermal management problems make it desirable, or necessary, to operate the engine with higher fuel inlet temperature, fuel conforming to the requirements of PWA 533 specification shown on table 2, may be used. This fuel is the same as commercial Jet A aviation kerosene in all significant respects except that the thermal stability requirements are increased 50°F from 300°/400°F to 350°/450°F as measured in an ASTM-CRC coker. This increase in thermal stability permits a 50°F increase in fuel temperature into the engine without reduction in service life. Fuel temperature limits into the engine that may be used with PWA 533 fuel are shown in figure 6. This fuel is expected to be available to the same extent, and at the same price, as commercial aviation kerosene by the time the SST becomes operational. Ninety percent of all domestic fuels now delivered for current subsonic jet aircraft meet the requirements of this specification, as shown in figure 12.

Background data are available that indicates that de-oxygenation of type Jet A fuel, which can be accomplished in the aircraft in flight through the use of special tank management procedures to control pressure, temperature, and agitation, will result in a significant increase in thermal stability. If it can be demonstrated that such de-oxygenation can be accomplished, and that this procedure results in fuel conforming to specification PWA 533 being delivered to the engine when the aircraft is fueled with aviation Jet A kerosene, it will be permissible to use the higher engine inlet temperatures now specified for PWA 533 fuel with Jet A fuel.

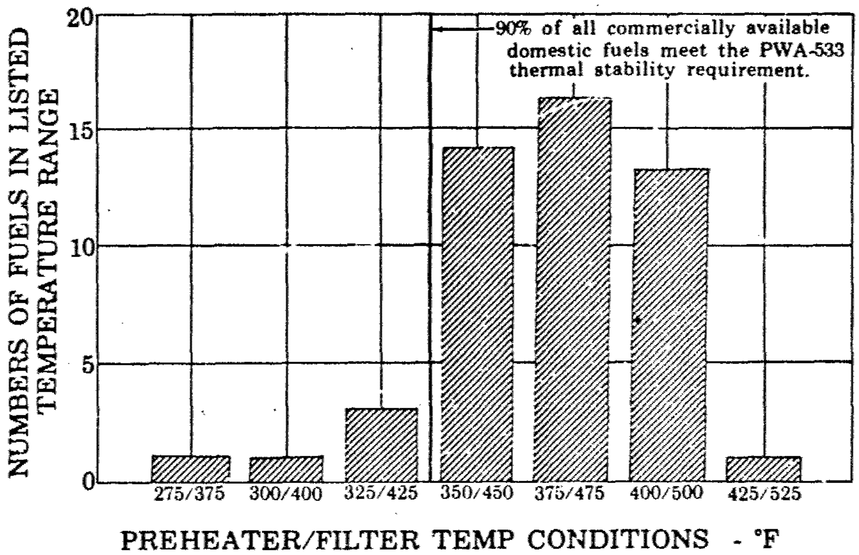

Figure 12. Results of Pratt & Whitney Aircraft 1964 Fuel Thermal Stability Survey FD 16993 BIV

2. Fuel Specifications

PWA Specification PWA 522 (ASTM D1655) for commercial aviation kerosene are shown on table 1.

P&WA Specification PWA 533 for aviation kerosene with improved thermal stability requirements is shown on table 2.

Table 1. Aircraft Turbine Engine Fuel (PWA 522) Specification

Pratt & Whitney Aircraft — DIVISION OF UNITED AIRCRAFT CORPORATION	P&WA SPECIFICATION	PWA 522-E
		ISSUED 4/10/54
		REVISED 11/5/65

FUEL, COMMERCIAL AIRCRAFT TURBINE ENGINE

1. ACKNOWLEDGMENT: Vendor shall mention this specification number and its revision letter in all quotations and when acknowledging purchase orders.

2. PURPOSE: To establish requirements for JP4 and kerosene type fuels for use in Pratt & Whitney Aircraft commercial turbine engines.

3. TECHNICAL REQUIREMENTS: Tests shall be performed, insofar as practicable, in accordance with the latest issue of the listed ASTM test methods.

3.1 Properties:

Gravity, deg API	37	57	ASTM D287
Distillation Temperature, F			ASTM D86
10% Evaporated	max	400	
90% Evaporated	max	500	
End Point	max	572	
Loss, %	max	1.5	
Residue, %	max	1.5	
Sulfur, %	max	0.30	ASTM D1266
Mercaptan Sulfur, % by weight (See Note 1)	max	0.005	ASTM D1323 or D1219
Potential Gum, mg per 100 ml (16 hr)	max	14.0	ASTM D873
Net Heat of Combustion, Btu per lb	18,400	min	ASTM D240
Freezing Point, F	See Note 2		ASTM D2386
Reid Vapor Pressure, lb	max	3	ASTM D323
Aromatic Content, % by volume	max	20	ASTM D1319
Burning Quality			
Luminometer Number	45 min		ASTM D1740 (See Note 3)
Copper Strip Corrosion	Slight discoloration permitted		ASTM D130
Viscosity, Cs at -30 F	max	15	ASTM D445
Water Tolerance, ml	max	2	ASTM D1094
High Temperature Stability			
Pressure Change, in. Hg	max	12	ASTM D1660 (See Note 4)
Preheater deposit	max	Code 2	

Note 1. Mercaptan sulfur determination may be omitted provided Doctor Test in accordance with ASTM D484 is conducted and results are negative.

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Table 1. Aircraft Turbine Engine Fuel (PWA 522) Specification (Continued)

Note 2. The requirements for this property shall be determined by the user and shall appear on all purchase orders for this fuel. The freezing point shall be at least 6 F below the minimum engine fuel inlet temperature.

Note 3. Fuels will be acceptable provided they meet one of the following alternative requirements or combination of requirements:

a. Smoke point of not less than 25 mm when determined in accordance with ASTM Method D1322.

b. Smoke point of not less than 20 mm when determined in accordance with ASTM Method D1322 provided fuel does not contain more than 3.0% by volume of naphthalene as determined by ultra violet spectrometry in accordance with Federal Test Method Standard 791, Method 3704T.

c. Smoke Volatility Index of not less than 54. The smoke volatility index is determined from the following equation:

SVI = smoke point (D1322) + 0.42 x volume % boiling under 400 F (D86)

Note 4. The high temperature stability property shall be measured in an ASTM-CFR FUEL COKER after 5 hr of operation at conditions of 300 F preheater temperatures, 400 F filter temperature, and 6 lb per hr fuel flow rate.

3.2 Additives:

3.2.1 One or a combination of the following inhibitors may be added to the basic fuel in total concentration not greater than 1.0 lb of inhibitor, not including weight of solvent, per 5000 gal of fuel, to prevent the formation of gum:

N, N' -di-secondary-butyl-para-phenylenediamine
2, 4-dimethyl-6-tertiary-butyl phenol
2, 6-ditertiary-butyl-4-methyl phenol
2, 6-ditertiary-butyl phenol
75% 2, 6-ditertiary-butyl phenol
10-15% 2, 4, 6-tritertiary-butyl phenol
10-15% ortho-tertiary butyl phenol

3.2.2 The use of Pratt & Whitney Aircraft approved additives such as corrosion inhibitors, metal deactivators, anti-icing inhibitors, and additives to extend the high temperature stability is permitted.

4. QUALITY:

4.1 Fuel shall consist solely of hydrocarbon compounds except as otherwise specified herein. It shall be free from water, sediment, and suspended matter, and shall be suitable for use in aircraft turbine engines.

4.2 The odor of the fuel shall not be nauseating or irritating. No substances of known dangerous toxicity under usual conditions of handling and use shall be present.

5. CONTROL: Control of quality and control of shipments shall be in accordance with the latest issue of PWA Specification 300.

6. REJECTIONS: Fuel not conforming to this specification or to authorized modifications will be subject to rejection.

NOTE: Sections 1 and 5 apply only to Pratt & Whitney Aircraft operation.

Table 2. Aircraft Turbine Engine Fuel (PWA 533) Specification

Pratt & Whitney Aircraft — DIVISION OF UNITED AIRCRAFT CORPORATION	P&WA SPECIFICATION	PWA 533
		ISSUED 11/10/64
		REVISED

Fuel, Aircraft Turbine Engine

1. ACKNOWLEDGMENT: Vendor shall mention this specification number in all quotations and when acknowledging purchase orders.

2. PURPOSE: To establish requirements for fuel for use in high Mach number Pratt & Whitney Aircraft commercial turbine engines.

3. TECHNICAL REQUIREMENTS: Tests shall be performed, insofar as practical, in accordance with the latest issue of the listed ASTM test methods.

3.1 Properties:

Gravity, deg API	37 - 57	ASTM D287
Distillation Temperature, F		
10% Evaporated	max 400	ASTM D86
50% Evaporated	max 450	
Final Boiling Point	max 550	
Loss, %	max 1.5	
Residue, %	max 1.5	
Freezing Point, F	See Note 1	ASTM D1477
Viscosity, Cs at -30 F	max 15	ASTM D445
Net Heat of Combustion, Btu per lb	18,400 min	ASTM D240
Copper Strip Corrosion at 212 F	Slight discoloration permitted	ASTM D130
Sulfur, %	max 0.3	ASTM D1266
Mercaptan Sulfur, % by wt (See Note 2)	max 0.003	ASTM D1323 or D1219
Water Tolerance, ml	max 1	ASTM D1094
Existent Gum, mg per 100 ml	max 7	ASTM D381
Burning Quality (Luminometer Number) (See Note 3)	45 min	ASTM D1740
Aromatics, % vol	max 20	ASTM D1319
High Temperature Stability		ASTM D1660 (See Note 4)
Pressure Change, in. Hg	max 12	
Preheater Deposit	max Code 2	

E-4077

E.O. 665 NEW 9-64

Table 2. Aircraft Turbine Engine Fuel (PWA 533) Specification (Continued)

Note 1. Freezing point shall be not less than 10 F below the minimum engine fuel inlet temperature. Actual freezing point requirement will be determined by user and will appear on all purchase orders for this fuel.

Note 2. Mercaptan sulfur determination may be omitted provided doctor test in accordance with ASTM D484 is conducted and results are negative.

Note 3. Fuels will be acceptable provided they meet one of the following alternative requirements or combination of requirements:

a. Smoke point of not less than 25 mm when determined in accordance with ASTM D1322.

b. Smoke point of not less than 20 mm when determined in accordance with ASTM D1322 provided fuel does not contain more than 3.0% by vol of naphthalene as determine by ultra violet spectrometry in accordance with Federal Test Method Standard 791, Method 3704T.

c. Smoke point of not less than 20 mm when determined in accordance with ASTM D1322 providing the following requirements of the 16 hr burning test as defined in ASTM D187 are also met:

1. After the first weighing, the rate of burning shall be not greater than 29 ml per hr using the IP burner or 45 ml per hr using the ASTM 27 burner.

2. The lamp chimney shall be clean or only slightly clouded at the end of the test.

3. The wick shall not have appreciable hard incrustation at the end of the test.

4. The flame shall not be smoky and shall be practically as large at the end of the test as when the final adjustment of the wick was made.

Note 4. The high temperature stability shall be measured in an ASTM-CFR Fuel Coker after 5 hr of operation at conditions of 350 F preheater temperature, 450 F filter temperature, 6 lb per hour flow rate or in an equivalent acceptable test device.

3.2 Additives:

3.2.1 One or a combination of the following inhibitors may be added to the basic fuel in total concentration not greater than 1.0 lb of inhibitor, not including weight of solvent, per 5,000 gal of fuel, to prevent formation of gum:

N, N' -di-secondary-butyl-para-phenylenediamine
2, 4-dimenthyl-6-tertiary-butyl phenol
2, 6-ditertiary-butyl-4-methyl phenol
2, 6-ditertiary-butyl phenol
75% 2, 6-ditertiary-butyl phenol
10-15%, 2, 4, 6-tritertiary-butyl phenol
10-15% ortho-tertiary butyl phenol

Table 2. Aircraft Turbine Engine Fuel (PWA 533) Specification (Continued)

3.2.2 Use of Pratt & Whitney Aircraft approved additives such as corrosion inhibitors, metal deactivators, and additives to extend high temperature stability, lubricity, etc., will be permitted.

4. QUALITY:

4.1 Fuel shall consist solely of hydrocarbon compounds except as otherwise specified herein. It shall be free from water, sediment, and suspended matter, and shall be suitable for use in aircraft turbine engines.

4.2 Odor of the fuel shall not be nauseating or irritating. No substance of known dangerous toxicity under usualy conditions of handling and use shall be present.

5. CONTROL: Control of quality and control of shipment shall be in accordance with the latest issue of PWA Specification 300.

6. REJECTIONS: Fuel not conforming to this specification or to authorized modifications will be subject to rejection.

Note. Sections 1 and 5 apply only to Pratt & Whitney Aircraft operation.

3. Projection of Costs and Availability

As stated above, the JTF17 engine is designed to operate with commercial aviation kerosene which is currently being used in subsonic jet aircraft, and is available throughout the world. Results of preliminary coordination with fuel suppliers indicate that the alternative fuel with higher thermal stability characteristics (PWA 533) can be made available on the same basis with no increase in cost once the SST aircraft becomes operational and the demand for such fuel is high enough to warrant its production and distribution. As stated above, 90% of all domestic fuels now used in commercial jet aircraft meet the requirements of this specification. This coordination effort with the fuel suppliers will be continued throughout the Phase III and IV portions of this program.

E. LUBRICANTS

1. General

The JTF17 engine is designed to operate on currently available type II lubricants that meet the requirements of Specification PWA 521B. These type II lubricants are currently being used by many airlines and are generally available. Their use in the JTF17 engine will not increase lubricant cost per unit volume over those now in effect.

Advanced lubricants designed to reduce direct operating costs through improved lubrication characteristics and increased time between overhaul are in the process of development, and will ultimately be made available for service use. Work on those is described in the Test and Certification Plan, Volume III, Report E, Section II.

2. Lubricant Specification

PWA Specification PWA 521B for Type II lubricant to be used in the JTF17 engine is shown in table 3.

F. PHASE II-C TEST RESULTS

1. Fuel Systems

Results of Phase II testing of fuel system and fuel systems components are discussed in Volume III, Report B, Section III and Volume III, Report E.

2. Lubrication System

Results of Phase II testing of the lubrication system and lubrication system components is discussed in Volume III, Report E.

G. PLANS FOR PHASE III TESTS

1. Fuel Systems

Plans for conducting tests on fuels, fuel systems, and fuel system components in Phase III are outlined in Volume III, Report E.

2. Lubrication System

Plans for conducting tests on lubricants, lubricant systems and lubrication system components in Phase III are outlined in Volume III, Report E.

H. FUEL AND LUBRICATION SYSTEM HEAT BALANCE ANALYSIS

Because of high environmental temperatures encountered at supersonic flight conditions, the fuel and lubricacion system heat transfer characteristics of the engine must be thoroughly analyzed to ensure that maximum use is made of the fuel as a heat absorption source and that fluid and component operating temperatures can be maintained within acceptable limits at all flight conditions and engine power levels. To accomplish this, an analytical heat balance analysis has been completed on the JTF17 engine. This analysis was conducted for all flight conditions within the operating envelope and at all engine power levels.

The results of the analysis shows that the maximum operating fuel and oil temperatures are within the required temperature limits for all flight conditions throughout the flight envelope and that fuel will not be returned to the aircraft tank at any flight condition except during descent after sustained high Mach number cruise.

The JTF17 heat balance analysis techniques are the same as used during the J58 engine development program. These methods have been proven highly accurate by comparing analytical results with test data obtained on the J58 engine operating at high Mach numbers and altitudes.

Table 3. Aircraft Turbine Engine Lubricant (PWA 521-B) Specification

E-3285

Issued 5/1/51
Revised 1/9/52
Revised 6/25/63

PRATT & WHITNEY AIRCRAFT SPECIFICATION

PWA 521-B

LUBRICANT, AIRCRAFT TURBINE ENGINE

1. ACKNOWLEDGMENT: Vendor shall mention this specification number in all quotations and when acknowledging purchase orders.

2. TYPE: Blend of synthetic and/or mineral oils capable of meeting certain performance requirements as measured by special tests.

3. TECHNICAL REQUIREMENTS:

3.1 ø Composition of lubricant is not limited; formulations composed substantially or entirely of non-petroleum materials are permitted, provided the following requirements are met. Tests shall be performed, insofar as practicable, in accordance with the latest issue of the listed ASTM test methods. Properties marked with an asterisk (*) are applicable only when this specification number followed by "Type II" is specified on the purchase order.

	Specific Gravity		
	At 60/60 F	To be reported	ASTM D1298
ø	Viscosity, Kinematic, Cs		ASTM D445
	At -40 F	max 13,000	
	At 100 F	max 100	
	At 210 F	max 5.5	
	At 400 F	1.0 min	
	Pour Point, F	max -75	ASTM D97
ø	Flash Point, F	400 min	ASTM D92
ø	Evaporation Loss after 6-1/2 hr, %		ASTM D972
	Sea Level (29.9 in. Hg) 400 F	max 25	
	40,000 Feet (5.5 in. Hg abs) 450 F	max 50*	
ø	Foaming, ml (volume), max		ASTM D892
	Sequence 1	25 - 0	
	Sequence 2	25 - 0	
	Sequence 3	25 - 0	

Table 3. Aircraft Turbine Engine Lubricant (PWA 521-B) Specification (Continued)

E-3285 -2- PWA 521-B

ø	Bearing Test, Condition 2 (100 hr)	Pass*	See Note 1
ø	Carbon Seal Wear Test	Pass*	See Note 1
ø	Vapor Phase Coking Test	Pass*	See Note 1
ø	Gear Scuffing Load, lb per in.	1,900 min 2,400 min*	See Note 1
ø	Pitting Fatigue, hr to failure	100 min*	See Note 1
ø	Corrosion-Oxidation Stability, Weight change, mg per sq cm	After 72 hr at 347 F ± 5	See Note 1 After 48 hr at 425 F ± 5
	Steel) Silver) Copper) Aluminum) Titanium) Magnesium)	max ± 0.30	max ± 0.30*
	Viscosity at 100 F, % change from original	-5 to +15	max +50
	Total Acid Number, change from original	max 2.0	To be reported
ø	SOD Lead Corrosion, weight loss, mg per sq. in. 1 hr at 325 F ± 5 5 hr at 375 F ± 5	max 6.0 max 6.[illegible]*	See Note 1
ø	Refractive Index	To be reported	ASTM D1218

Note 1. Test method shall be acceptable to Pratt & Whitney Aircraft.

4. QUALITY: Lubricant shall be free from suspended matter, grit, water, and objectionable odor.

5. CONTROL: Control of quality and control of shipments shall be in accordance with the latest issue of PWA Specification 300.

6. APPROVAL:

6.1 Vendor shall not supply lubricant to this specification until samples have been approved by purchaser. After approval, compounds and method of manufacture shall not be changed without notification to purchaser prior to first shipment of lubricant embodying such change. To qualify for batch acceptance, results of tests on subsequent lubricant lots shall be essentially equal to those on approved samples, except that the bearing test and the pitting fatigue will not be required for batch acceptance.

6.2 Right is reserved to submit oils of new or unusual composition to such additional tests as are considered necessary to assure serviceability of the material.

7. REJECTIONS: Lubricants not conforming to this specification or to authorized modifications will be subject to rejection.

NOTE: Sections 1 and 5 apply only to Pratt & Whitney Aircraft operation.

The basic approach in the heat balance analysis is to calculate the heat input to the fuel and lubricant from all heat sources, and to calculate the heat returned to the aircraft fuel tanks during descent.

The engine heat energy sources fall into four general categories:

1. Ambient or environmental - Heat flow into the fluid due to conduction, convection, and radiation.
2. Flow inefficiency - Temperature rise associated with pressure drop in an incompressible fluid.
3. Component Efficiency - Temperature rise in a fluid associated with pump inefficiency.
4. Mechanical friction - Heat input to a fluid associated with rotating and stationary interfaces such as gears, bearings, and seals.

The contribution of each of these sources to the total engine fuel system is shown for a typical supersonic cruise condition and airframe installation in table 4.

A listing of individual mechanical and ambient heat energy sources and heat transfer rates for a typical supersonic flight condition are shown in table 5.

Table 4. Total Heat Transferred to Fuel at Mach 2.7 at 65,000 feet Altitude for a Typical Installation

Heat Energy Source	Heat Transfer Rate, Btu/min
Main Burner Fuel System	
Ambient, Components	60
Ambient, Plumbing	237
Ambient, Main Manifold	814
Pump, Impeller Stage	646
Pump, Gear Stage	420
Flow Inefficiency	488
Hydraulic System	
Ambient, Components	44
Ambient, Plumbing	401
Pump	950
Flow Inefficiency	553
Duct Heater Fuel System	
Ambient, Components	41
Ambient, Plumbing	167
Ambient, Duct Manifold	834
Turbopump	1000
Heat Transferred from Lubrication System Thru the Heat Exchangers	
Ambient	740
Mechanical	2855
Total	10,250

Table 5. Mechanical and Ambient Heat Energy Sources and Heat Transfer Rates to the Lubricant at Mach 2.7 and 65,000 ft. Altitude for a Typical Installation

Heat Energy Source	Heat Transfer Rate, Btu/min
1. Mechanical	
No. 1 and No. 2 Bearing Compartment	
Seals	241
No. 1 Thrust Bearing	123
No. 2 Thrust Bearing	556
Pinion Gear, Aircraft Power Takeoff Gearbox	297
Pinion Gear, Main Accessory Gearbox	89
No. 3 Bearing Compartment	
Seals	67
Roller Bearing	268
No. 4 Bearing Compartment	
Seals	40
Roller Bearing	50
Gearboxes	
Main Accessory Gearbox	281
Aircraft Power Takeoff Gearbox	548
Oil Pump Gearbox	295
Sub Total	2855
2. Ambient	740
Total	3595

To fully analyze the system for all aircraft flight and engine power conditions, the characteristic sources of heat energy were developed as a function of measurable parameters such as rotor speed, fuel flow, secondary airflow, and engine inlet temperature and pressure.

The characteristic data were developed in the form of heat rejection maps, which were plotted against one or more of the above-mentioned parameters. An example of this mapping is shown in figure 13, which shows the total lubrication system mechanical friction heat input at full power conditions plotted against flight Mach number and altitude.

Testing of the fuel and lubrication system has been performed on the Phase II-C initial experimental engines. This provides a source of data that is being used to support this analytical work. An example of the resulting test data is shown in figure 14, which compares calculated-to-measured engine heat rejection. The data are obtained by measuring oil and fuel flow rates and fluid temperatures at both coolers. These data verify that the techniques used to account for the total heat from all energy sources are accurate.

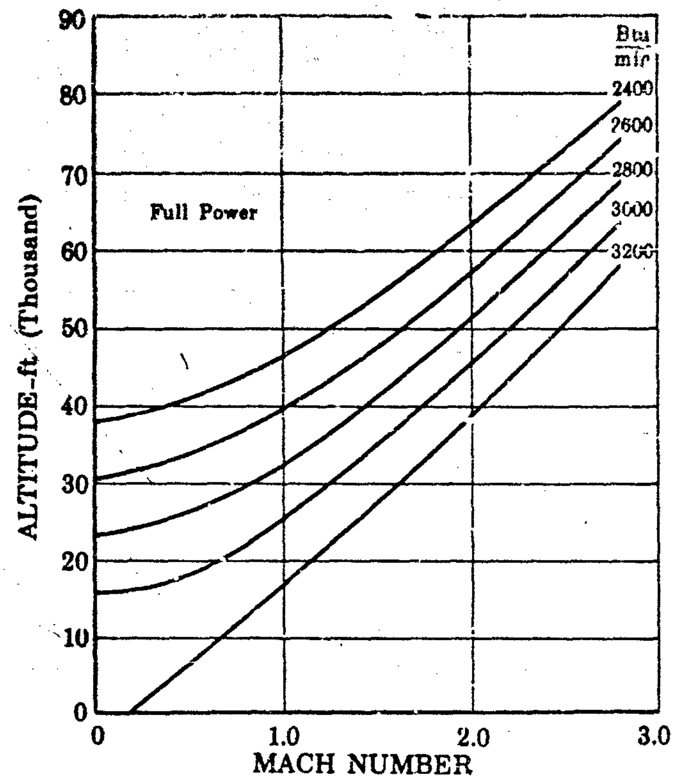

Figure 13. Mechanical Heat Rejection Map FD 17173 BIV

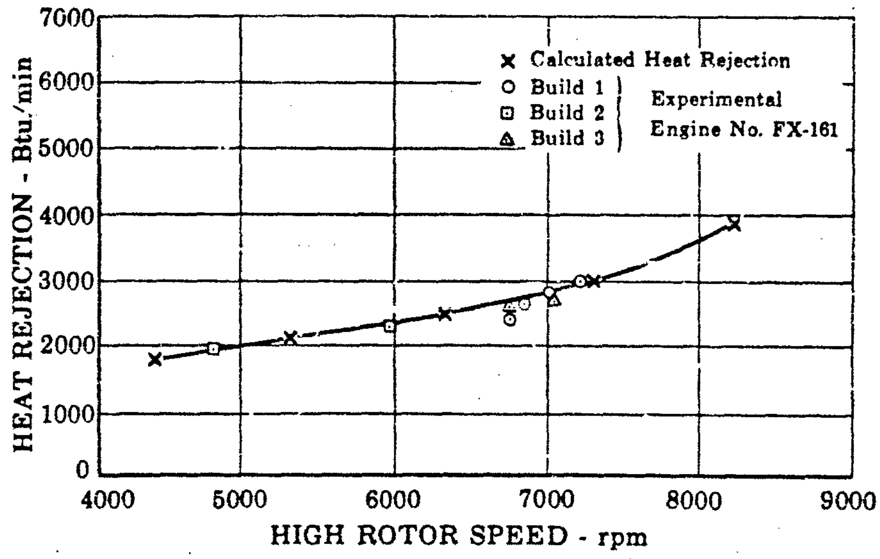

Figure 14. Comparison of Calculated and Measured Total Engine Heat Rejection for Initial Experimental Engine FD 17638 BIV

Although individual heat input rates to components are calculated separately, the integrated fuel, hydraulic, and lubrication systems require detailed analysis to calculate steady-state and transient fluid temperatures. Fluid temperatures were calculated at each junction of two or more flow streams using appropriate mixture equations, and influenced by J58 engine development experience where fluid temperatures are affected uniquely by the hardware configuration.

Such a detailed analysis was made for all critical combinations of engine and aircraft operating conditions to confirm that the system designed for the JTF17 engine is satisfactory throughout the entire flight envelope. Typical results are shown on figures 1,2,3, and 4. This analysis showed that both fuel and oil temperatures will be maintained below the point at which experience shows that varnish deposits might occur in the fuel systems and coking might occur in the oil system at all flight conditions. The heat rate which must be returned back to the aircraft tank during descent when the heat absorbing capacity of the fuel being burned is low was also calculated as a part of this analysis. Figure 15 shows typical results of a calculation of the heat return rate during descent. The curves show how the heat return rate is influenced by fuel inlet temperature and time.

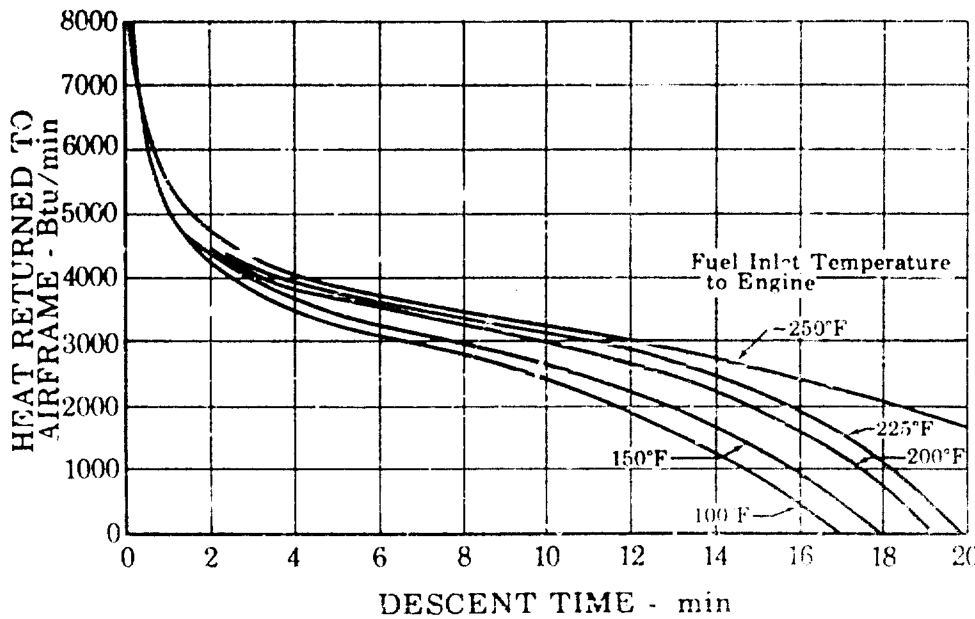

Figure 15. Typical Plot of Heat Return Rate to Airframe During Descent FD 17639 BIV

I. PRODUCT ASSURANCE CONSIDERATIONS

Quality assurance and inspection procedures, reliability procedures, human engineering, and value engineering procedures that will be used in the design, development and manufacture of the fuel and lubrication system components are described in detail in Volume IV, Report F.

Specific maintainability, reliability, and safety advantages are listed in the following paragraphs.

1. Maintainability and Servicing

   1. All components subject to possible servicing are located outside of the engine.
   2. Oil contaminant collection and detection units, such as filters and chip detectors, are located outside of the engine at easily accessible locations.
   3. Servicing of the oil system will be held to a minimum by maintaining low oil consumption through the use of a highly efficient de-oiler.
   4. A valve is located at the bottom of the oil tank to facilitate draining and flushing. These operations can be performed without removing the tank or main oil pump supply line. The tank assembly also includes an easily accessible filling and overflow fitting.
   5. A filter is incorporated in the bearing compartment seal pressurization supply air system to prevent dirt accumulation in the lubricant.

2. Reliability

   1. Pressure operated bypass valves will be incorporated on all components to ensure sustained system cooling and lubrication in the event of flow blockage within the component.
   2. Two oil coolers will be used, each with the capacity to maintain acceptable oil temperatures in the event of a single cooler or bypass valve malfunction.
   3. The redundant oil filtering system is composed of a primary filter, standby filter and pressure operated bypass valves for each.
   4. Fuel dump valves will be used in the primary cumbustor and duct heat fuel supply systems. These valves allow the residual fuel to be drained when the combustor or heater are not operating. This feature prevents fuel decomposition.
   5. The system uses a regulated oil system that maintains constant flow and pressure at all engine speeds.
   6. A pressure relief valve is used within the main breather pressurizing valve to prevent high breather pressure.
   7. Thermostatically operated bypass valves are used in both fuel systems to regulate the nozzle fuel temperature to levels below the decomposition point.
   8. Additional foreign object protection for the bearings is provided by passing oil through externally located "last chance" filters prior to reaching the oil jets in each cooling and lubrication point.

3. Safety

   1. Prevention against possible auto-ignition of the oil is achieved by providing a low flow of cool air to pressurize the breather system.
   2. Protection against oil fires outside bearing compartments is achieved through the incorporation of a unique venting system. Any oil vapor leakage past the primary compartment seal is vented overboard.

3. The venting system also prevents possible primary seal oil vapor leakage from entering the cabin air pressurization supply. Details of the vent system are described in section G.
4. An oil pressure and temperature transmitter is provided for continual cockpit monitoring of oil conditions.

J. MATERIALS SUMMARY

Table 6 shows a summary of all lubrication system components, the materials used, and the primary reason for the material selection. Materials used in fuel and lubrication system lines are included in the lines and fittings section. Materials used in the fuel system are included in the controls and accessories section.

Table 6. Lubrication System Component Material Summary

Component	Material	Substantiation of Material Selection
Oil Tank	AMS 5510 (AISI 321)	Proved durability on both commercial and J58 engines.
Main Oil Pump and Scavenge Pumps	Gears - PWA 724 - (Alloy Steel)	Low wear (J58 experience).
	Housing -PWA 4134 (Aluminum)	Ability to pass contaminants without jamming pumps (Commercial, TF30)
	Journal Inserts AMS 6322 (Alloy Steel)	Exceptionally low wear - TF30 high pressure pump.
	Plating-Lead/ Indium/Silver	Low wear characteristics.
Main Filters	Stainless Steel Mesh	Proved durability in J58 engine.
Oil Cooler	Housing - AMS 5646 (AISI 347)	Proved durability, weldability, and high ductility on commercial and J58 engines.
	Core - AMS 5646 (AISI 347)	Proved durability in J58 engine.

Table 6. Lubrication System Component Material Summary (Cont'd)

Component	Material	Substantiation of Material Selection
Duct Cooler Thermal Bypass Valve	Housing - AMS 5646 (AISI 347)	Proved durability on commercial and J58 engines.
	Liner - AMS 5616 (Greek Ascoloy)	Low thermal expansion; low wear from J58 experience.
	Valve - AMS 5616 (Greek Ascoloy)	Low thermal expansion; low wear from J58 experience.
	Spring - AMS 5698 (Inconel X)	Proved durability, corrosion resistance, and resistance to relaxation at high temperature.
Pressure Regulating Valve	AMS 5613 (AISI 410)	Low thermal expansion, low wear, proved durability on commercial engines.
Duct Cooler Fuel and Oil Pressure Bypass Valves	Housing - AMS 5646 (AISI 347)	Proved durability on commercial and J58 engines.
	Valve - AMS 5646	Low thermal expansion; low wear from J58 experience.
	Spring - AMS 5698 (Inconel X)	Proved durability, corrosion resistance, and resistance to relaxation at high temperature.
De-Oiler	AMS 5350 (AISI 410 Precision Cast)	Easy casting, proved durability on J58 engines.
Breather Pressurizing Valve	Housing - AMS 4921 (Comm. Pure Titanium)	Weight
	Valve - AMS 5613 (AISI 410)	Low wear
	Bellows - AMS 5512 (AISI 347)	Formability, and proved durability on commercial and J58 engines.